CHAPTER 2A - PRODUCTION AIRCRAFT SYSTEMS
LIST OF EFFECTIVE PAGES

* Pages revised by Revision 22
** Pages added by Revision 22
*** Pages deleted by Revision 22

ATA	PAGE	DATE
2A-21-00	9	January 23/04
2A-21-00	10	January 23/04
2A-21-00	11 / 12	January 23/04
2A-21-00	13 / 14	January 23/04
2A-21-00	15 / 16	January 23/04
2A-21-00	17	January 23/04
2A-21-00	18	January 23/04
2A-21-00	19	January 23/04
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2A-21-00	30	January 23/04
2A-21-00	31 / 32	January 23/04
2A-21-00	33 / 34	January 23/04
2A-21-00	35 / 36	January 23/04
2A-21-00	37	January 23/04
2A-21-00	38	January 23/04
*2A-23-00	1	May 13/08
*2A-23-00	2	May 13/08
*2A-23-00	3 / 4	May 13/08
*2A-23-00	5	May 13/08
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*2A-23-00	11	May 13/08
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*2A-23-00	17	May 13/08
*2A-23-00	18	May 13/08
*2A-23-00	19	May 13/08
*2A-23-00	20	May 13/08
*2A-23-00	21 / 22	May 13/08
*2A-23-00	23 / 24	May 13/08

* Pages revised by Revision 22
** Pages added by Revision 22
*** Pages deleted by Revision 22

ATA	PAGE	DATE
*2A-23-00	25 / 26	May 13/08
*2A-23-00	27 / 28	May 13/08
*2A-23-00	29	May 13/08
*2A-23-00	30	May 13/08
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*2A-23-00	33	May 13/08
*2A-23-00	34	May 13/08
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*2A-23-00	37	May 13/08
*2A-23-00	38	May 13/08
*2A-23-00	39	May 13/08
*2A-23-00	40	May 13/08
*2A-23-00	41 / 42	May 13/08
*2A-23-00	43	May 13/08
*2A-23-00	44	May 13/08
*2A-23-00	45 / 46	May 13/08
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2A-24-00	2	September 18/07
2A-24-00	3	September 18/07
2A-24-00	4	September 18/07
2A-24-00	5 / 6	September 18/07
2A-24-00	7	September 18/07
2A-24-00	8	September 18/07
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2A-24-00	17	September 18/07
2A-24-00	18	September 18/07
2A-24-00	19	September 18/07
2A-24-00	20	September 18/07

* Pages revised by Revision 22
** Pages added by Revision 22
*** Pages deleted by Revision 22

ATA	PAGE	DATE
2A-24-00	21	September 18/07
2A-24-00	22	September 18/07
2A-24-00	23	September 18/07
2A-24-00	24	September 18/07
2A-24-00	25 / 26	September 18/07
2A-24-00	27 / 28	September 18/07
2A-24-00	29 / 30	September 18/07
2A-24-00	31 / 32	September 18/07
2A-24-00	33 / 34	September 18/07
2A-24-00	35 / 36	September 18/07
2A-24-00	37 / 38	September 18/07
2A-24-00	39 / 40	September 18/07
2A-24-00	41 / 42	September 18/07
2A-24-00	43 / 44	September 18/07
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2A-24-00	70	September 18/07
2A-24-00	71 / 72	September 18/07
2A-24-00	73 / 74	September 18/07
2A-24-00	75 / 76	September 18/07
2A-24-00	77	September 18/07

*** Pages revised by Revision 22**
**** Pages added by Revision 22**
***** Pages deleted by Revision 22**

ATA	PAGE	DATE
2A-24-00	78	September 18/07
2A-24-00	79	September 18/07
2A-24-00	80	September 18/07
2A-26-00	1	January 31/02
2A-26-00	2	January 31/02
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2A-26-00	7	January 31/02
2A-26-00	8	January 31/02
2A-26-00	9	January 31/02
2A-26-00	10	January 31/02
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2A-26-00	13	January 31/02
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2A-27-00	5 / 6	February 1/07
2A-27-00	7	February 1/07
2A-27-00	8	February 1/07
2A-27-00	9	February 1/07
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2A-27-00	13	February 1/07
2A-27-00	14	February 1/07
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2A-27-00	16	February 1/07
2A-27-00	17 / 18	February 1/07
2A-27-00	19 / 20	February 1/07
2A-27-00	21 / 22	February 1/07
2A-27-00	23 / 24	February 1/07
2A-27-00	25	February 1/07

*** Pages revised by Revision 22**
**** Pages added by Revision 22**
***** Pages deleted by Revision 22**

ATA	PAGE	DATE
2A-27-00	26	February 1/07
2A-27-00	27	February 1/07
2A-27-00	28	February 1/07
2A-27-00	29	February 1/07
2A-27-00	30	February 1/07
2A-27-00	31	February 1/07
2A-27-00	32	February 1/07
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2A-27-00	36	February 1/07
2A-27-00	37 / 38	February 1/07
2A-27-00	39 / 40	February 1/07
2A-27-00	41	February 1/07
2A-27-00	42	February 1/07
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2A-27-00	73	February 1/07
2A-27-00	74	February 1/07
2A-27-00	75	February 1/07
2A-27-00	76	February 1/07

*** Pages revised by Revision 22**
**** Pages added by Revision 22**
***** Pages deleted by Revision 22**

ATA	PAGE	DATE
2A-27-00	77 / 78	February 1/07
2A-27-00	79 / 80	February 1/07
2A-27-00	81 / 82	February 1/07
2A-27-00	83	February 1/07
2A-27-00	84	February 1/07
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2A-27-00	86	February 1/07
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2A-28-00	31 / 32	May 22/07
2A-28-00	33 / 34	May 22/07
2A-28-00	35 / 36	May 22/07
2A-28-00	37 / 38	May 22/07
2A-28-00	39	May 22/07
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*2A-29-00	2	May 13/08
*2A-29-00	3 / 4	May 13/08
*2A-29-00	5	May 13/08

* Pages revised by Revision 22
** Pages added by Revision 22
*** Pages deleted by Revision 22

ATA	PAGE	DATE
*2A-29-00	6	May 13/08
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2A-30-00	6	June 1/06
2A-30-00	7 / 8	June 1/06
2A-30-00	9	June 1/06
2A-30-00	10	June 1/06

* Pages revised by Revision 22
** Pages added by Revision 22
*** Pages deleted by Revision 22

* Pages revised by Revision 22
** Pages added by Revision 22
*** Pages deleted by Revision 22

ATA	PAGE	DATE
2A-31-00	25	July 26/07
2A-31-00	26	July 26/07
2A-31-00	27	July 26/07
2A-31-00	28	July 26/07
2A-31-00	29 / 30	July 26/07
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2A-31-00	32	July 26/07
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2A-31-00	34	July 26/07
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2A-31-00	44	July 26/07
2A-31-00	45 / 46	July 26/07
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2A-32-00	14	May 22/07
2A-32-00	15	May 22/07
2A-32-00	16	May 22/07
2A-32-00	17	May 22/07
2A-32-00	18	May 22/07
2A-32-00	19	May 22/07
2A-32-00	20	May 22/07
2A-32-00	21	May 22/07

*** Pages revised by Revision 22**
**** Pages added by Revision 22**
***** Pages deleted by Revision 22**

ATA	PAGE	DATE
2A-32-00	22	May 22/07
2A-32-00	23	May 22/07
2A-32-00	24	May 22/07
2A-32-00	25 / 26	May 22/07
2A-32-00	27	May 22/07
2A-32-00	28	May 22/07
2A-32-00	29 / 30	May 22/07
2A-32-00	31	May 22/07
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2A-32-00	34	May 22/07
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2A-32-00	48	May 22/07
2A-32-00	49 / 50	May 22/07
2A-32-00	51 / 52	May 22/07
2A-32-00	53 / 54	May 22/07
2A-32-00	55 / 56	May 22/07
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2A-33-00	4	May 30/02
2A-33-00	5	May 30/02
2A-33-00	6	May 30/02
2A-33-00	7	May 30/02

*** Pages revised by Revision 22**
**** Pages added by Revision 22**
***** Pages deleted by Revision 22**

ATA	PAGE	DATE
2A-33-00	8	May 30/02
2A-33-00	9	May 30/02
2A-33-00	10	May 30/02
2A-33-00	11 / 12	May 30/02
2A-33-00	13 / 14	May 30/02
2A-33-00	15 / 16	May 30/02
2A-33-00	17 / 18	May 30/02
2A-33-00	19 / 20	May 30/02
2A-33-00	21	May 30/02
2A-33-00	22	May 30/02
2A-33-00	23	May 30/02
2A-33-00	24	May 30/02
2A-33-00	25 / 26	May 30/02
2A-33-00	27	May 30/02
2A-33-00	28	May 30/02
2A-33-00	29	May 30/02
2A-33-00	30	May 30/02
2A-33-00	31	May 30/02
2A-33-00	32	May 30/02
2A-33-00	33 / 34	May 30/02
2A-33-00	35 / 36	May 30/02
2A-33-00	37	May 30/02
2A-33-00	38	May 30/02
2A-33-00	39	May 30/02
2A-33-00	40	May 30/02
2A-33-00	41 / 42	May 30/02
2A-33-00	43	May 30/02
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2A-34-00	8	January 23/04
2A-34-00	9	January 23/04
2A-34-00	10	January 23/04
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2A-34-00	14	January 23/04
2A-34-00	15	January 23/04
2A-34-00	16	January 23/04
2A-34-00	17	January 23/04

*** Pages revised by Revision 22**
**** Pages added by Revision 22**
***** Pages deleted by Revision 22**

ATA	PAGE	DATE
2A-34-00	18	January 23/04
2A-34-00	19	January 23/04
2A-34-00	20	January 23/04
2A-34-00	21	January 23/04
2A-34-00	22	January 23/04
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2A-34-00	24	January 23/04
2A-34-00	25	January 23/04
2A-34-00	26	January 23/04
2A-34-00	27	January 23/04
2A-34-00	28	January 23/04
2A-34-00	29	January 23/04
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2A-34-00	62	January 23/04
2A-34-00	63	January 23/04
2A-34-00	64	January 23/04
2A-34-00	65	January 23/04

*** Pages revised by Revision 22**
**** Pages added by Revision 22**
***** Pages deleted by Revision 22**

ATA	PAGE	DATE
2A-34-00	66	January 23/04
2A-34-00	67	January 23/04
2A-34-00	68	January 23/04
2A-34-00	69	January 23/04
2A-34-00	70	January 23/04
2A-34-00	71 / 72	January 23/04
2A-34-00	73 / 74	January 23/04
2A-35-00	1	June 3/05
2A-35-00	2	June 3/05
2A-35-00	3	June 3/05
2A-35-00	4	June 3/05
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2A-80-00	4	May 31/01
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2A-80-00	9	May 31/01
2A-80-00	10	May 31/01

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2A-00-10: Introduction

1. **Purpose:**

   The purpose of this chapter is to provide detailed airplane system and subsystem information that has both operational and training significance. This chapter is composed of sections. The sections present the systems installed on Gulfstream IV airplanes, in an order reflecting Air Transport Association (ATA) Specification 100 guidelines. These sections, where applicable, are divided into subsections that reflect the subsystems that compose the systems.

2. **Presentation of Information (Section-Level System Descriptions):**

   Each section will begin with a section-level system description. It consists of a brief description of the system at its highest level. It will contain the following data, as applicable:

   - The purpose of the system with respect to the airplane.
   - A general description of the system.
   - The division of the section, i.e., the subsections (subsystems) that together compose the system.
   - Illustrations of control components and their location, if they are numerous and integrated within the overall system.

3. **Presentation of Information (Subsection-Level System Descriptions):**

   Following the section-level system description, the subsections (subsystem descriptions) are presented in the order outlined in the system-level system description. Each subsystem description, where applicable, is presented using the following topics:

   A. **General Description:**

   A brief description of the subsystem, with its material presented at a high level. It will contain the following data, as applicable:

   - The purpose of the subsystem with respect to the system and the airplane.
   - A general description of the subsystem.
   - The division of the subsection, i.e., the sub-subsystems (usually referred to in general terms as systems), units and/or components that together compose the subsystem.

   B. **Description of Subsystems, Units and Components:**

   A detailed description of the units and components that compose the subsystem, with applicable block diagrams.

   C. **Controls and Indications:**

   A detailed description of the controls of the subsystem, and the indications provided by the subsystem due to commanded or automatic operation. Accompanying the description are illustrations of the controls, indications and applicable displays.

   D. **Limitations:**

   The approved limitations from the Airplane Flight Manual (AFM) are repeated in this module to provide flight crews with a ready source of

limitations information directly related to the system being studied. Additional limitations, not included in the AFM, will also be presented in this module. Observance of these limitations will increase the service life of the airplane and prevent equipment damage.

DIMENSIONS AND AREAS, MAJOR COMPONENT LOCATIONS

2A-06-10: General

1. Aircraft:

The Gulfstream IV is a twin engine, swept wing, long range, high speed aircraft. It is a pressurized, transport category aircraft, certified to operate at altitudes up to 45,000 feet. Certification of the aircraft is based on Federal Aviation Regulation (FAR) Part 25.

Two Rolls-Royce Tay 611-8 high bypass ratio turbofan engines, mounted on pylons located on the aft upper fuselage, power the aircraft. Each engine produces 13,850 lb of rated takeoff thrust (5 minutes maximum) at sea level on a standard day.

An AiResearch GTCP36-100(G) Auxiliary Power Unit (APU) provides bleed air for air conditioning and engine starting, and AC power for the aircraft electrical power system.

2. Fuselage:

The fuselage is of semimonocoque metal construction, with five distinct sections spliced together to form a single structure. All areas are pressurized with the exception of the nose radome and aft equipment (tail) compartment. The fuselage is divided lengthwise into an above-floor section and a below-floor section. For the purposes of this manual, the fuselage is divided into six compartments as follows:

A. Nose Compartment:

The unpressurized nose compartment extends forward from the forward pressure bulkhead. It includes an avionics rack and a fiberglass honeycomb radome. In addition to its aerodynamic necessity, the radome serves to cover and protect the weather radar and glideslope antennas. Conductive strips on the radome provide an electrical bond to the aircraft to minimize lightning strike damage and prevent static accumulation.

B. Cockpit:

The cockpit extends from the forward pressure bulkhead to the entrance compartment. It includes a pressurized flight compartment and an unpressurized nose wheel well.

The cockpit is conventionally arranged and contains two flight crew stations with individually adjustable seats. The logical arrangement of each station allows easy access to all aircraft controls and displays. A two-piece, electrically heated, bird-proof and splinter-proof windshield and two fixed side windows enclose the cockpit.

See Section 2A-06-40, Flight Crew Station Components, for a description of the cockpit compartment.

C. Entrance Compartment:

The entrance compartment is located immediately aft of the cockpit bulkhead. It contains the left and right radio racks and main entrance door. See Section 2A-52-20, Main Entrance Door, for a description of the main entrance door.

D. Passenger Cabin:

The passenger cabin is pressurized and air conditioned. Depending on outfitting and configuration, the passenger cabin may seat up to 19

passengers.

Cabin outfitting is installed by either a furnishing agency or the airframe manufacturer according to the owner-operator specifications. The majority of the configurations, however, provide for a 30 inch cabin aisle, with galley and lavatory areas located aft of the main cabin area.

The location and number of cabin windows and emergency exits differs based upon whether or not an optional cargo door is installed. Cabin windows and emergency exits are discussed briefly in Section 2A-06-30, Entrances, Exits and External Access Doors, and in greater detail in Section 2A-52-30, Emergency Exits.

E. Baggage Compartment:

The baggage compartment is located immediately aft of the passenger cabin and galley area. The aft pressure bulkhead serves as the aft wall of the baggage compartment. An inward opening, plug-type baggage compartment door is installed on the left side. It may be opened from either inside or outside the aircraft. See Section 2A-52-40, Baggage Doors, for a description of the baggage door.

F. Tail Compartment:

The tail compartment includes all fuselage structure aft of the aft pressure bulkhead. Within this area are the APU, battery compartment and the following major system components:

- Bleed air ducting
- Air conditioning system components
- Battery chargers and converters
- Electrical junction boxes
- Engine and APU fire extinguisher bottles and lines
- Elevator and stall barrier control quadrant
- Horizontal stabilizer dwell box assembly
- Utility hydraulic system pump
- Combined and Flight hydraulic system reservoirs and accumulators
- Hydraulic reservoir replenishment system
- Engine oil replenishment system

The tail compartment is accessed through an outward opening door on the lower rear fuselage. A removable telescoping ladder is attached to the door. The ladder can also be removed and used to reach engine areas for inspection and maintenance. See Section 2A-52-50, Tail Compartment Door, for a description of the tail compartment door.

3. Pylons:

The pylon assembly, located between the engine nacelle and fuselage, supports the engine and transmits thrust to the airframe. Shaped much like an airfoil, it contains passageways for engine services and controls. In addition, each pylon contains a precooler assembly with bypass ducting and a control valve that uses engine fan air to regulate High Pressure (HP) turbine bleed air temperature.

4. Powerplant:

The Gulfstream IV is powered by two Rolls-Royce Tay Mark 611-8 high bypass ratio turbofan engines.

The Tay Mark 611-8 turbofan engine is a two spool engine with a single-stage Low Pressure (LP) compressor (fan), a three-stage Intermediate Pressure (IP) compressor, and a twelve stage High Pressure (HP) compressor. Immediately following the HP compressor is a diffuser case and a turbo-annular combustion section composed of ten chambers. Hot gases produced in the combustion section drive a two-stage HP turbine and a three-stage LP turbine. The HP turbine drives the HP compressor while the LP turbine drives the IP and LP compressors.

Each engine has a hydraulically powered thrust reverser that directs engine thrust forward to reduce landing roll and brake wear.

Each engine is encased in a nacelle (or cowling) that includes a nose cowl that forms the engine air inlet, a fixed cowl on the inboard side of the engine, and upper and lower hinged cowls that allow access to the engine.

For detailed descriptions and illustrations of the engine, its accessories and systems, see Sections 2A-71-00 through 2A-80-00.

5. Stabilizer:

A fully cantilevered, sweptback vertical stabilizer is attached to the upper rear fuselage. The High Frequency (HF) antenna is built into the vertical stabilizer leading edge; the rudder is attached to the trailing edge on the rear spar. At the top of the rear spar is the horizontal stabilizer pivot point. The VOR/Localizer antennas are attached to the left and right sides of the vertical stabilizer.

A sweptback horizontal stabilizer is attached to the top of the vertical stabilizer. The horizontal stabilizer pivots at the vertical stabilizer rear spar, and an actuator in the forward upper portion of the vertical stabilizer drives the horizontal stabilizer leading edge up or down to compensate for flap extension or retraction. Elevators are attached to the trailing edge of the horizontal stabilizer.

6. Wing:

The Gulfstream IV wing is an all metal, fully cantilevered wing with a 3° dihedral and a sweepback of approximately 27°. Each wing consists of forward and rear spars covered by aluminum alloy skin panels. Ribs between the forward and rear spars carry airloads, define airfoil shape and act as fuel baffles for the integral fuel tanks. A sealed rib at the fuselage centerline and winglet attach point define the inboard and outboard confines of the wing fuel tank.

A Fowler-type flap and an aileron attach to each wing's rear spar. A hydraulically operated single panel ground spoiler and two flight spoiler panels attach to each wing's rear structure. When extended, the ground spoiler panels destroy lift and increase drag during landing rollout. The flight spoilers increase drag in flight to slow the aircraft.

A winglet is attached to the outboard edge of each wing. Winglets increase the efficiency of the wing by decreasing drag and improving airflow over the wing's upper surface. Vortex generators are installed on the wing's upper surface to improve high speed performance and smooth the airflow over the flight controls.

The main landing gear is attached to the wing structure between the forward and rear spars. When retracted, the landing gear is stored in a covered wheel well. Each main landing gear consists of two wheels with high speed tires. An anti-skid system provides maximum braking efficiency on all runway surfaces an surface

conditions.

7. **Subsections:**

The Dimensions, Areas and Major Component Locations section is divided into the following subsections:

- 2A-06-20: Principal Dimensions
- 2A-06-30: Entrances, Exits and External Access Doors
- 2A-06-40: Flight Crew Station Components

2A-06-20: Principal Dimensions

1. **Dimensions, Areas and Distances:**

See Figure 2 through Figure 5 for principal dimensions, areas and distances.

2. **General Weight Data:**

Airplane Serial Number:	Maximum Zero Fuel Weight:	Maximum Ramp Weight:	Maximum Takeoff Weight (1):	Maximum Landing Weight (2):
1000 - 1213 Without ASC 190 / ASC 61	46,500 lb (21,092 kg)	73,600 lb (33,385 kg)	73,200 lb (33,204 kg)	58,500 lb (26,536 kg)
1000 - 1213 With ASC 61	49,000 lb (22,226 kg)	73,600 lb (33,385 kg)	73,200 lb (33,204 kg)	58,500 lb (26,536 kg)
1000 - 1213 With ASC 190	49,000 lb (22,226 kg)	75,000 lb (34,020 kg)	74,600 lb (33,838 kg)	66,000 lb (29,937 kg)
1214 And Subs	49,000 lb (22,226 kg)	75,000 lb (34,020 kg)	74,600 lb (33,838 kg)	66,000 lb (29,937 kg)

NOTE(S):

(1) Maximum takeoff weight, unless restricted by climb performance, brake energy or tire speed for approved altitudes and ambient temperature or by field length. Refer to GIV Airplane Flight Manual Section 5, Performance.

(2) Maximum landing weight, unless restricted by climb requirements. Refer to GIV Airplane Flight Manual Section 5, Performance.

3. **General Powerplant Data:**

   A. **Manufacturer:** Rolls Royce
   B. **Model:** Tay Mark 611-8
   C. **Number Installed:** 2
   D. **Static Takeoff Thrust (Sea Level):** 13,850 lb (6282.4 kg)
   E. **Maximum Continuous Thrust:** 12,420 lb (5633.7 kg)
   F. **Engine Bypass Ratio:** 3.10:1
   G. **Engine Pressure Ratio:** 16.0:1

4. **External Component Locations:**

   A. **Antenna Locations:**

   Antenna locations are shown in Figure 1. For a description of the aircraft antennas, see Section 2A-23-00, Communications.

B. Light Locations:

Light locations are shown in Figure 6. For a description of the external lights, see Section 2A-33-00, Lighting.

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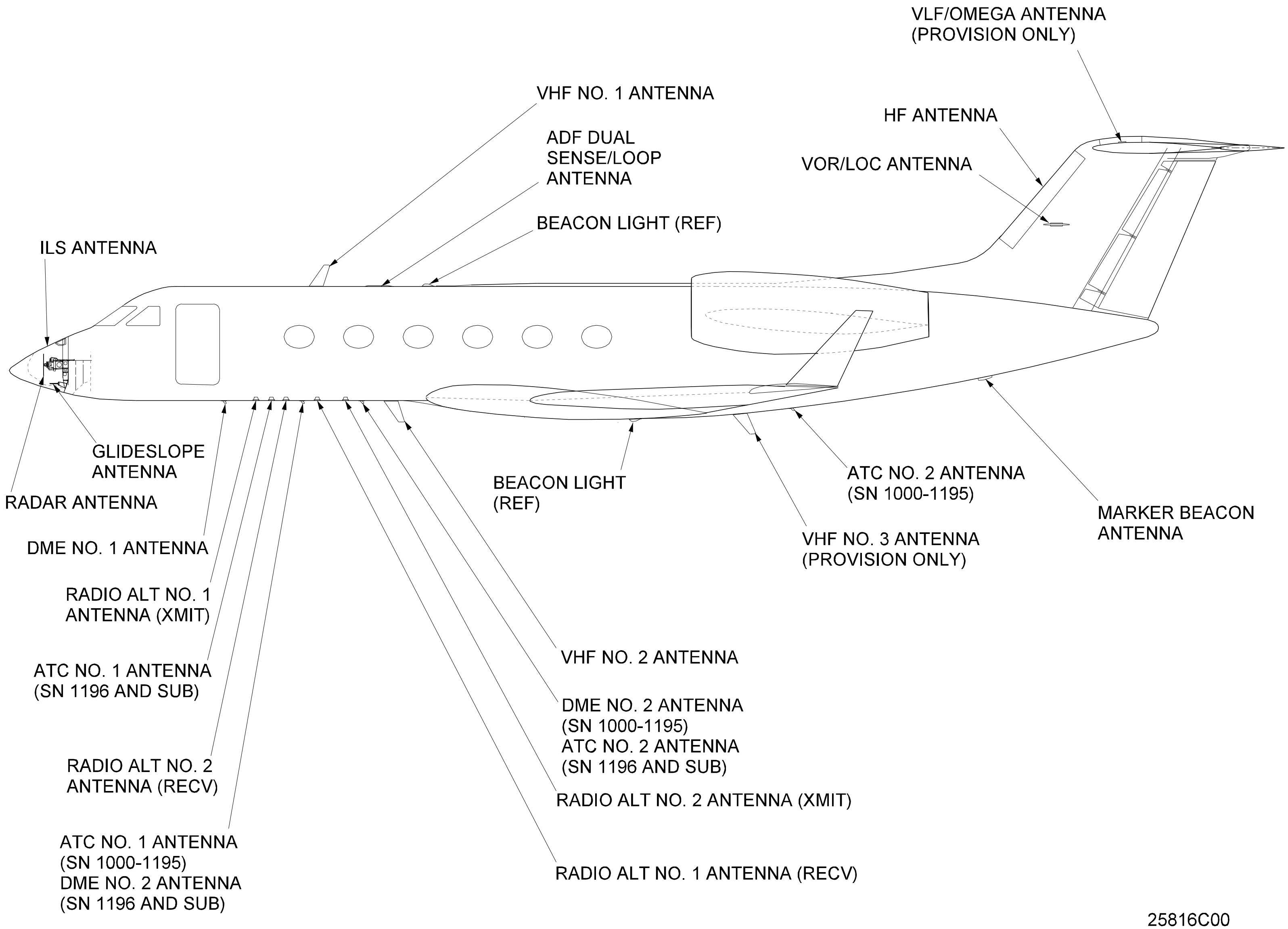
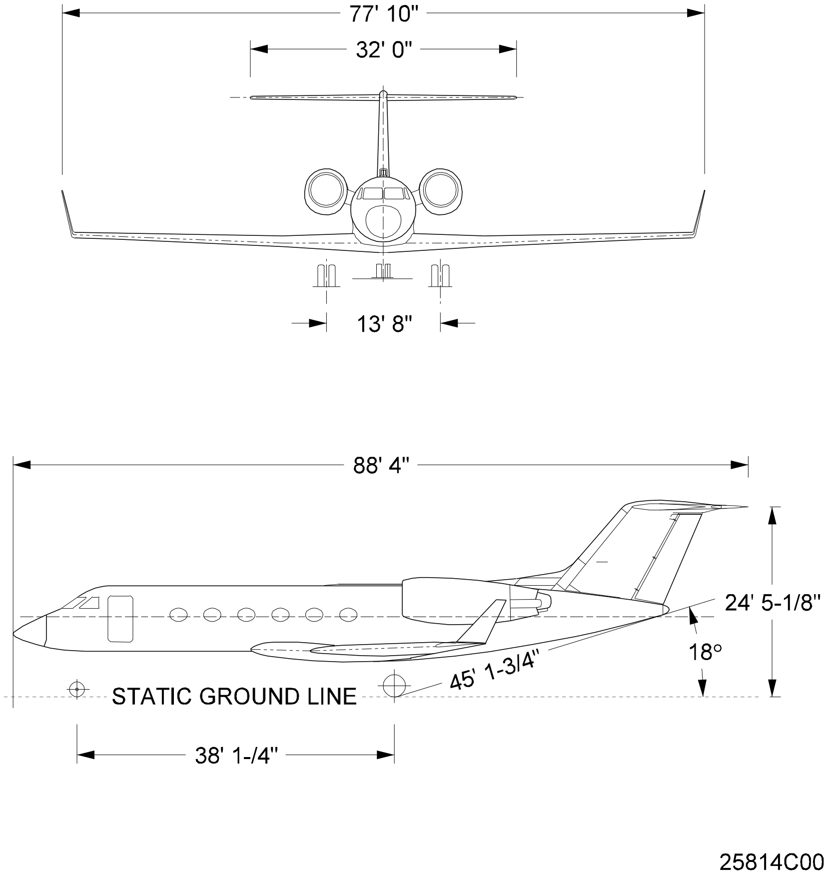
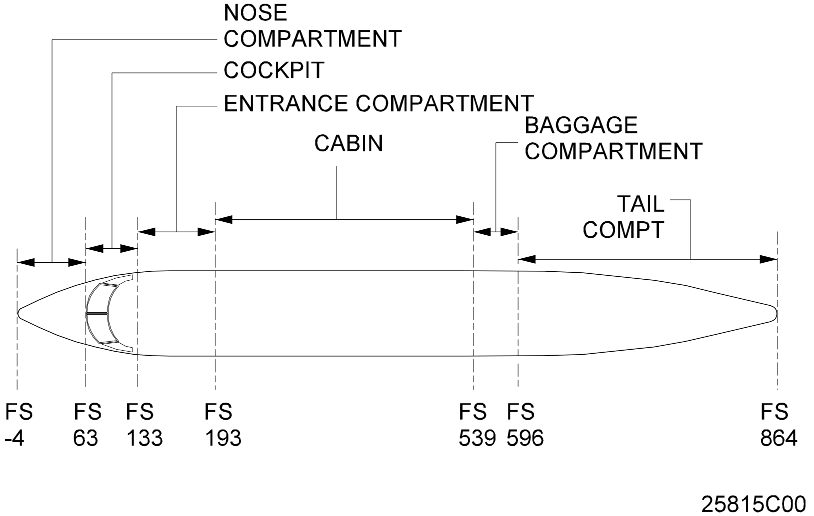


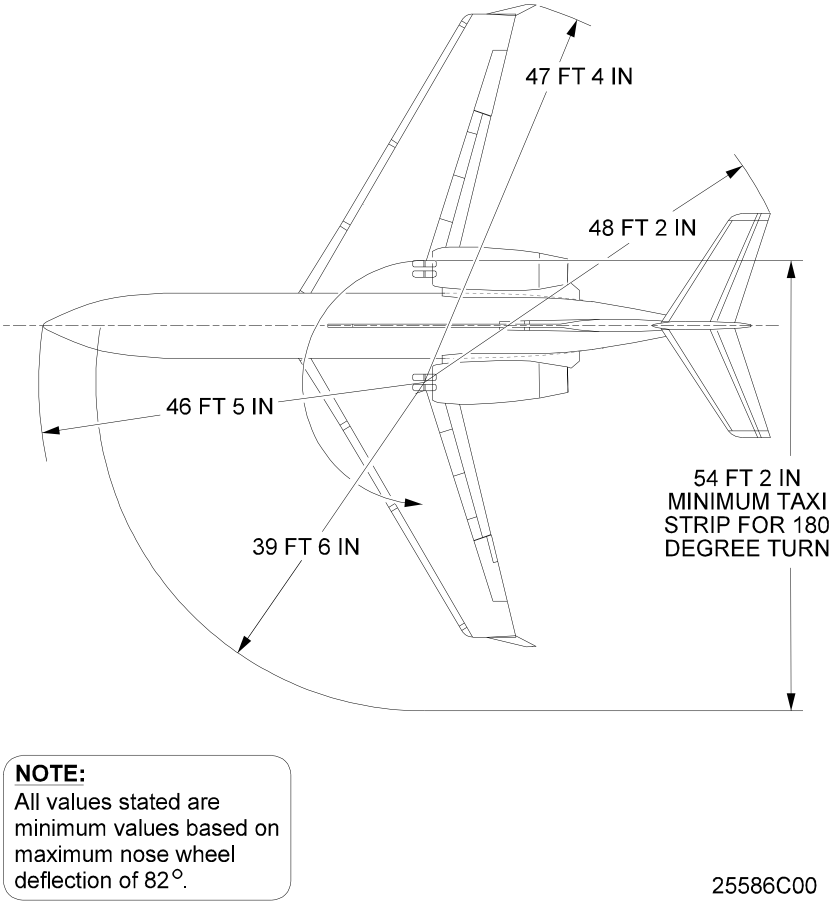
External Antennas
Figure 1



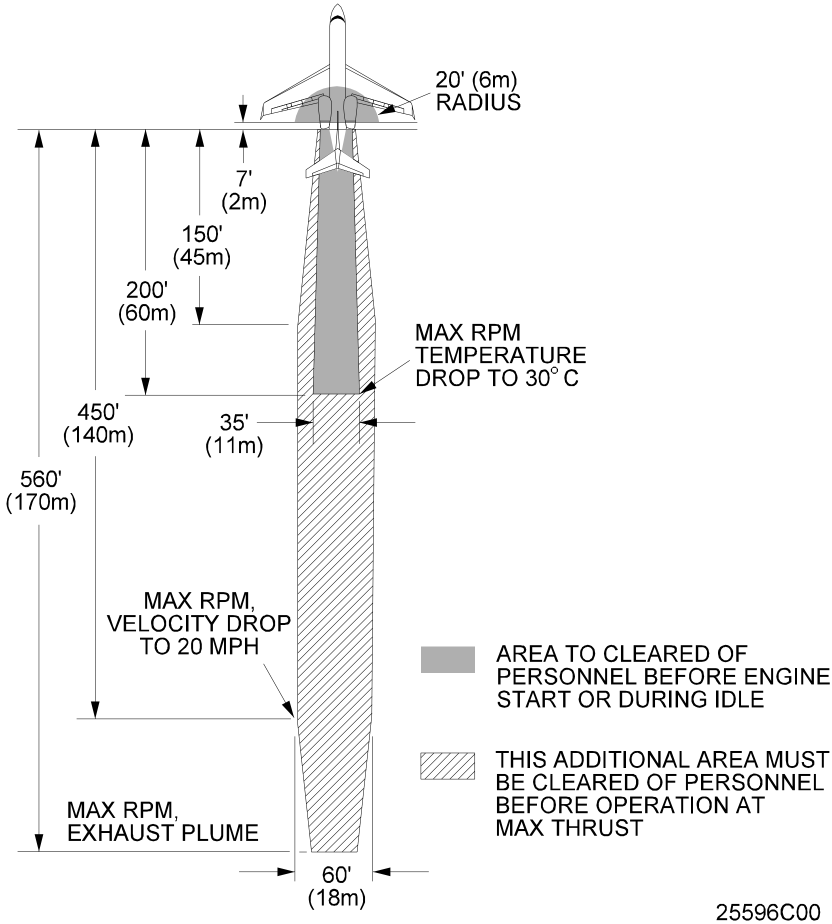
Exterior Dimensions
Figure 2



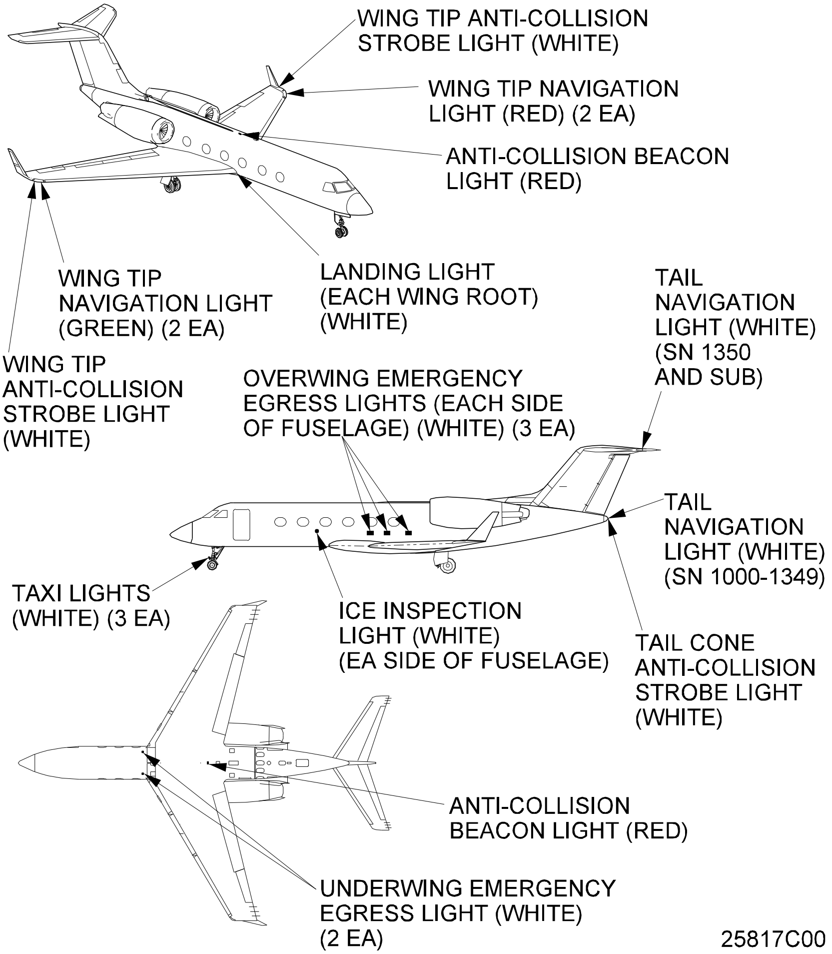
Fuselage Dimensions
Figure 3



Turning Dimensions and Distances
Figure 4



Danger and Caution Areas
Figure 5



External Lights
Figure 6

2A-06-30: Entrances, Exits and External Doors

1. **Main Entrance Door:**

   The main entrance door, located on the left side of the forward fuselage, is the primary means of access for the passengers and flight crew. It consists of a door structure with a folding stairway and handrails.

   The main entrance door is hinged at its bottom to the aircraft fuselage. It can be opened from either inside or outside the aircraft. Neither electrical nor hydraulic power is required; the door free falls to the extended position under its own weight.

   See Section 2A-52-20, Main Entrance Door, for a description of the main entrance

door.

2. **Cargo Door:**

If Aircraft Service Change (ASC) 213 is incorporated, an upward opening, hydraulically operated cargo door is installed on the right side of the fuselage forward of the wing leading edge. Measuring 80.5” high and 81” wide, the cargo door is powered by the Auxiliary (AUX) hydraulic pump.

3. **Baggage Door:**

An inward opening, plug-type baggage compartment door is installed on the left side of the aft fuselage. It can be opened from either inside or outside the aircraft. Neither electrical nor hydraulic power is required to open the door. See Section 2A-52-40, Baggage Doors, for a description of the baggage door.

4. **Tail Compartment Door:**

The tail compartment is accessed through an outward opening door on the lower rear fuselage. A removable telescoping ladder is attached to the door. The ladder can also be removed and used to reach engine areas for inspection and maintenance. See Section 2A-52-50, Tail Compartment Door, for a description of the tail compartment door.

5. **Emergency Exit Windows:**

   A. **Aircraft Without ASC 213 (Cargo Door):**

   The two aft cabin windows on the left and right fuselage are manufactured and designated as Type II emergency exit windows. Specifically, they are designated as the **primary** emergency exits from the cabin. (The main entrance door and baggage door are Type I emergency exits, designated as **secondary** emergency exits, to be used only at the direction of the flight crew.)

   The emergency exit windows can be manually opened from either inside or outside the aircraft at all times. Pulling the EMERGENCY EXIT handle releases the window, allowing the window to be pulled inward from its fuselage frame.

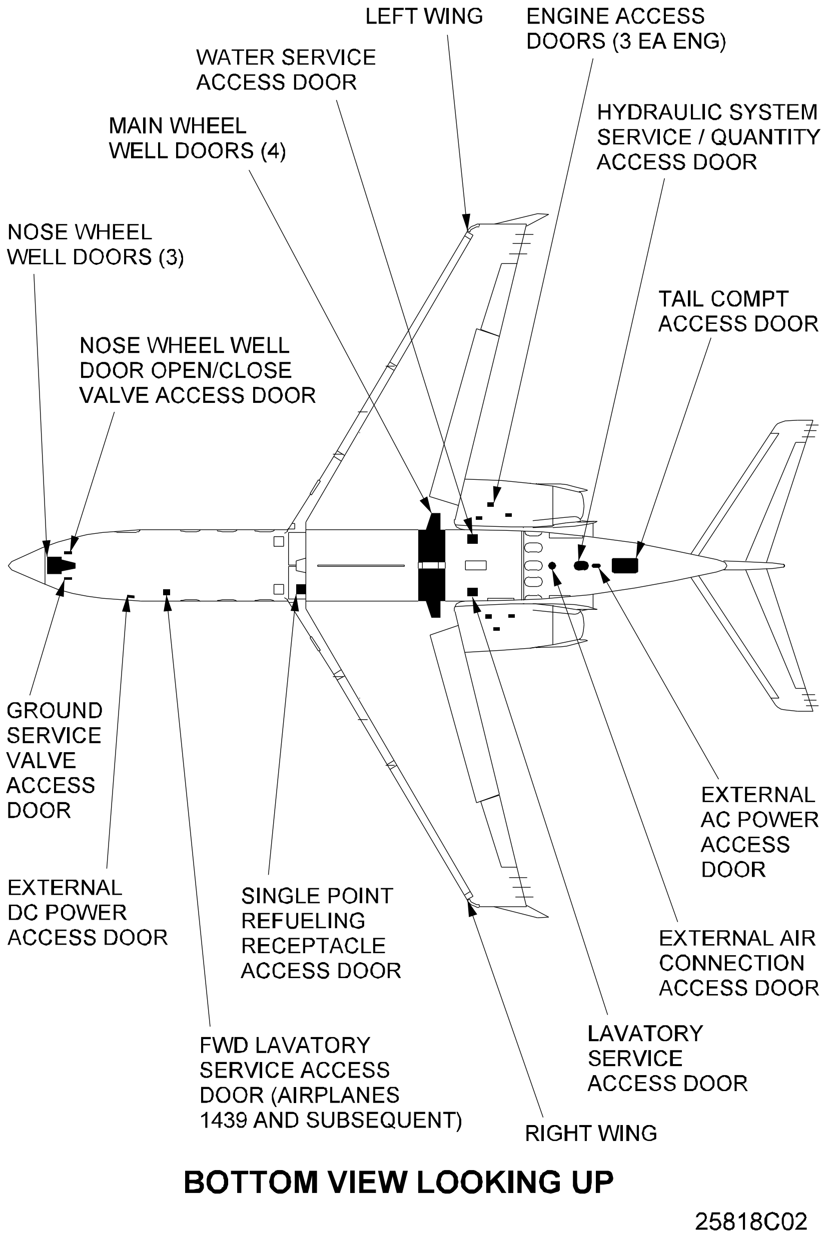
   See Section 2A-52-30: Emergency Exits, for a description of the emergency exit windows.

   B. **Aircraft With ASC 213 (Cargo Door):**

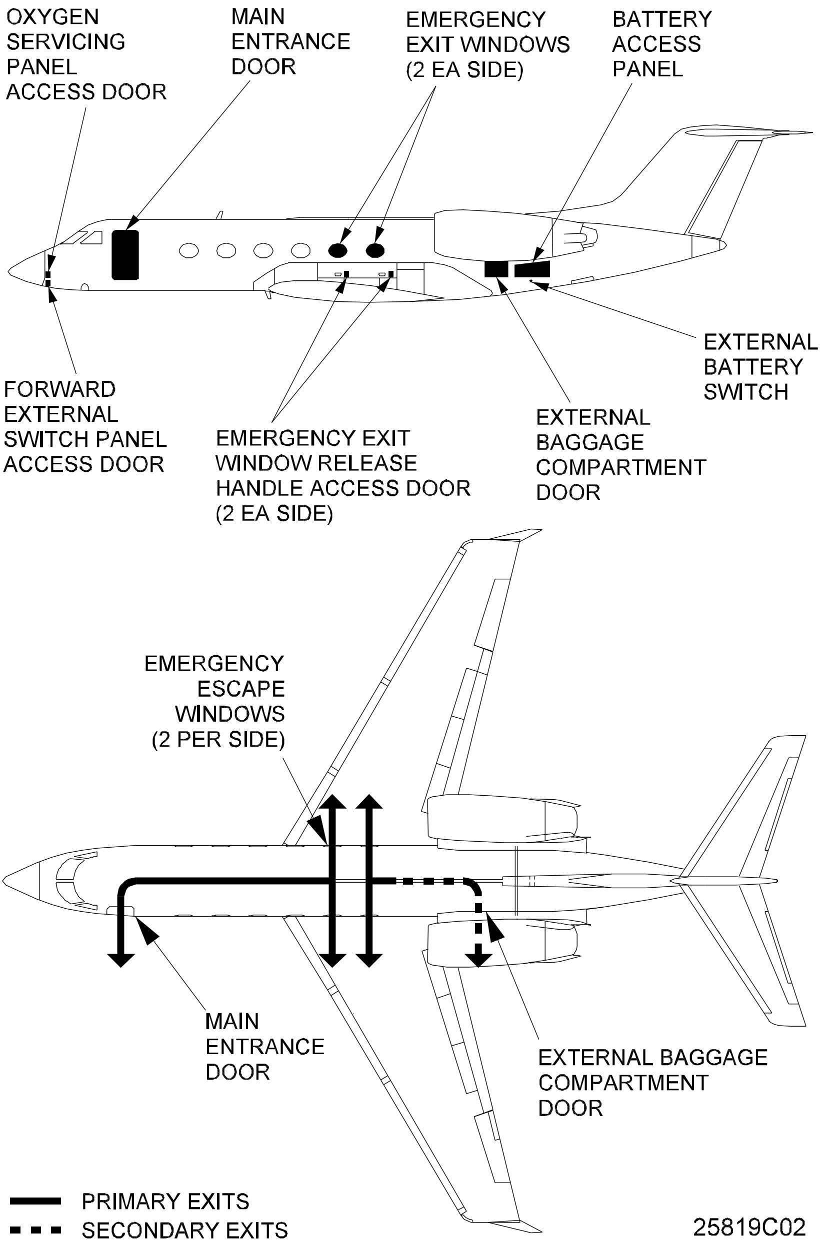
   On aircraft with a cargo door, there is a single Type II emergency exit opposite the main entrance door, and a Type III overwing emergency exit over each wing where the aft cabin window would normally be located. The Type II emergency exit measures 44” high and 20” wide, while the Type III emergency exits are 36” high and 20” wide. They are designated as the **primary** emergency exits from the cabin.

   Both the Type II and Type III emergency exits can be manually opened from either inside or outside the aircraft at all times. When inside the aircraft, releasing and then pulling the EMERGENCY EXIT handle releases the exit. When outside the aircraft, pushing a flush plate inward releases the exit.

   The main entrance door and baggage door are Type I emergency exits, designated as **secondary** emergency exits, to be used only at the direction of the flight crew. The location, type and number of emergency exits in aircraft with ASC 213 incorporated meet all civil aviation regulations for high-denstity seating arrangements (more than 19 passengers).

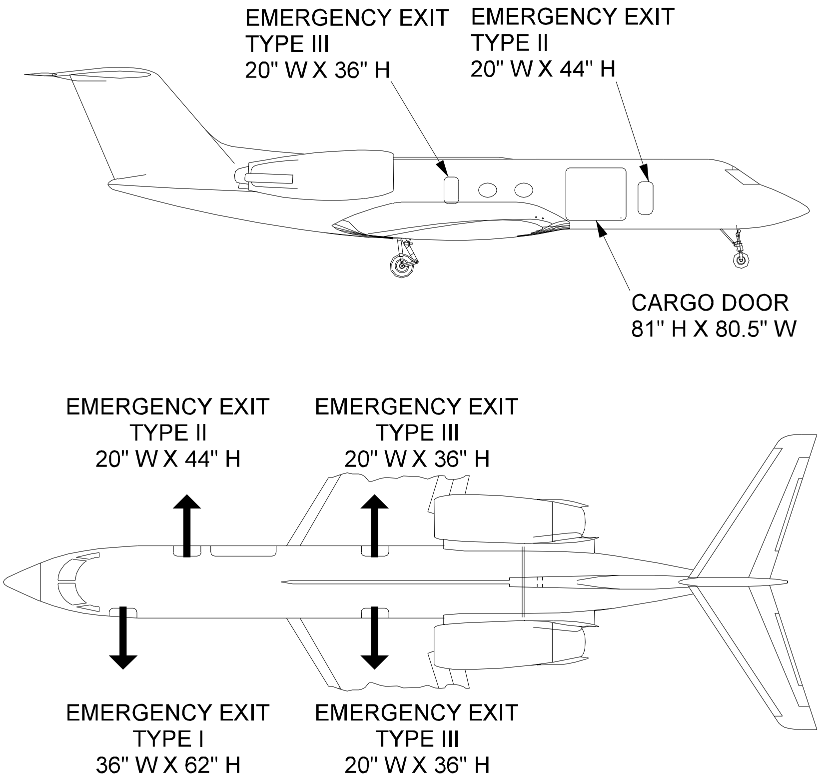


External Doors
Figure 7



Entrances, Exits and External Doors (Aircraft Without ASC 213)
Figure 8

NOTE: DEPICTS AIRCRAFT WITH ASC 213



Entrances, Exits and External Doors (Aircraft With ASC 213)
Figure 9

2A-06-40: Flight Crew Station Components

1. **General:**

   The Gulfstream IV cockpit is conventionally arranged and contains two flight crew stations with individually adjustable seats. The logical arrangement of each station allows easy access to all aircraft controls and displays. A two-piece, electrically

heated, bird-proof and splinter-proof windshield and two fixed side windows enclose the cockpit.

NOTE:

Equipment and furnishings shown in Figure 10 through Figure 19 are representative of the production standard Gulfstream IV aircraft. Actual layout may differ due to outfitting. See the appropriate system descriptions for additional details.

2. Instrument Panels:

(See Figure 12 through Figure 17.)

In front of each crew station is an instrument panel, more commonly known as the pilot's and copilot's flight panel. Each flight panel contains two Display Units (DUs). One DU is used primarily as a Primary Flight Display (PFD); the other is used primarily as a Navigation Display (ND). A display controller is located above each PFD / ND. Each crew station also has a Digital Distance Remote Magnetic Indicator (DDRMI).

The pilot's flight panel also has standby engine instruments, an oxygen supply control panel and a yaw damper / pitch trim control panel. The copilot's flight panel also has a Standby Warning Lights Panel (SWLP) (if installed), brake accumulator pressure gauge, stabilizer / flap position indicator, pressurization controller and fuel quantity gauges.

The center flight panel has two DUs. The top DU (DU 3) primarily shows Engine Instrument (EI) information; the bottom DU (DU 4) primarily shows aircraft system pages and the Crew Alerting System (CAS).

To the right of the two DUs are a standby gyro horizon, airspeed indicator, altimeter and landing gear control panel. A guidance panel is located in the center of the glareshield.

In addition to the display controllers and guidance panel, the glareshield contains a master warning / caution panel on each side. Atop the glareshield is an instrument cluster containing the Angle Of Attack (AOA) indexer lights, spoiler warning lights, engine idle annunciators and cockpit area microphone. Directly above this cluster on the windshield center post is the design eye locator.

A pedestal between the crew seats includes the navigation and communication radio control heads, Flight Management System (FMS) Control Display Units (CDUs), engine power levers and thrust reverser levers, speed brake and flap handles and miscellaneous engine and system lights and controls.

An overhead panel above and between the crew stations contains controls, lights and indicators for the electrical, engine starting, lighting control, Auxiliary Power Unit (APU), fuel, bleed air, temperature control, cabin pressure control and anti-ice systems. The overhead panel also contains the Electronic Flight Instrument System (EFIS) reversionary panel.

Each side console panel contains an audio control panel, cockpit and auxiliary lighting controls and an EROS oxygen mask storage box. The pilot's side console contains the Nosewheel Steering (NWS) controls and handwheel, while the copilot's side console has the passenger oxygen control panel, Cockpit Voice Recorder (CVR) control head, emergency landing gear handle and the emergency flap handle.

3. Cockpit Windows:

A two-piece, electrically heated, bird- and splinter-proof windshield and two fixed side windows enclose the cockpit. Each windshield consists of three ply layers with two interlayers. The outer layer is a thin layer of glass having superior abrasion resistance over all-plastic windshields. The inner surface of this glass layer also has a conductive coating for the windshield anti-icing system.

The left and right side windows consist of two ply layers with an interlayer and an Aircon heating mat sandwiched between the inner and outer ply.

4. Circuit Breaker (CB) Panels:

(See Figure 11 and Figure 18.)

There are four CB panels in the cockpit. Two panels, one on either side of the overhead control panel, are referred to as the Pilot's Overhead CB Panel and Copilot's Overhead CB Panel. The remaining two panels, located behind the pilot's and copilot's seats, are referred to as the Pilot's Aft CB Panel and Copilot's Aft CB Panel. Design and location of these four CB panels is such that neither crewmember shall be required to leave their duty station to access critical CBs.

5. Flight Compartment Seating:

(See Figure 19.)

A. Description:

Each crewmember seat has adjustments for height, recline and forward and aft travel. The seat also has an adjustable back cushion, lumbar support, thigh pad and headrest. A book storage pouch and life jacket are behind the seat.

The height and lumbar vertical adjustment controls are on the outboard side of each seat while the height, forward / aft travel, lumbar horizontal and thigh pad adjustment controls are on the inboard side.

A five-point seat belt / shoulder harness attaches to the seat. The lap and crotch belts attach to the seat bottom and the inertia reel shoulder harness attaches to the seat back. A shoulder strap release tab is located on the back of the shoulder harness attach points. Pushing forward on the tab releases the shoulder straps without releasing the lap or crotch straps. Normally, with the inertia reel lever in the AUTOMATIC LOCK position, the inertia reel allows unrestricted movement forward and aft until a two to three G deceleration force locks the reel. Relaxing forward pressure on the shoulder straps releases the reel lock.

An optional third crewmember jumpseat is located in the aisle behind the pilot and copilot seats. When not in use, the jumpseat is stored out of the way.

B. Operation:

For optimum comfort and safety, the following description is provided to ensure the seat is properly adjusted and positioned:

(1) Adjust the thigh pads to approximately mid-position.

(2) Adjust the lumbar in / out support and up / down support to the out and down positions.

(3) Adjust the recline angle to approximately mid-position.

(4) Adjust the back cushion to approximately mid-position.

(5) Ensure the occupant is correctly positioned in the seat pan.

(6) Adjust the recline angle until the shoulders are positioned at the desired sitting angle.

(7) Adjust the lumbar in / out support and up / down support to the desired positions.

(8) Using the vertical adjustment control and fore / aft track lock control, position the seat to properly align the occupant with the Design Eye Locator on the center windshield post.

(9) Verify thigh pads “break away” to allow sufficient movement on the rudder pedals and toe brakes.

6. Standard Aircraft Equipment and Furnishings:

The equipment listed in the following tables is supplied as standard equipment on each aircraft:

AIR CONDITIONING	
Quantity	**Nomenclature**
1	APU Exhaust Plug
1	Anti-Ice Discharge Port Cover
1	LH Ram Air Scoop Plug
1	RH Ram Air Scoop Plug
1	Outflow Valve / Safety Valve Cover
1	LH Ram Air Exhaust Port Cover
1	RH Ram Air Exhaust Port Cover
1	Tail Compartment Vent Cover

ELECTRICAL	
Quantity	**Nomenclature**
1	AC / DC Electrical Load Analysis
2	Battery Cell Cap Remover

FURNISHINGS	
Quantity	**Nomenclature**
3	Pitot Tube Cover
2	Flashlight
2	Rechargeable Maglite (Cockpit / Baggage Compartment) **(1)**
2	Logo Button
2	Airstair Door Emblem
4	Static Port Cover
4	Oxygen Cylinder Cushion, Clamp and Nut

NOTE(S):

(1) SN 1462 and subs.

FURNISHINGS	
Quantity	**Nomenclature**
2	Fuel Vent Screen Assembly

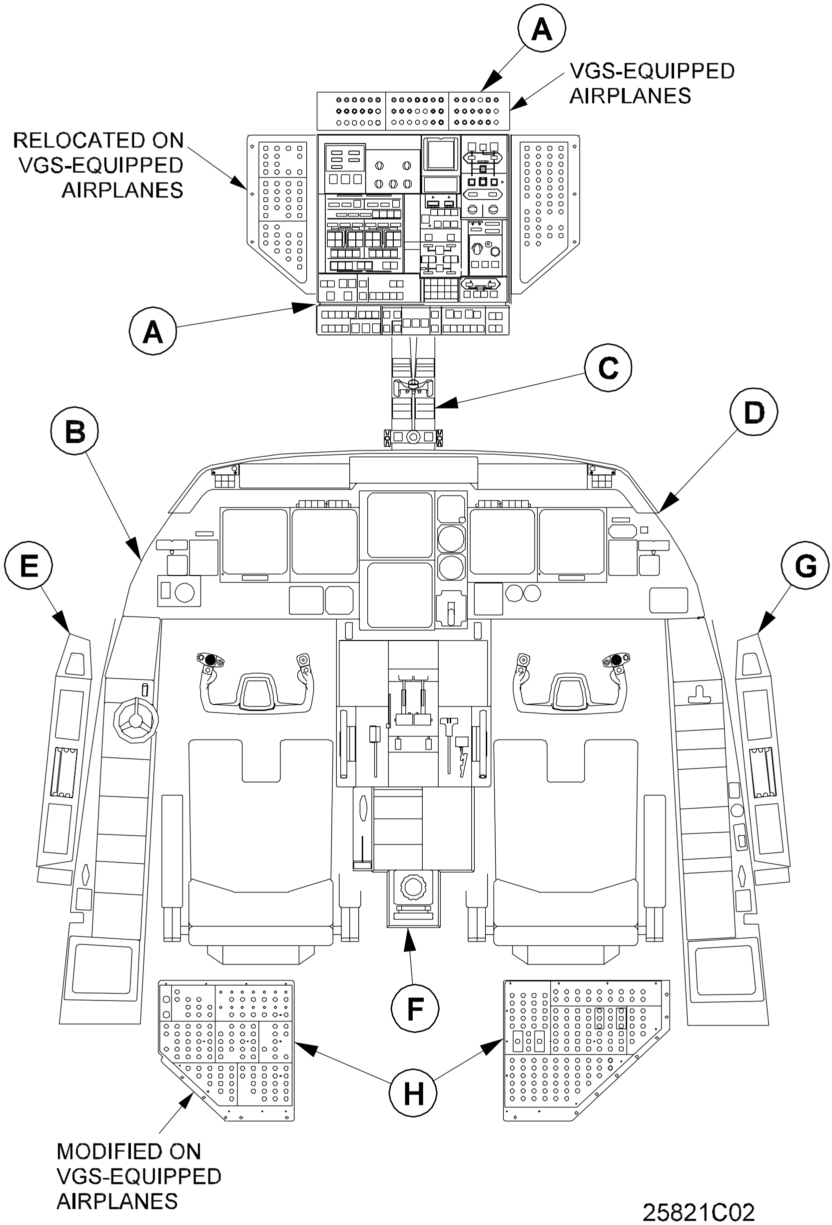
FURNISHINGS	
Quantity	**Nomenclature**
2	Engine Intake Cowling Cover
2	Engine Exhaust Reverser Doors Cover
2	Upper Cowl Inlet Scoop Plug
2	NAC Air Inlet Cover
2	NAC Air Outlet Cover
2	Generator Air Exhaust Plug
2	Low Pressure Cooling Exhaust Plug
2	Generator Air Inlet Plug
2	Pylon Outlet Cover
2	Pylon Air Inlet Plug
1	Exhaust Louver Covers Assembly

PRODUCT SUPPORT	
Quantity	**Nomenclature**
1	Nose Wheel Door Ground Safety Strut
2	Main Landing Gear Safety Lock
2	Main Landing Gear Door Safety Valve Lock
1	Nose Landing Gear Safety Lock
1	Nose Landing Gear Door Safety Valve Lock
2	Smoke Goggles

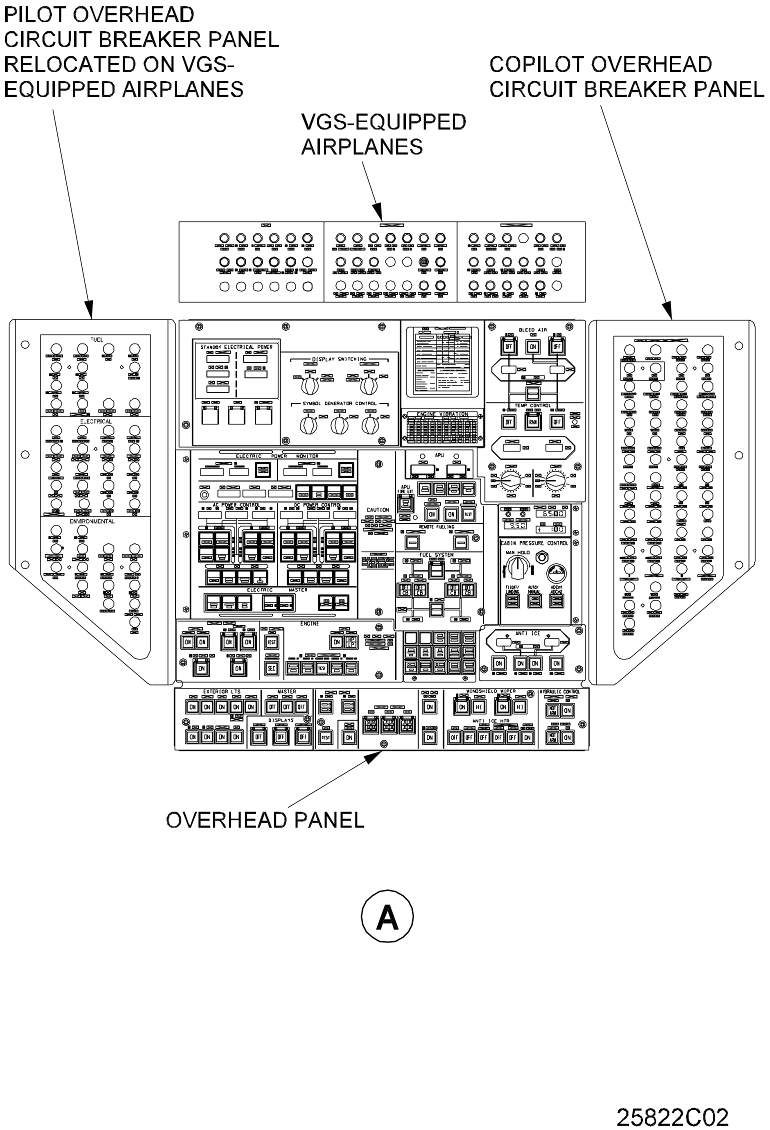
PUBLICATIONS	
Quantity	**Nomenclature**
1	Weight and Balance Report
2	Engine Log Book
1	Airplane Log Book
1	Aeronautical Equipment Service Record
1	Maintenance Manual
1	Wiring Diagram Manual
1	Illustrated Parts Catalogue
1	Airplane Flight Manual
1	Oxygen System Report
1	Soundproofing Report

STRUCTURES	
Quantity	**Nomenclature**
7	Cabin Floor Blowout Cover

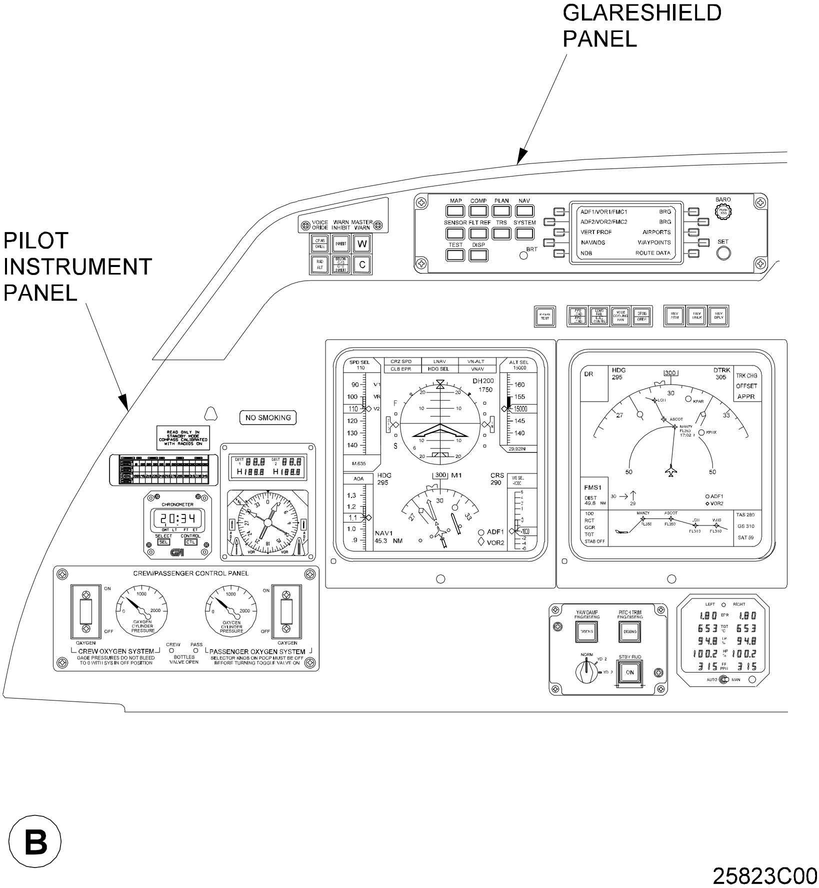
HYDRAULICS	
Quantity	**Nomenclature**
3	Hydraulic Fluid Sample Kit



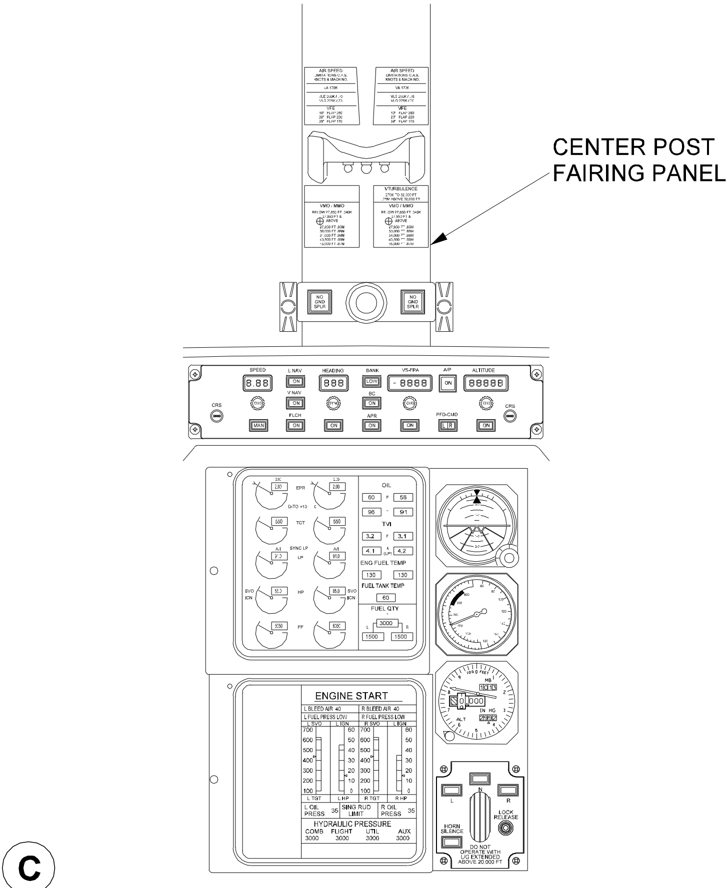
Cockpit Areas
Figure 10



Overhead Panel / Circuit Breaker Panels
Figure 11



Pilot Flight Panel
Figure 12



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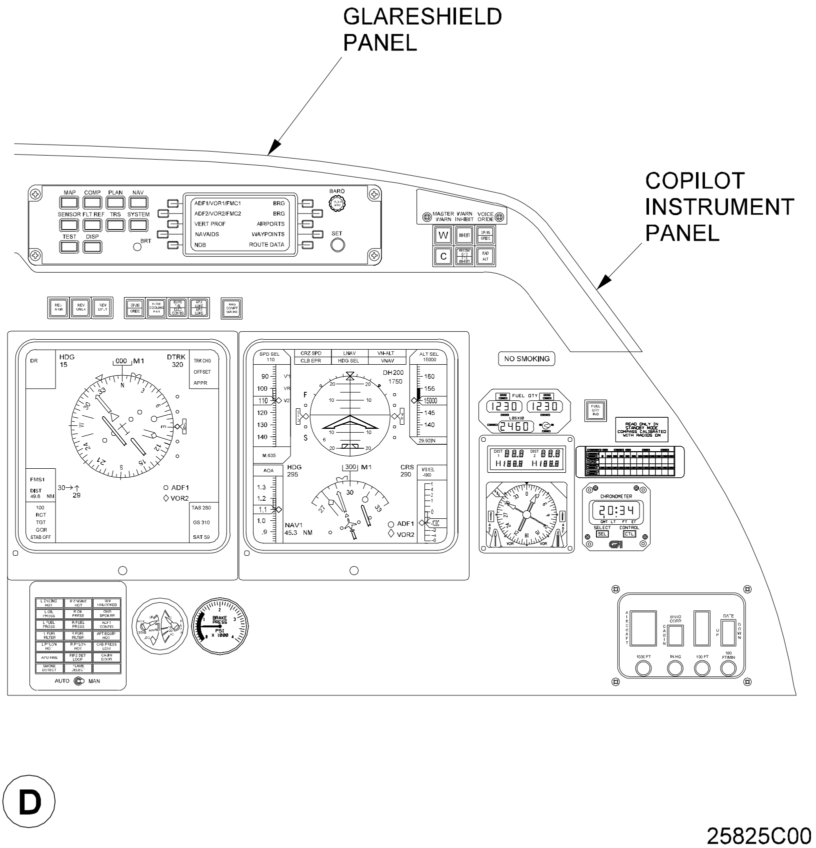
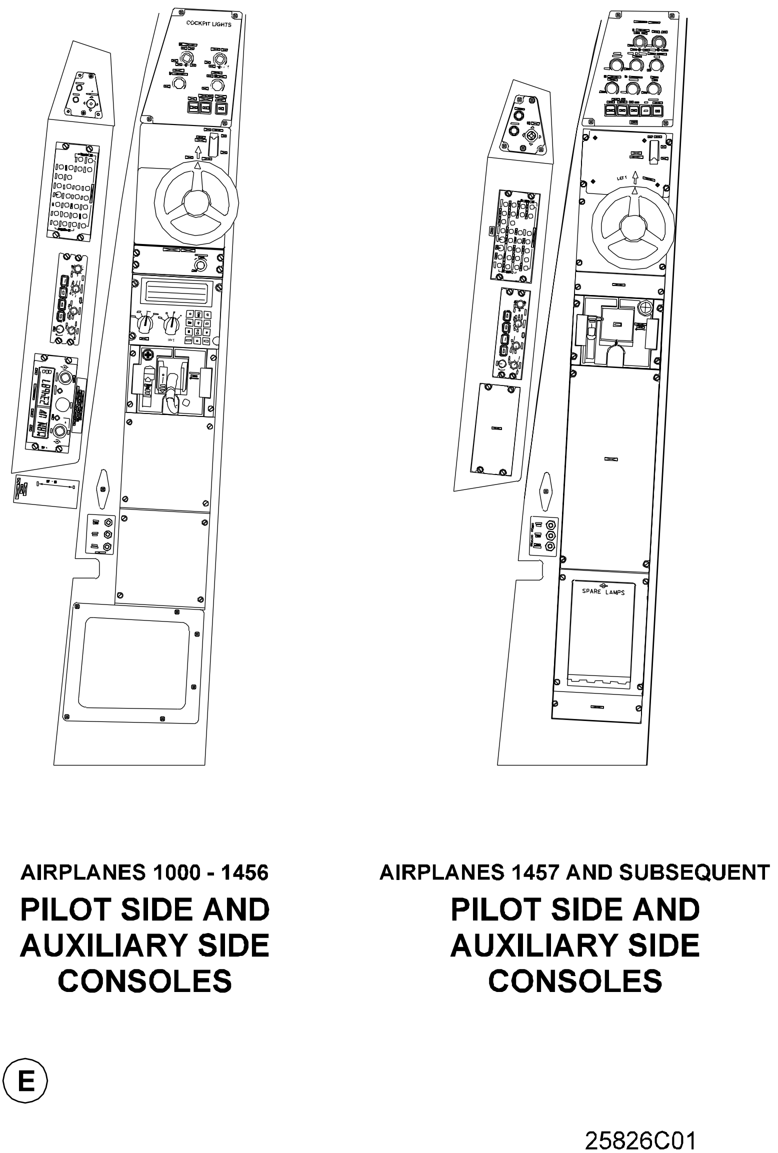
Center Flight Panel
Figure 13



Copilot Flight Panel
Figure 14



Pilot Side Console
Figure 15

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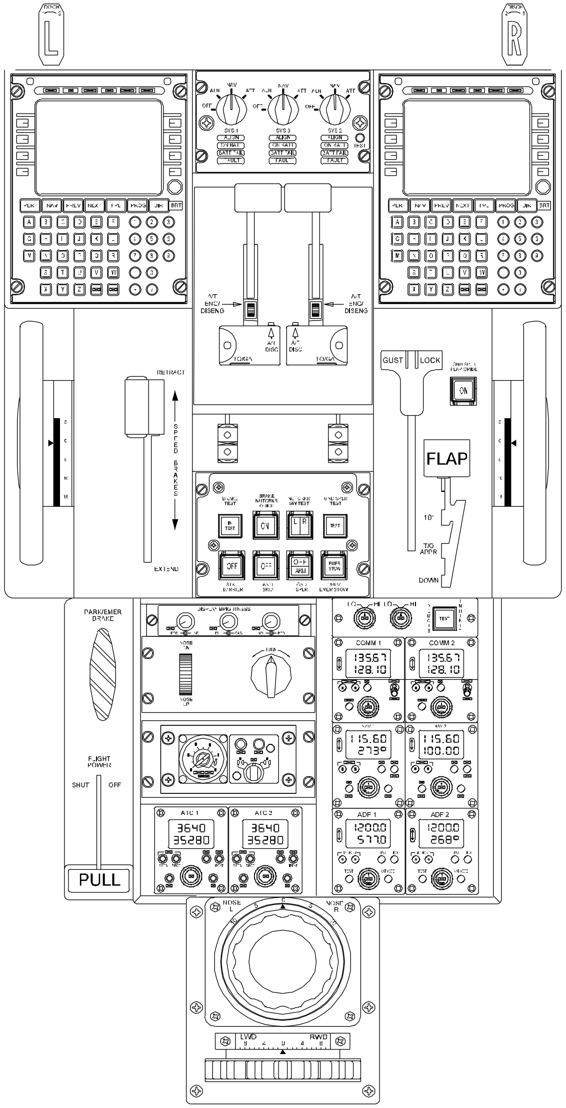
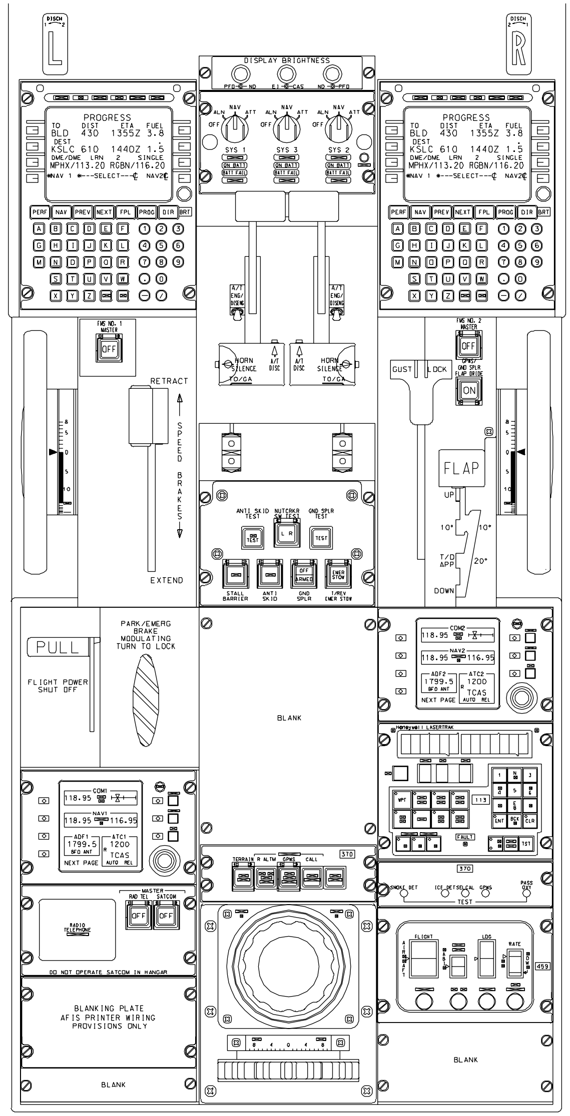


AIRPLANES 1000 - 1456

CENTER PEDESTAL

F



AIRPLANES 1457 AND SUBSEQUENT

CENTER PEDESTAL

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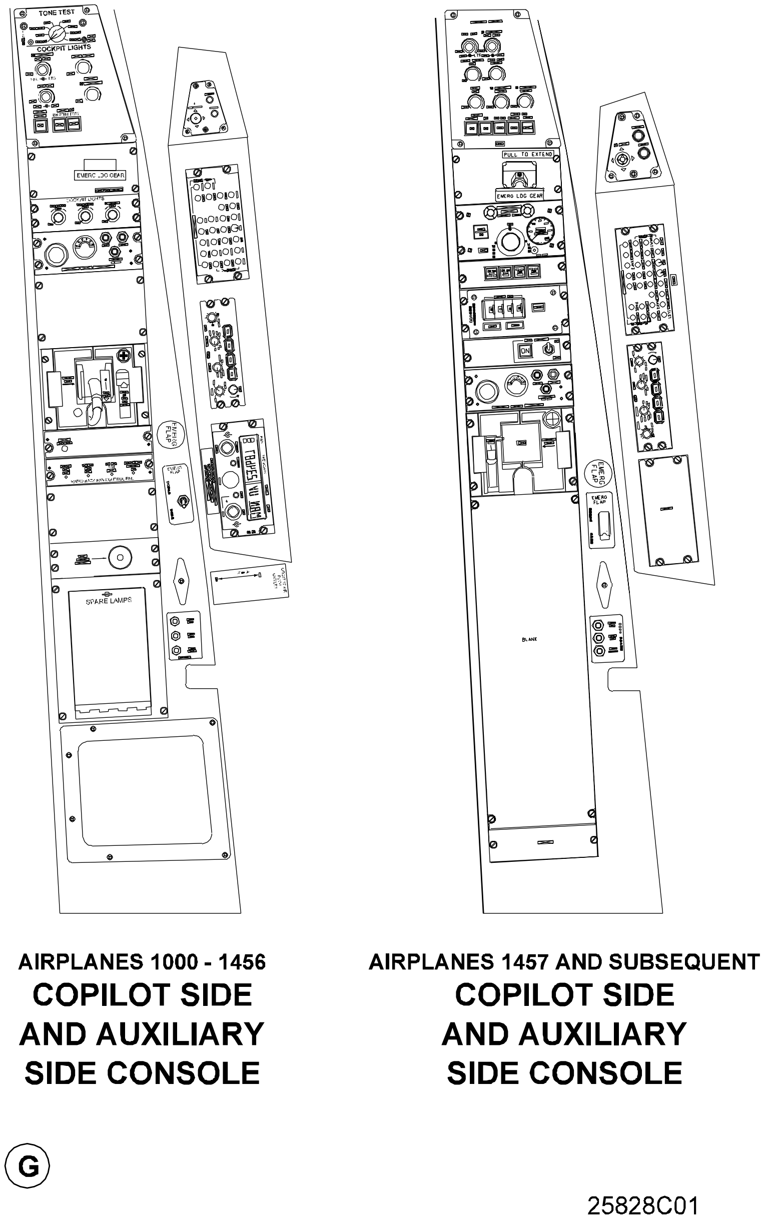
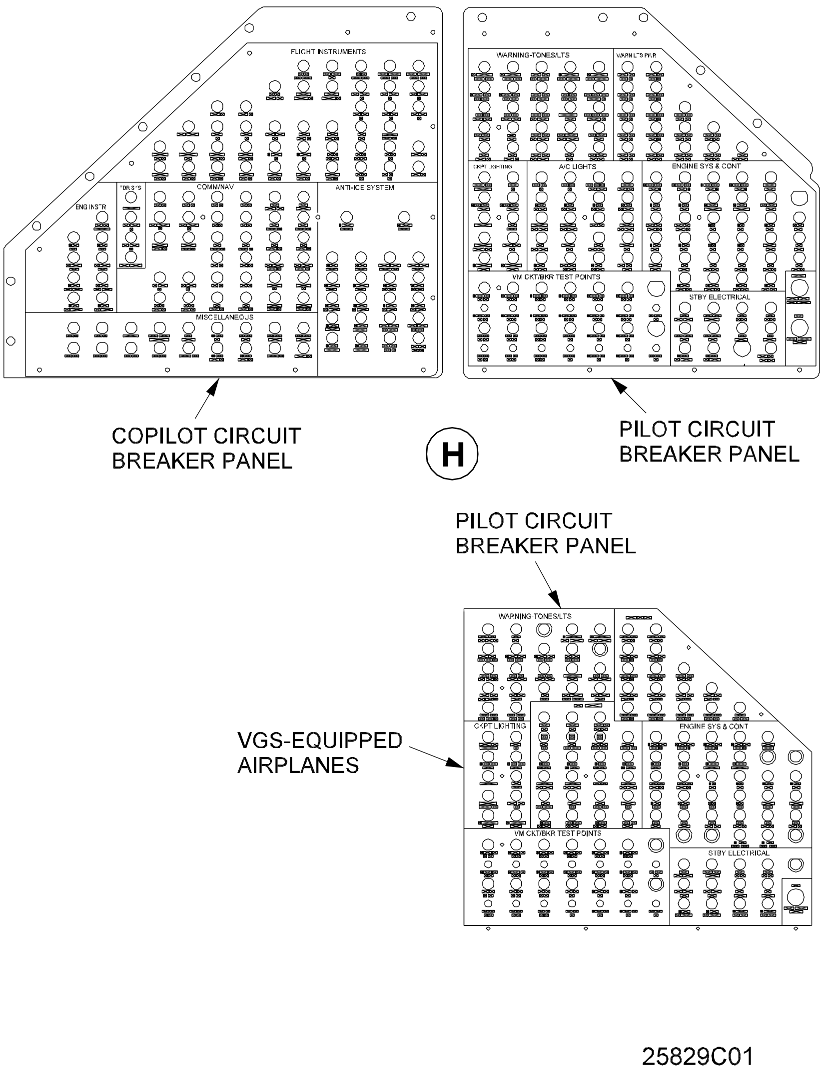
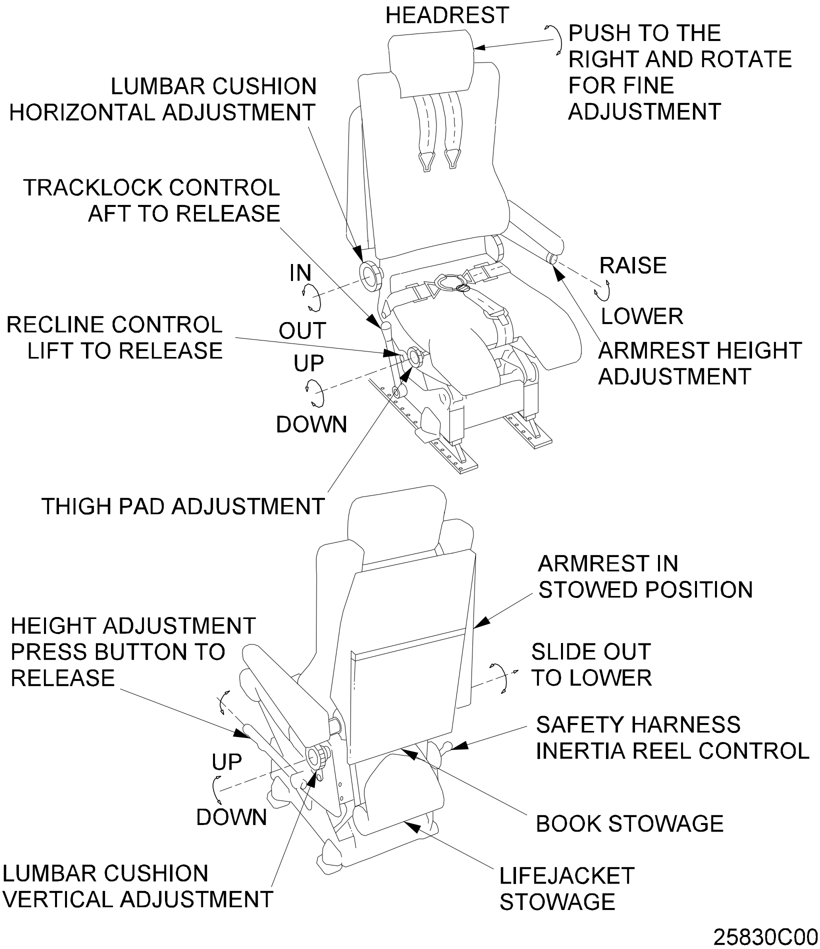
Center Pedestal
Figure 16



Copilot Side Console
Figure 17



Pilot / Copilot Aft Circuit Breaker Panels
Figure 18



Pilot / Copilot Seats
Figure 19

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2A-21-10: General

The air conditioning system for the Gulfstream IV is designed to provide all areas within the pressure vessel with a safe and comfortable temperature and pressure (cabin altitude) throughout the aircraft's operating envelope. The system employs a "dual pack" concept and, although each pack is controlled separately, failure of one pack still leaves the remaining pack capable of supplying conditioned air to both cabin and cockpit, if required.

To achieve this function, the air conditioning system provides the flight crew with a means to accomplish the following:

- Control, regulate and monitor the amount of conditioned air within the pressure vessel to achieve and maintain selected or preprogrammed cabin pressure (altitude), while still allowing exchange of the air at regular intervals for occupant comfort. This is accomplished by modulation of a single outflow valve. The outflow valve is automatically controlled by the cabin pressure controller in the normal operating mode, but can be controlled manually.
- Select and control bleed air entering and exiting two environmental control system refrigeration pack assemblies, referred to as the Left and Right ECS Packs. Source air into the ECS Packs is provided to a bleed air manifold by either external air or APU air (while on the ground), or by the aircraft's engines (on ground or in flight). Through use of an isolation valve, air from the bleed air manifold can be directed to either ECS Pack. This results in a constant mass of conditioned air for all areas within the pressure vessel.
- Control the temperature of conditioned air delivered to the cockpit and cabin areas (referred to as zones) within the pressure vessel. This is done using the two ECS packs to cool incoming air and deliver it to a conditioned air manifold. Valves mix hot bleed air with cold conditioned air to modulate the temperature of the air coming from the manifold into the pressure vessel. Cockpit control and indication is also provided.

The Air Conditioning system is divided into the following subsystems:

- 2A-21-20: Pressurization Control System
- 2A-21-30: Airflow and Temperature Control System

2A-21-20: Pressurization System

1. **General Description:**

   The pressurization system controls, regulates and monitors the amount of conditioned air within the pressure vessel to achieve and maintain a safe and comfortable cabin pressure (cabin altitude), up to the airplane's maximum operating altitude. While normally preprogrammed, cabin altitude can also be controlled manually. Cabin conditioned air is also exchanged at regular intervals for occupant comfort.

   With airflow supplied from the ECS packs, the pressurization system maintains cabin altitude by regulating the amount of air exhausted overboard through a single outflow valve. Once the flight crew has programmed the system, operation is virtually automatic.

   The pressurization system is capable of maintaining a cabin altitude of 6,550 feet at a maximum inflight altitude of 45,000 feet. Sea level cabin pressure can be

maintained to a maximum inflight altitude of 22,000 feet. If cabin altitude should exceed 10,000 feet, warnings and annunciations are provided to the flight crew so that appropriate action may be taken.

The pressurization system performs the following functions:

- Automatically maintains selected cabin altitude through isobaric pressurized operation
- Automatically limits maximum cabin pressure differential
- Provides safety pressure relief operation
- Provides negative (vacuum) pressure differential control
- Allows manual barometric correction for pre-programmed Landing Field Elevation (LFE)
- Permits manual cabin altitude control through control of the outflow valve
- Automatically limits cabin altitude rate-of-change to a maximum of 3,000 feet-per-minute (FPM) during pressurization and depressurization
- Provides crew-selected cabin altitude rate-of-change
- Regulates and smooths cabin pressurization to prevent pressurization surges or “bumps”
- Provides rapid cabin ventilation for smoke removal

Normally the pressurization system limits cabin pressurization differential to 9.55 ±0.1 psid. As differential pressure reaches 9.55 psid, an amber CABIN DFRN 9.6 caution message is displayed on the Crew Advisory System (CAS) and the pressurization system begins limiting outflow valve closure.

If the pressurization system malfunctions and cannot limit maximum cabin pressure differential to 9.55 ±0.1 psid, a safety valve limits pressure differential to 9.7 ±0.1 psid. As differential pressure reaches 9.8 psid, a red CABIN DFRN 9.8 warning message is displayed on CAS.

The pressurization system receives information from the two Air Data Computers (ADC #1 and ADC #2). It also uses the Weight-On-Wheels (WOW) system (commonly referred to as the nutcracker system) to control various system operating modes.

Under typical flight conditions, the flight crew programs the system prior to takeoff. Apart from selection of the FLIGHT/LANDING mode switch or adjustment of LFE, system operation is virtually automatic.

Major components of the pressurization system are:

- Cabin Pressurization Transducer
- Cabin Pressure Outflow Valve
- CABIN PRESSURE CONTROL Panel
- Cabin Pressurization Selector Panel
- Cabin Pressure Safety Valve
- Cabin Rate Pressure Switch
- Cabin Pressure Warning Switch
- Cabin Differential Pressure / Altimeter / Rate-of-Climb Indicator

2. Description of Subsystems, Units and Components:

(See Figure 1.)

A. Cabin Pressurization Transducer:

The solid-state cabin pressurization transducer is the heart of the pressurization system. Located in the electronics area of the entrance compartment, produces the output signal to the outflow valve motor based on the following input data:

- Inputs from the cabin pressurization selector panel
- Corrected static pressure data from the ADCs (or Digital Air Data Computers [DADCs])
- FLIGHT or GROUND mode from the nutcracker system
- Sensed cabin pressure from inside the pressure vessel

To prevent undesired outflow valve movement from transient power surges, the transducer obtains a clean, stable supply of power from an Electromagnetic Interference (EMI) filter installed in the radio rack.

Within the transducer are circuits that limit cabin differential pressure in flight to 6,550 feet with an airplane altitude of 45,000 feet.

B. Cabin Pressure Outflow Valve:

A single pressurization outflow valve is installed under the lower shelf of the radio rack. It is an electrically-controlled, motor-driven valve that determines the amount of cabin air exhausted overboard, thus controlling cabin pressurization. It is capable of moving from fully open to fully closed in approximately ten (10) seconds.

The outflow valve is a butterfly type valve with an electro-mechanical actuator and a potentiometer that provides valve position information to the position indicator on the CABIN PRESSURE CONTROL panel. Within the actuator are a DC motor with DC brakes, an AC motor with AC brakes and a motor-generator.

The AC motor, which operates the outflow valve in the Automatic (AUTO) mode, responds to signals from the cabin pressurization transducer. The DC motor operates the outflow valve in the MANUAL mode by responding to signals from the CABIN PRESSURE CONTROL panel manual control knob. Both motors are capable of extremely slow or fast operation, or at any intermediate speed required by the controlling device.

During normal system operation (AUTO mode), the cabin pressurization transducer opens and closes the outflow valve using the AC motor. While the AC motor is in operation, the DC motor brake engages to prevent DC motor movement. The motor-generator in turn provides a rate-of-change signal back to the cabin pressurization transducer.

With the system in MANUAL mode, the manual control knob provides a DC signal to position the outflow valve. The AC motor brake engages to prevent AC motor movement. Although the motor-generator is inactive, the potentiometer still provides valve position information to the valve position indicator.

In the unlikely event that both Essential AC and Essential DC bus power sources are lost, the outflow valve will cease operation and remain at the last position commanded.

C. CABIN PRESSURE CONTROL Panel:

(See Figure 3.)

The CABIN PRESSURE CONTROL panel, located on the cockpit overhead panel, has controls and indicators for:

- Manual control of the outflow valve
- Position of the outflow valve
- Selection of either FLIGHT or LANDING pressurization schedules
- Selection of either AUTO or MANUAL mode of operation
- Selection of either ADC #1 or ADC #2 to supply static pressure information to the pressurization transducer

The cabin pressure control panel operates on 28V DC from the Essential DC bus and 115V AC, 400 Hz from the Essential AC bus.

D. Cabin Pressurization Selector Panel:

(See Figure 4.)

Automatic operation of the pressurization system requires crew inputs on the cabin pressurization selector panel, located on the copilot's flight panel. It contains control knobs and indicator tapes for setting the following:

- FLIGHT: flight plan's maximum aircraft altitude and corresponding cabin altitude to be maintained
- BARO CORR: barometric pressure correction to local conditions (28.00 to 31.00 inches of mercury)
- LDG: preprogrammed LFE (-1,000 to +15,000 feet)
- RATE: cabin altitude rate-of-change for climb and descent in FPM (minimum UP: 50 FPM, minimum DOWN: 30 FPM, maximum UP: 2,000 FPM, maximum DOWN: 2,000 FPM)

Each control knob is connected to a variable resistor and gear train that drives the indicator tape. Once programmed, the cabin pressurization selector panel supplies driving signals to the pressurization transducer.

E. Cabin Pressure Safety Valve:

A pressurization safety valve, located below the radio rack, provides safety pressure relief, vacuum relief and pressurization rate limiting. Because it operates entirely on cabin and ambient pressure, it is independent of all other components in the pressurization system and requires no external power source.

Should the AUTO or MANUAL control mode of the pressurization system malfunction and cabin pressure builds up to approximately 9.7 ±0.1 psid, the safety valve will open. It then modulates to limit cabin pressure to the safety relief pressure of 9.8 psid.

F. Cabin Rate Pressure Switch:

The cabin rate pressure switch functions strictly as a safety device by sensing the rate at which the cabin altitude is increasing (losing pressure). If a failure should occur that results in rapidly rising cabin altitude, the rate switch inhibits automatic (AC) control of the outflow valve and shifts to manual (DC) control. This would occur at loss rates of approximately 3,000 FPM.

G. Cabin Pressure Warning Switch:

The cabin pressure warning switch, located on the right side of the entrance compartment, is an aneroid-operated switch that reacts to cabin altitude. If cabin altitude exceeds 9,250 ±750 feet, the switch causes a red CABIN PRESS LOW warning to be displayed on CAS and, if installed, on the Standby Warning Lights Panel (SWLP).

H. Cabin Differential Pressure / Altimeter / Rate-of-Climb Indicator:

(See Figure 5.)

The cabin differential pressure / altimeter / rate-of-climb indicator is located on the overhead panel above the CABIN PRESSURE CONTROL panel. It provides the following indications:

(1) Cabin Differential Pressure: The cabin differential pressure indicator (labeled DFRN PRESS) is driven by the cabin pressurization transducer. The display indicates cabin differential pressure to the nearest 1/100th psid on a four digit display.

If cabin differential pressure reaches 9.6 psid, an amber CABIN DFRN-9.6 caution message is displayed on CAS and an amber light above the indicator will illuminate. If differential pressure reaches 9.8 psid, a red CABIN DFRN-9.8 warning message is displayed on CAS and a red light above the indicator will illuminate.

(2) Cabin Altimeter: The cabin altitude indicator (labeled CABIN ALT) is located adjacent to the differential pressure indicator. It displays cabin altitude in feet (FT) on a five digit display.

(3) Cabin Rate-of-Change Indicator: The cabin rate-of-change indicator (labeled RC) is located beneath the cabin altimeter. It displays cabin altitude rate-of-change in FPM on a four digit display. A plus or minus sign precedes the digits to show cabin climb or descent.

3. Modes of Operation:

A. Automatic Operation Mode:

Automatic operation of the cabin pressurization system can best be understood by using a typical flight scenario shown in Figure 2 as an example. In this scenario, the crew begins their flight with a standard sea level field pressure altitude and progresses through closing the doors, engine start, taxi out, takeoff, climb to cruise, cruise, descent to approach altitude, executing the approach, landing, engine shut down and opening the doors.

The flight crew initially sets 45,000 feet on the FLIGHT scale of the cabin pressurization selector panel. This results in the adjacent CABIN scale reading of 6,550 feet, the isobaric cabin altitude that corresponds to the maximum differential pressure of 9.41 psid. The cabin RATE dial is set at 500 FPM UP, resulting in the adjacent DOWN scale reading 300 FPM. The average rate of climb to 45,000 feet is approximately 2,700 FPM, with a cabin rate of climb of 500 FPM. The departure runway field elevation altitude is then programmed into the LDG (Landing) dial. Destination runway field elevation altitude (4,000 feet in this example) is set during descent. At this point, the pressurization system is programmed for automatic operation and, assuming the AUTO/MANUAL switch is in AUTO, the system will operate in automatic mode when electrical power is applied

to the aircraft.

With the airplane on the ground, the AUTO/MANUAL switch in AUTO and the FLIGHT/LANDING switch in LANDING, the outflow valve will cycle open as soon as electrical power is applied. This is due to the absence of airflow and the cabin pressurization transducer sensing an on-ground condition from the nutcracker.

With the main entrance door still open at this point, there is no pressure buildup. A negligible pressure increase (approximately 0.05 psid, equalling pressure drop across the outflow valve) occurs after closing the main entrance door, remaining negligible through engine start, APU shutdown and transfer of air source to the engines. During taxi, the pressure remains low until lineup in preparation for takeoff. At this point, the flight crew selects the FLIGHT position on the FLIGHT/LANDING switch on the CABIN PRESSURE CONTROL panel.

With the FLIGHT position selected, the outflow valve immediately begins to close under rate control, “holding” at approximately 0.25 psid of pressure buildup. This “holding”, known as ground differential pressure control, serves to minimize the possibility of “pressure bumps” during takeoff.

As soon as the aircraft becomes airborne, the nutcracker system sends an inflight signal to the cabin pressurization transducer. The transducer in turn commands the outflow valve to close, thus pressurizing the cabin at the programmed rate of 500 FPM UP until the maximum differential pressure of 9.55 ±0.10 psid is reached. The cabin altitude remains stable within ±25 feet of the final cabin altitude, in this case 6,550 feet at an aircraft altitude of 45,000 feet. Stability is maintained provided the aircraft does not climb above 45,000 feet, the maximum allowable altitude of the aircraft.

In preparation for descent, the flight crew enters the destination runway field elevation altitude (4,000 feet in this example) in the LDG window. A discriminator circuit is incorporated within the cabin pressurization transducer to prevent inadvertent programming of an isobaric cabin altitude (CABIN value) lower than the landing field altitude (LDG value). This feature prevents landing with excessive cabin differential pressure. If the discriminator circuit detects a LDG value higher than the CABIN value, it allows the LDG value to override the CABIN value and control that outflow valve. This is the only instance in which the LDG value can override the FLIGHT position on the FLIGHT/LANDING switch while in the AUTO mode. When ready to descend, the FLIGHT/LANDING switch is positioned to LANDING.

Noting that the cabin rate of climb was preprogrammed to be 300 FPM DOWN prior to takeoff, the flight crew adjusts the BARO CORR (barometric correction), if necessary. This completes the descent programming. As the aircraft descends, the system automatically begins to open the outflow valve until a descent rate of 300 FPM is reached. Cabin pressure is maintained at approximately 400 to 500 feet below the actual LFE, or at approximately 0.25 psid.

Upon touchdown, the nutcracker system signals the cabin pressurization transducer of an on-ground condition, causing the outflow valve to be driven fully open. With the outflow valve fully open, cabin pressure drops to a negligible pressure of approximately 0.05 psid until the air supply is shut off.

B. Manual Operation Mode:

To operate the pressurization system manually, the AUTO/MANUAL switch, located on the CABIN PRESSURE CONTROL panel, is selected to MANUAL. This causes the MANUAL switch legend to illuminate amber, the illumination of an amber light (referred to as the motor power indicator) above the AUTO/MANUAL switch and display of an amber CABIN PRES MANUAL caution message on CAS.

The outflow valve may then be controlled by the flight crew, using its DC motor, by means of a knob on the CABIN PRESSURE CONTROL panel. The knob is spring-loaded to return the vertical position, labeled MAN HOLD, when released. Rotating the knob toward OPEN drives the valve open; rotating the knob toward CLOSE drives the valve closed.

The manual control knob circuit is designed such that the further the knob is moved away from MAN HOLD toward OPEN or CLOSE, the more pulses are applied to the motor, i.e., the faster the valve moves in that direction. The amber motor power indicator above the AUTO/MANUAL switch will blink in proportion to the speed of the pulses applied to the motor.

Valve position is shown by an indicator to the right of the control knob. The indicator displays outflow valve position at all times, whether in AUTO or MANUAL modes of operation. When desired valve position is attained, releasing the knob will return the knob to MAN HOLD and the outflow valve will hold (stop) in that position.

To return system to automatic mode, the AUTO/MANUAL switch is selected to AUTO. The annunciations will be extinguished and the MAN HOLD knob function will become inoperative.

NOTE:

On aircraft having ASC 295 incorporated, loss of AC power will automatically switch the pressurization system to MANUAL control. This causes the MANUAL switch legend to illuminate amber, the illumination of an amber motor power indicator above the AUTO/MANUAL switch and display of an amber CABIN PRES MANUAL caution message on CAS.

C. Cabin Pressure Safety Valve Limiting Mode:

Should the AUTO or MANUAL control mode of the pressurization system malfunction and cabin pressure builds up to approximately 9.7 ±0.1 psid, the safety valve will open. It then modulates to limit cabin pressure to the safety relief pressure of 9.8 psid.

D. Depressurization Rate Limiting Mode:

Depressurization rate limiting prevents excessive rates of pressure loss, as might occur, for instance, if a malfunction drives the outflow valve to the full open position. Detection of an excessive rate of pressure loss triggers the cabin rate pressure switch, in turn causing the outflow valve to be driven fully closed and automatic control of the system is inhibited. The MANUAL legend of the AUTO/MANUAL switch illuminates amber and an amber CABIN PRES MANUAL caution message is displayed on CAS.

The flight crew can restore the system to automatic operation after

depressurization rate limiting by first hard-selecting the AUTO/MANUAL switch to MANUAL, enabling normal manual control. Reset is then completed by hard-selecting the AUTO/MANUAL switch to AUTO.

E. Pressurization System Check:

The flight crew normally performs the following pressurization system check in the course of their normal procedures:

(1) Ensure the main entrance door is open.

(2) Configure APU BLEED AIR, L ENG BLEED AIR and R ENG BLEED AIR as required.

(3) Select the FLIGHT/LANDING switch to LANDING (green).

(4) Select the AUTO/MANUAL switch to AUTO (green).

(5) Select the ADC #1/ADC #2 switch to ADC #1 (green).

(6) Verify the outflow valve is OPEN on the position indicator. Perform system check as follows:

(7) Select the FLIGHT/LANDING switch to FLIGHT (green).

(8) Verify the outflow valve is driving toward CLOSE. At the midway point:

(9) Select the ADC #1/ADC #2 switch to ADC #2 (amber).

(10) Verify the outflow valve drives fully to CLOSE.

(11) Select the FLIGHT/LANDING switch to LANDING (green).

(12) Verify the outflow valve is driving toward OPEN. At the midway point:

(13) Select the ADC #1/ADC #2 switch to ADC #1 (green).

(14) Verify the outflow valve drives further toward OPEN. Before the valve reaches fully OPEN:

(15) Select the AUTO/MANUAL switch to MANUAL (amber). Verify amber “cabin pressure manual” light illuminates.

(16) Verify manual control of outflow valve to OPEN and CLOSE positions using the manual control knob.

(17) Select the AUTO/MANUAL switch to AUTO (green).

(18) Verify the outflow valve drives fully OPEN if not already fully OPEN. Set final configuration as follows:

(19) Select the FLIGHT/LANDING switch to LANDING (green).

(20) Select the AUTO/MANUAL switch to AUTO (green).

(21) Select the ADC #1/ADC #2 switch to ADC #1 (green).

(22) Verify the outflow valve is OPEN on the position indicator.

(23) Verify amber “cabin pressure manual” light is extinguished.

4. Controls and Indications:

(See Figure 4 and Figure 5.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
CABIN PRESS 115V	PO	D-11	ESS AC Bus, ϕA

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
CABIN PRESS 28V	PO	B-11	ESS DC Bus
CABIN PRESS IND	PO	A-12	ESS DC Bus

B. Warning (Red) CAS Messages:

CAS Message:	SWLP Indication	Cause or Meaning:
CABIN DFRN-9.8	None	Cabin differential pressure approaching upper limit (9.8).
CABIN PRESSURE LOW	CAB PRESS LOW	Cabin altitude has climbed above limits (9250 ft ±750 ft).

C. Caution (Amber) CAS Messages:

CAS Message:	Cause or Meaning:
CABIN DFRN-9.6	Cabin differential pressure has reached 9.6 psi.
CABIN PRES MANUAL	Cabin pressurization controller has been switched to MANUAL control, either automatically or manually.

D. Other Annunciations:

Indication:	Cause or Meaning:
Amber Cabin Pressure Manual Light	System in MANUAL control mode.
Amber MANUAL Legend On AUTO/ MANUAL Switch	System in MANUAL control mode.
Amber ADC #2 Legend On ADC #1 / ADC #2 Switch	ADC #2 providing static pressure signal to transducer.

5. Limitations:

A. Flight Manual Limitations:

(1) Cabin Pressurization Control System:

(a) Maximum Cabin Pressure Differential Permitted: 9.80 psi

(b) Maximum Cabin Pressure Differential Permitted For Taxi, Takeoff Or Landing: 0.3 psi

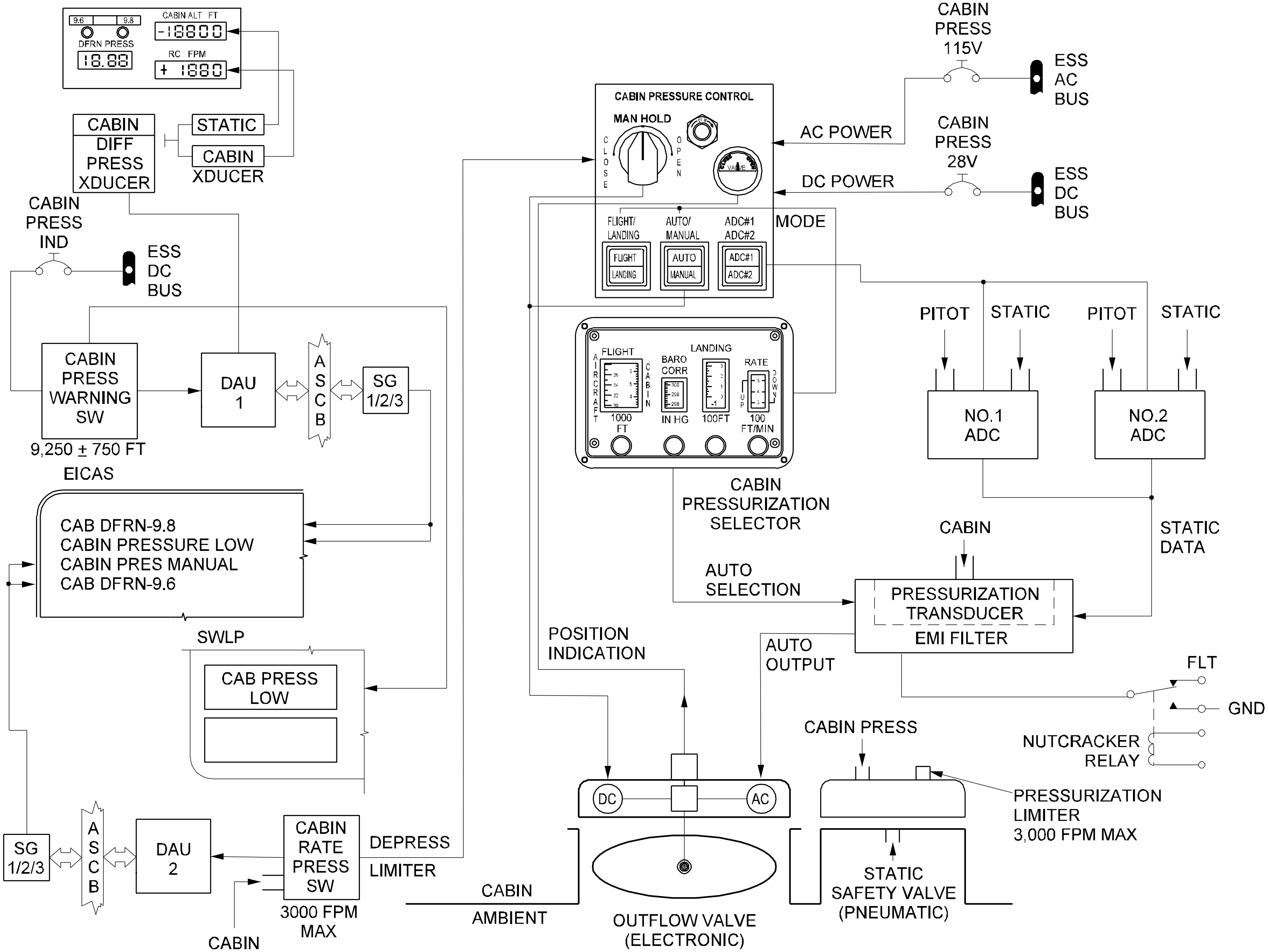
(2) Bleed Air System:

Do not operate above 41,000 ft without both engine bleeds ON and each engine being bled by either the air conditioning system or engine cowl anti-ice. See Section 05-01-10, Air Conditioning System Shut Down Or Inoperative.

B. Operational Data:

Function:	Value:
Normal Maximum Pressure Differential	9.55 ±0.1 psi
Safety Pressure Relief	9.70 ±0.1 psi
Maximum Negative Differential	-0.25 psi
Pressurization Rate Limiting	3,000 FPM
Depressurization Rate Limiting	3,000 FPM

Function:	Value:
Ground Differential Reference Signal	0.5 In Hg
Rate-of-Change Control	±10% of selected value at all cabin altitudes
Barometric Correction Range	28.00 to 31.00 In Hg Absolute Pressure
Cabin Altitude-Isobaric Programming Range	-1000 to 15,000 feet
Landing Altitude Selection Range	-1000 to 15,000 feet
Rate-of-Change Selection Range	Minimum: • 50 FPM UP • 30 FPM DOWN Maximum: • 2000 FPM UP • 2000 FPM DOWN
Rate to Maximum Differential Control Transition	Not to exceed 50 feet with no overshoot beyond the final control value.
Final Absolute Control Pressure	Within 140 feet of selected value at all cabin inflow rates from outflow valve flow of 5 ppm to maximum flow.



Cabin Pressure Control System Block Diagram
Figure 1

TYPICAL FLIGHT PLAN

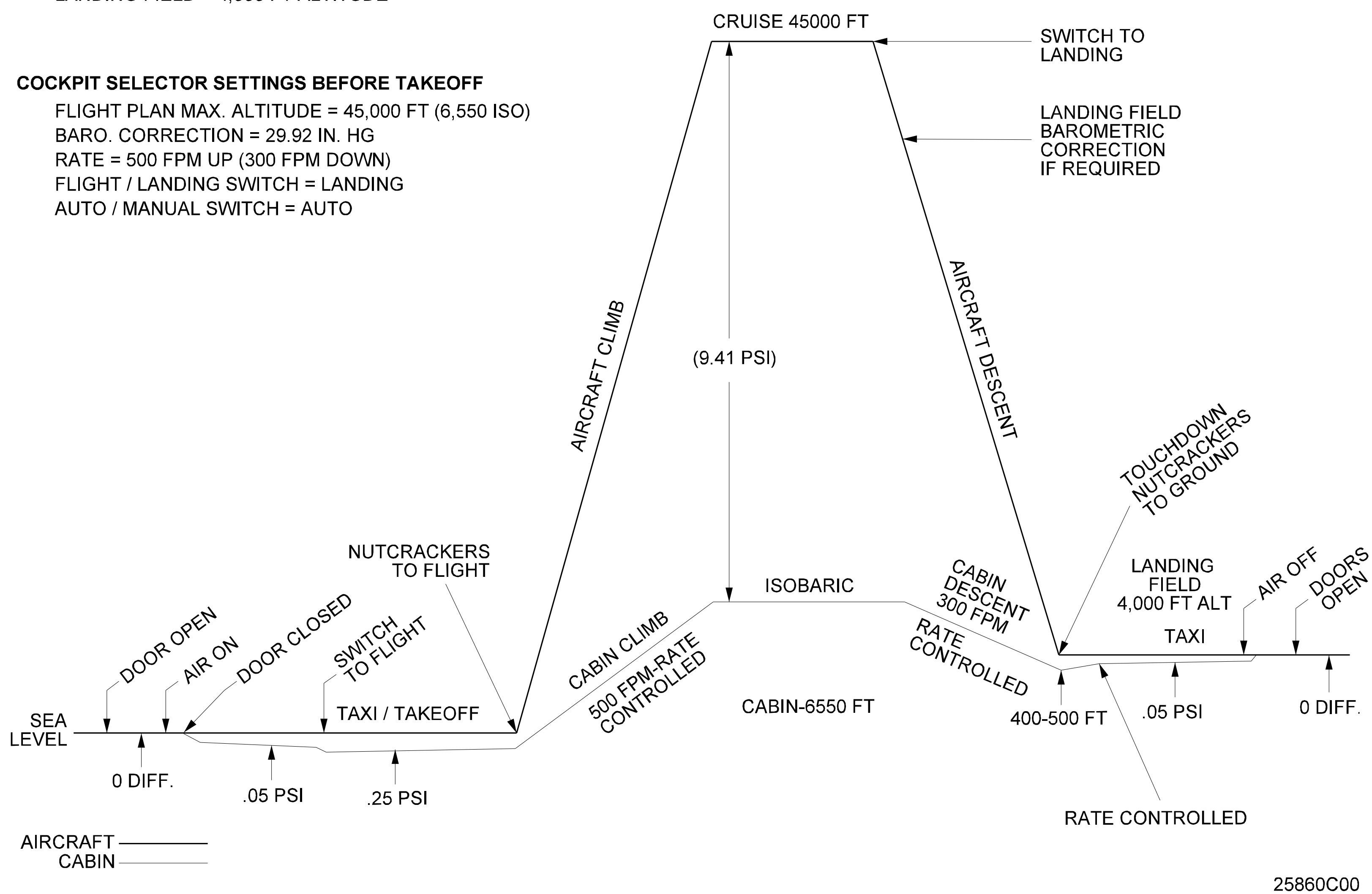
TAKEOFF FIELD = SEA LEVEL ALTITUDE
29.92 IN. HG PRESSURE

CRUISE ALTITUDE = 45,000 FT

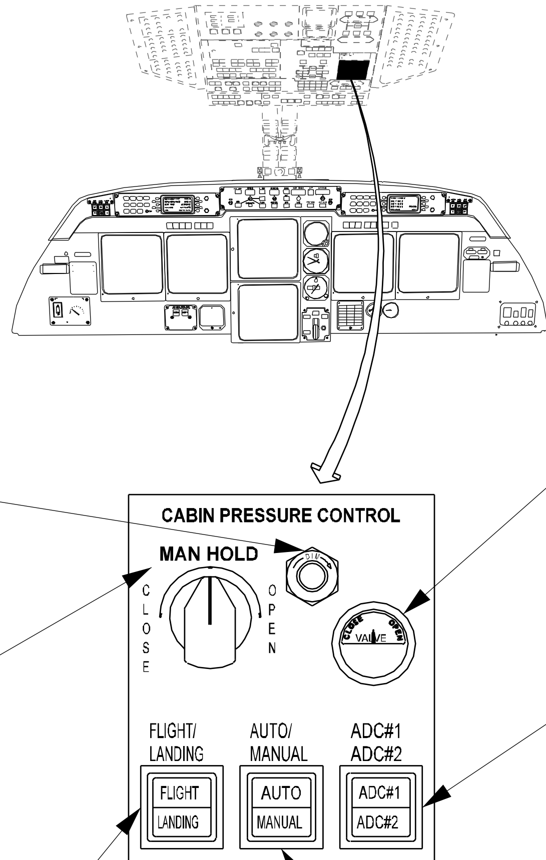
LANDING FIELD = 4,000 FT ALTITUDE

COCKPIT SELECTOR SETTINGS BEFORE TAKEOFF

FLIGHT PLAN MAX. ALTITUDE = 45,000 FT (6,550 ISO)
BARO. CORRECTION = 29.92 IN. HG
RATE = 500 FPM UP (300 FPM DOWN)
FLIGHT / LANDING SWITCH = LANDING
AUTO / MANUAL SWITCH = AUTO



Typical Flight Profile
Figure 2



MOTOR POWER INDICATOR

Illuminates amber to indicate DC power is provided to the outflow valve motor in the MANUAL mode. Flashes proportionate to amount MAN HOLD knob is rotated.

OUTFLOW VALVE POSITION INDICATOR

- Pointer moves to indicate outflow valve butterfly position.
- Functional in all modes of operation.

MAN HOLD

In MANUAL mode, provides DC motor control of outflow valve position. Knob is spring-loaded to HOLD. Outflow valve remains in last selected position.

ADC #1 / ADC #2

- Allows selection of the ADC to supply static pressure information to the pressurization transducer.
- ADC #1 (Green): Primary ADC.
- ADC #2 (Amber): Alternate ADC.

FLIGHT/ LANDING

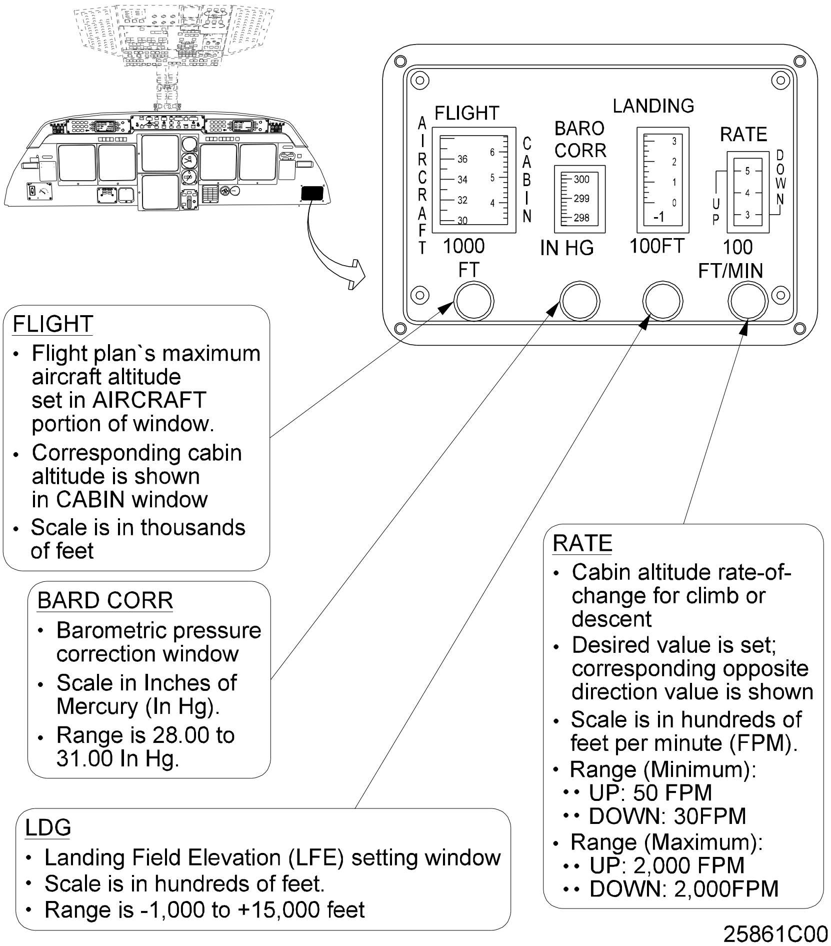
- Allows selection of pressurization schedule.
- FLIGHT (Green): Closes outflow valve; commences and maintains pressurization schedule.
- LANDING (Green): Opens outflow valve; commences and maintains depressurization schedule.

AUTO/ MANUAL

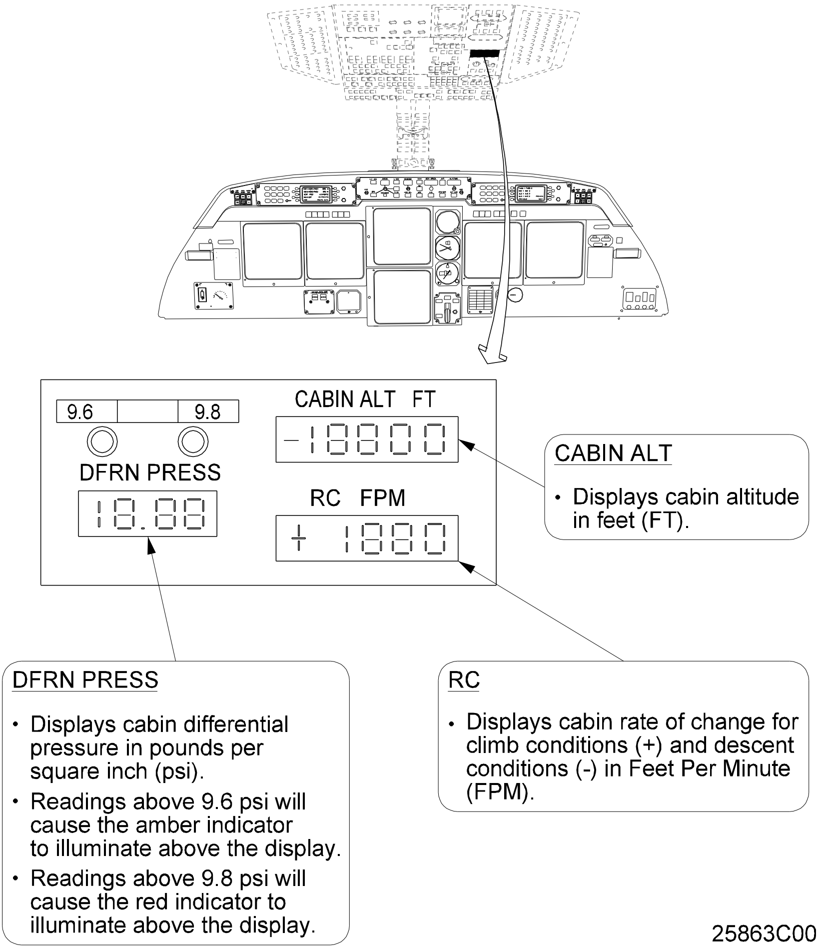
- Used to select and display pressurization system mode of operation.
- AUTO (Green): Automatic mode; normal mode of operation. Pressurization transducer controls outflow valve.
- MANUAL (Amber): Manual mode of operation has been hard-selected or AC power has been lost (aircraft with ASC 295) or depressurization rate limiting has been initiated. MAN HOLD knob controls outflow valve.

25862C01

CABIN PRESSURE CONTROL Panel
Figure 3



Cabin Pressurization Selector Panel
Figure 4



Cabin Differential Pressure / Altimeter / Rate-of-Climb Indicator
Figure 5

2A-21-30: Airflow and Temperature Control System

1. General Description:

The airflow and temperature control system for the Gulfstream IV provides for comfortable cabin and cockpit temperatures throughout the operating envelope of the aircraft by enabling the flight crew to perform the following functions:

- Select and control bleed air entering and exiting two Environmental Control System (ECS) refrigeration pack assemblies, referred to as the Left and Right ECS Packs. This dual pack concept provides redundancy in the event one pack should fail. Source air into the ECS packs is provided to a bleed air manifold by either an approved external air source or APU air (while on the ground), or by High Pressure (HP) turbine bleed air from

either or both of the aircraft's engines (on ground or in flight). Through use of an isolation valve, air from the bleed air manifold can be directed to either ECS pack. This results in a constant mass of conditioned air for all areas within the pressure vessel.

- Control the temperature of conditioned air delivered to the cockpit and cabin areas (referred to as zones) within the pressure vessel. This is done using the two ECS packs to cool incoming air and deliver it to a conditioned air manifold. Valves mix hot bleed air with cold conditioned air to modulate the temperature of the air coming from the manifold into the pressure vessel. Water separation, for humidity reduction, is also provided.

During normal inflight operation, temperature-controlled and pressure-controlled HP bleed air from either or both engines is supplied to the bleed air manifold. Air from the manifold is provided to the ECS packs, where cooling takes place. Each pack consists of a primary heat exchanger, secondary heat exchanger and an air cycle machine (ACM). (The term "air cycle" means that cooling is produced by a thermodynamic cycle, using only air as a medium, as opposed to a vapor cycle, which employs Freon™ or other similar gases.) The ECS packs then reduce air temperature to values above freezing. Humidity reduction is accomplished by a mechanical water separator.

Temperature control of the zones within the pressure vessel is accomplished by varying the amount of hot bleed air which bypasses the cooling equipment. Separate temperature control is provided for the cockpit and cabin zones using a control panel located on the overhead panel in the cockpit. Provisions for manual control of the system are included in the event of a failure rendering electronic control inoperative.

An advantage of the GIV airflow and temperature control system is that it is capable of functioning independently while on the ground. With the engines not operating, the bleed air manifold can be supplied with air from either the APU or from an approved external air cart. System operation on the ground is virtually the same as in flight, the difference being that ram air flow across the heat exchangers is provided by a cooling fan.

Should either or both engines be operating, the flight crew may select either or both engines to supply bleed air to the manifold through use of the ISOLATION valve switch.

During certain emergency procedures, the flight crew may induce ram air ventilation into the airflow and temperature control system. Ram air is supplied from a dorsal fin duct and controlled by a RAM AIR switch located on the overhead panel in the cockpit.

For the purposes of this description, the airflow and temperature control system is divided into the following subsystems:

- Air Control System
- Temperature Control System
- Distribution System
- Refrigeration System
- Ram Air Ventilation System
- Temperature Indication System
- Equipment Cooling System

2. Description of Subsystems, Units and Components:

A. Air Control System:

(See Figure 6 and Figure 7.)

(1) Bleed Air Manifold:

The bleed air manifold is used as the source of bleed air for the air conditioning system. Air to the manifold is supplied by either an approved external air source or APU air (while on the ground), or by HP turbine bleed air from the aircraft's engines (on ground or in flight).

The bleed air manifold delivers air to using systems, one being the air conditioning system. Delivered air is approximately 400° F at a maximum pressure of 40 psig. For the air conditioning system, air is delivered to the air conditioning shutoff and flow regulating valves in the tail compartment.

(2) Air Conditioning Shutoff and Flow Regulating Valves:

The air conditioning shutoff and flow regulating valves perform two functions in the air conditioning system:

- Act as shutoff valve when system operation is terminated.
- Act as a flow regulator when the system is operating.

The air conditioning shutoff and flow regulating valve is pneumatically operated butterfly valve, using upstream duct pressure as the operating force. An internal electrical solenoid is installed and, when energized, pressurizes the valve to close it. This prevents air entry, ending system operation. When de-energized, upstream duct pressure opens the butterfly valve and airflow starts again. Airflow through the valve is regulated to a maximum of 42.1 (±1.5) pounds per minute (ppm) of flow.

There are several ways to energize the solenoid and close the valve:

- Selection of the RAM AIR switch to RAM.
- Selection of the L PACK or R PACK switch to OFF.
- ACM compressor discharge temperature reaching 450° F (on ground).

For aircraft 1000 through 1155 (excluding 1034) having ASC 135, aircraft 1034, and aircraft 1156 and subsequent, the following additional functions exist:

- Selection of the MASTER CRANK or MASTER START switch closes the LEFT ECS PACK valve (on ground). The valve will automatically reopen when the MASTER CRANK or MASTER START switch is deselected.
- Selection of the L ENG START or R ENG START switch closes the RIGHT ECS PACK valve (on ground). The valve will automatically reopen when the start valve closes.

Exiting the air conditioning shutoff and flow regulating valve, air flows through an ozone filter that reduces concentration to a maximum of 0.1 part per million. The air is then delivered to the temperature control system and split into two paths. One path is

routed to the primary heat exchangers (for primary cooling) and the other path is routed to the cabin and cockpit temperature control valves (bypass air).

B. Temperature Control System:

(See Figure 8.)

Air leaving the ozone filter is delivered to the temperature control system and can take two paths: either through the refrigeration unit or through the cabin and cockpit temperature control valves. The amount of air that can pass through the temperature control valves is dependent upon valve butterfly position. The remaining bleed air is delivered to the refrigeration unit to be cooled. This now cold air then rejoins the hot bleed air from the temperature control valves, thereby becoming temperature-controlled (conditioned) air. The position of the temperature control valve determines the compartment temperature by mixing the hot and cold air to maintain the desired temperature. All temperature control devices in this system are directed toward control of the appropriate temperature control valve.

(1) Cabin/Cockpit TEMP CONTROL Panel:

The Cabin/Cockpit TEMP CONTROL panel, located on the cockpit overhead panel, is used to automatically or manually set a desired cabin or cockpit temperature. Each portion of the panel (CABIN and CKPT) is isolated and independent of the other.

The selector knob provides automatic and manual temperature control selection based upon knob position. In the OFF position, no signal is applied to the temperature control valve from either the manual selector knob or the temperature controller. Clockwise rotation of the knob from OFF places the system in the automatic mode of operation (the normal mode of operation), with the temperature controller providing electrical signals to the temperature control valve. Range of the automatic mode of operation is COLD (60° F) to HOT (80° F).

Counterclockwise rotation of the knob from OFF places the system in the manual mode of operation, with the manual selector knob providing electrical signals to the temperature control valve. Range of the manual mode of operation is COLD (temperature control valve fully closed) to HOT (temperature control valve fully open). Returning the knob to OFF defaults the system to full cold operation.

NOTE:

Some aircraft are outfitted with additional manual temperature controls located aft of the baggage compartment door. These controls, however, are pneumatic, not electric and are dependent upon a minimum 3 psid cabin pressure to manipulate the temperature control valves.

(2) Cabin/Cockpit Temperature Control Valves:

The cabin/cockpit temperature control valve is a pneumatic modulating butterfly valve. Pneumatic pressure is required to open the butterfly and the amount of opening is controlled by, and proportional to, the amount of pneumatic pressure applied to an

internal diaphragm chamber. With no pneumatic pressure applied to the diaphragm chamber, an internal mechanism maintains the butterfly in the closed position.

The pneumatic opening pressure, referred to as servo pressure, originates from pressure ducted from upstream of the valve. The ducted pressure is routed to a servo air pressure regulator and torque motor.

(3) Servo Air Pressure Regulator and Torque Motors:

The servo air pressure regulator and torque motor controls the pressure to the temperature control valve using an electrical signal received from the cabin/cockpit temperature controller. The electrical signal is converted into a pneumatic signal and the pneumatic signal positions the temperature control valve accordingly.

(4) Cabin/Cockpit Temperature Controllers:

The cabin/cockpit temperature controller receives and interprets various inputs in order to derive an output signal. These inputs are:

- Two cabin/cockpit temperature sensors, for ambient temperature
- One cockpit temperature sensor, for ambient temperature
- Two cabin/cockpit duct temperature anticipators, for duct temperature and exhausted air temperature

These inputs are compared to the desired ambient temperature commanded by the TEMP CONTROL panel knob and an output signal is then sent to the servo air pressure regulator and torque motor to position the temperature control valve.

The cabin/cockpit temperature controller receives power from the 28V Right Main DC bus (aircraft 1000 through 1143, excluding 1034) or the Essential DC bus (aircraft 1034, and aircraft 1144 and subsequent).

(5) Cabin/Cockpit Temperature Sensors:

The cabin/cockpit temperature sensors are dual elements consisting of two separate sections. One section provides temperature information to the cabin temperature controllers, while the other element is actually a temperature bulb for the digital cabin air temperature indicator located on the cockpit overhead panel. On airplanes SN 1437 and subsequent and SN 1000 through 1436 having ASC 162A, a second cabin temperature sensor is also installed. It is used for temperature indication only.

(6) Cockpit Temperature Sensor:

The cockpit temperature sensor is a single thermistor sensing element. It is used with the compartment thermostat to provide temperature information to the cockpit temperature controller.

(7) Crossover Function:

Airflow and temperature control capability is maintained in the event of failure of either engine or either ECS pack.

In the event of an engine failure, the BLEED AIR switch for the

operative engine is left ON while the BLEED AIR switch for the inoperative engine is selected OFF. The ISOLATION valve would be opened to allow operative engine bleed air to both ECS packs. Both PACK switches are left ON for full airflow and temperature control capability.

In the event of an ECS pack failure, the PACK switch for the operative pack is left ON while the PACK switch for the inoperative pack is selected OFF. The BLEED AIR switch for the operative pack's engine is left ON for full airflow and temperature control capability.

Depending on flight conditions and mission requirements, problem ECS packs should be managed as necessary. Airplane Flight Manual limitations should be consulted if an ECS pack is to be shut down in flight.

NOTE:

Do not select both ECS packs OFF at altitude.

C. Distribution System:

(See Figure 6 and Figure 7.)

Exiting the temperature control valves, the hot bleed air is joined with the cooled air from the refrigeration unit to become temperature-controlled air. This air is then introduced into the cabin and cockpit distribution systems, each having separate temperature control valves and ductwork.

The cockpit distribution system consists of ducting from the cockpit temperature control valve, the refrigerated air duct, an air duct check valve and a silencer. Final distribution is from four outlets in the cockpit: a controllable side (or shoulder) outlet and a non-adjustable foot outlet, for each pilot.

The cabin distribution system consists of ducting from the cabin temperature control valve, a air duct check valve and a silencer. Final distribution is through two louvered baseboard ducts running most of the cabin's length, one on each side.

Cabin and cockpit check valves are installed in the compartment ducting allowing air to flow only in a forward direction. Should the air attempt to reverse flow, the check valve closes to prevent backflow.

Cabin and cockpit silencers are installed in the ducting under the floor to silence air noise from the bleed air ducts.

D. Refrigeration System:

(See Figure 7.)

Bleed air which does not bypass the cabin and cockpit temperature control valves is routed into the refrigeration unit (ECS pack). The ECS pack consists of the following major components:

- Primary Heat Exchanger
- ACM and ACM Overtemperature Thermal Switch
- Secondary Heat Exchanger
- Water Separator System

- Water Separator Anti-Ice System
- Cooling Fan
- Cooling Air Distribution

(1) Primary Heat Exchanger:

The primary heat exchanger is the first stage of the refrigeration process. Ram air from the dorsal fin inlet is used as a coolant. The heat exchanger is a “single pass” type exchanger and is located in the tail compartment.

Air exiting the heat exchanger is split into two ducts. One duct routes air into the eye of the compressor section of the ACM. The other duct routes air through the water separator anti-ice valve into the anti-ice muff assembly, bypassing the ACM and secondary heat exchanger.

(2) Air Cycle Machine (ACM) and ACM Overtemperature Thermal Switch:

The ACM is an expansion turbine which reduces air temperature by causing the air to perform useful work, resulting in lower air pressure and thus, lower air temperature. The work extracted from the airstream in the turbine section is absorbed a compressor wheel directly shafted to the turbine wheel and located in a separate chamber on the upstream side of the unit. A large percentage of the work extracted from the airstream by the turbine wheel is used by the compressor wheel. As the compressor wheel is performing work on the upstream air, its pressure and temperature are increased. This work arrangement is called a pressure recovery system or “bootstrap” system.

With the ACM in full operation (no airflow to the anti-ice valve), airflow moves through the compressor section to the secondary heat exchanger, then to the turbine section nozzle, exiting out the eye of the turbine section into the mixing muff.

A 450° F thermal switch is incorporated into the discharge side of the bootstrap compressor to monitor discharge air temperature. If a malfunction causes low or no airflow across the heat exchangers, the compressor discharge temperature will rise accordingly. At 450° F, the switch causes an amber L-R COOL TURB HOT message to be displayed on CAS. Additionally, when on the ground, the ECS pack will be automatically shut off due to a protection circuit passing through the ground configuration of the nutcracker system.

An amber L-R COOL TURB HOT CAS message in flight may be the result of either an excessive air supply to the ACM, resulting in an overspeed/overtemperature condition, or an air bearing failure in the ACM. Excessive airflow from 12th stage compensation can be the result of operating cowl anti-ice and/or wing anti-ice at high altitudes.

(3) Secondary Heat Exchangers:

Secondary heat exchangers are installed adjacent to the primary heat exchangers. Ram air from the dorsal fin inlet is used as a coolant.

(4) Water Separator System:

Air expansion inside the cooling turbine and resultant discharge temperatures below ambient forces moisture into the air where it condenses. A two-section water separator provides a means for removing the water.

The first section consists of a coalescer which transforms the many small droplets into a few large drops by forcing the water through a coarse mesh cloth bag. In the second section, the airstream is forced to swirl by means of a series of vanes so that the large drops are spun to the outside walls. Water extracted from the air is also sprayed into the secondary heat exchanger cooling air inlet to assist in cooling.

The water separator is capable of removing approximately 80% of all liquid state water passing through it but it cannot, however, remove water vapor. The water separator also contains a relief valve which, if the coalescer becomes clogged, bypasses the air through the unit. In this case, dehumidification would not take place.

(5) Water Separator Anti-Ice System:

During cool, moist conditions, cooling turbine discharge temperatures can fall low enough that condensed water freezes. To prevent the coalescer from becoming clogged with ice crystals and restricting airflow, a water separator anti-ice system is installed in the tail compartment. The system consists of a water separator anti-ice valve, a sensor and a bypass duct mixing muff assembly.

The water separator anti-ice valve is a butterfly-type shutoff and modulating valve. It controls the refrigeration unit cold air outlet temperature to a minimum of 37° F (nominal) by modulating the flow of compressor inlet air to the anti-ice muff at the turbine discharge. Valve position is controlled by a torque motor in response to signals received from the water separator anti-ice sensor/controller.

The water separator anti-ice sensor/controller is a pneumatic thermostat, installed on the discharge side of the water separator. The thermostat is set to maintain anti-ice valve position so that air moving through the water separator is held at a temperature of approximately 37° F. The sensor/controller remains function throughout all altitude, temperature and humidity ranges.

One cockpit indication that a water separator may be frozen is an absence of cool air in all modes of operation. To determine whether the water separator is merely frozen or the ACM is faulty, the affected PACK switch is selected OFF and a ten minute time period is allowed to elapse. The affected PACK switch is then selected ON and the air temperature is checked. If cool air returns, the water separator was most likely frozen, thus a warmer temperature should be selected. If warm or hot air returns, the ACM is most likely faulty.

(6) Cooling Fan:

Ground cooling places additional requirements on the air conditioning system. As there is no ram air flow, the possibility exists that the air conditioning equipment would overheat unless a source of cooling air is supplied.

A turbofan is installed downstream of the ACM in the dorsal fin ram

air duct. The fan assembly provides airflow across the primary and secondary heat exchangers whenever the air conditioning system is in operation on the ground. Fan air leaving the heat exchangers is then exhausted overboard.

(7) Cooling Air Distribution:

Dehumidified, refrigerated air from the discharge side of the water separator is ducted forward, joining that portion of the hot bleed air having passed through the cabin/cockpit temperature control valves to become temperature-controlled air. Also, dehumidified, refrigerated air from the ECS pack is ducted into one line which supplies cabin and cockpit eyeball ducts installed by the operator's completion agency. A check valve is installed in each line to prevent backflow.

E. Ram Air Ventilation System:

In the event of an emergency, the flight crew can ventilate the aircraft with ram air from the ram air dorsal fin inlet. A line is installed in the ram air duct just upstream of the primary heat exchanger. The line is routed to the ram air check valve and then to the downstream side of the left water separator refrigerated air duct. If ram air duct pressure is greater than refrigerated air duct pressure, the ram air check valve opens, allowing ram air into the refrigerated air line. The ram air then moves forward through the cabin duct check valve, into the distribution system.

The ram air check valve allows airflow to move only from the ram air duct into the air conditioning system ducting. During normal operations with the air conditioning system running, system pressure is always greater than ram air pressure, thus the ram air check valve is held closed.

Selection of ram air ventilation is accomplished using the RAM AIR switch located on the cockpit overhead panel. Selection of the switch to RAM supplies 28V Essential DC bus power to close both air conditioning shutoff and flow regulating valves, shutting off the air conditioning system. As air conditioning system duct pressure falls below ram air duct pressure, the ram air check valve opens, allowing ram air into the distribution system. Conversely, selection of the RAM AIR switch to OFF allows both air conditioning shutoff and flow regulating valves to open, restoring the air conditioning system. As air conditioning system duct pressure rises above ram air duct pressure, the ram air check valve closes, allowing air conditioning system air into the distribution system.

NOTE:

Selection of the RAM AIR switch to RAM results in the cabin altitude climbing, eventually causing the pressurization system outflow valve to close. Consideration should be given to manually opening the outflow valve to ensure adequate airflow for radio rack cooling.

NOTE:

During use of ram air ventilation, the flight crew has no control of cabin air pressure or temperature.

F. Temperature Indication System:

(See Figure 8.)

A digital cabin/cockpit air temperature indicator is installed on the TEMP CONTROL panel on the cockpit overhead panel. The CABIN temperature display is provided ambient temperature information from the cabin temperature sensor. The CKPT temperature display is provided ambient temperature information from the cockpit temperature sensor. Both sensors also provide temperature information to their respective temperature controller. Location of the sensors is determined by the operator's completion agency.

Power for the indicator is furnished by the 28V Essential DC bus through the CKPT/CABIN TEMP IND circuit breaker. The indicator is calibrated in degrees Fahrenheit, with a maximum display value of 199° F.

G. Equipment Cooling System:

(See Figure 9.)

Electric cooling fans are installed to force cooling air to the display unit cathode ray tubes (CRTs), center pedestal equipment, radio racks and nose compartment. With electrical power applied, operation of most fans is automatic and is transparent to the flight crew.

For aircraft 1000 through 1155 (excluding 1034) having ASC 87, and aircraft 1034, 1156 and subsequent, selection of the PILOT, EICAS or COPILOT DISPLAY switch while on the ground energizes both CRT cooling fans. In flight, the fans operate continuously, regardless of switch position.

Supplemental cooling air for center pedestal equipment is provided by a fan mounted in the aft right side of the pedestal. The fan is not thermostatically controlled and operates whenever 28V Right Main DC bus power is available.

With electrical power applied and the main entrance door open, the right-hand radio rack cooling fan operates. On aircraft 1156 and subsequent, a RH RR FAN MAN ON switch is installed on the copilot's side console to provide additional manual control of the right-hand radio rack fan. Additionally, blue RH RR FAN AUTO, RH RR FAN FAIL and LH RR FAN FAIL annunciators are installed on the copilot's side console to show radio rack fan status.

With the aircraft on the ground and nose compartment temperature above 90° F, a thermal switch opens the nose compartment cooling valve. This energizes a fan and illuminates an amber N COOL VALVE OPEN annunciator on each side of the flight panel. Once airborne, a nutcracker relay closes the valve. Should the valve fail to close in flight, the N COOL VALVE OPEN annunciators will illuminate.

3. Controls and Indications:

(See Figure 8 and Figure 9.)

A. Circuit Breakers (CBs):

The airflow and temperature control system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
L AIR COND	PO	C-13	ESS DC Bus
R AIR COND	PO	D-13	ESS DC Bus
CABIN PRESS 115V	PO	D-11	ESS AC Bus, ϕA
CABIN PRESS 28V	PO	B-11	ESS DC Bus
CABIN PRESS IND	PO	A-12	ESS DC Bus
CABIN TEMP CONT	PO	D-14	ESS DC Bus
CKPT/CABIN TEMP IND	PO	A-11	ESS DC Bus
CKPT TEMP CONT	PO	B-13	ESS DC Bus
DISPLAYS FAN #1	CP	D-5	ESS DC Bus (1)
DISPLAYS FAN #2	CP	D-6	R MAIN DC Bus (2)
NOSE COMPT COOL FAN	CP	M-1	R MAIN DC Bus (3)
NOSE COMPT COOL VLV	CP	L-1	R MAIN DC Bus
PED COOL FAN	CP	J-1	R MAIN DC Bus
LH RR COOL FAN	CP	K-1	ESS DC Bus
RH RR FAN CON	CP	I-1	ESS DC Bus
RH RR COOL FAN	CP	H-2	ESS DC Bus
SGL PACK	PO	C-14	ESS DC Bus
L TEMP CONT AC	P	H-11	ESS AC Bus, ϕA
R TEMP CONT AC	P	I-11	ESS AC Bus, ϕA

NOTE(S):

(1) ESS AC bus, ϕA, on aircraft 1000, 1002 through 1095 (excluding 1034) not having ASC 49/49A.

(2) R MAIN AC bus, ϕB, on aircraft 1000, 1002 through 1095 (excluding 1034) not having ASC 49/49A.

(3) R MAIN AC bus, ϕB, on aircraft 1000 and 1002 through 1095, excluding 1034.

B. Caution (Amber) CAS Messages:

CAS Message:	Cause or Meaning:
L-R COOL TURB HOT	Cooling turbine discharge air above 450° F (232° C).
DU FAN 1-2 FAIL	Respective DU cooling fan has failed.
FWD RADIO RACK HOT	Inside radome, left or right equipment bay temperature has exceeded 200° F (93° C).

C. Other Indications:

Indication:	Cause or Meaning:
Amber N COOL VALVE OPEN Annunciator (Pilot's/Copilot's Flight Panel)	Nose compartment cooling valve is open.
Blue LH RR FAN FAIL Annunciator (Copilot's Side Console) **(1)**	Left-hand radio rack cooling fan has failed.
Blue RH RR FAN FAIL Annunciator (Copilot's Side Console) **(1)**	Right-hand radio rack cooling fan has failed.
Blue RH RR FAN AUTO Annunciator (Copilot's Side Console) **(1)**	Right-hand radio rack cooling fan operating automatically.

Indication:	Cause or Meaning:
Blue RH RR FAN MAN ON Annunciator (Copilot’s Side Console) **(1)**	Right-hand radio rack cooling fan operating manually.

NOTE(S):

(1) Aircraft 1156 and subsequent.

4. Limitations:

There are no limitations for the airflow and temperature control system at the time of this revision.

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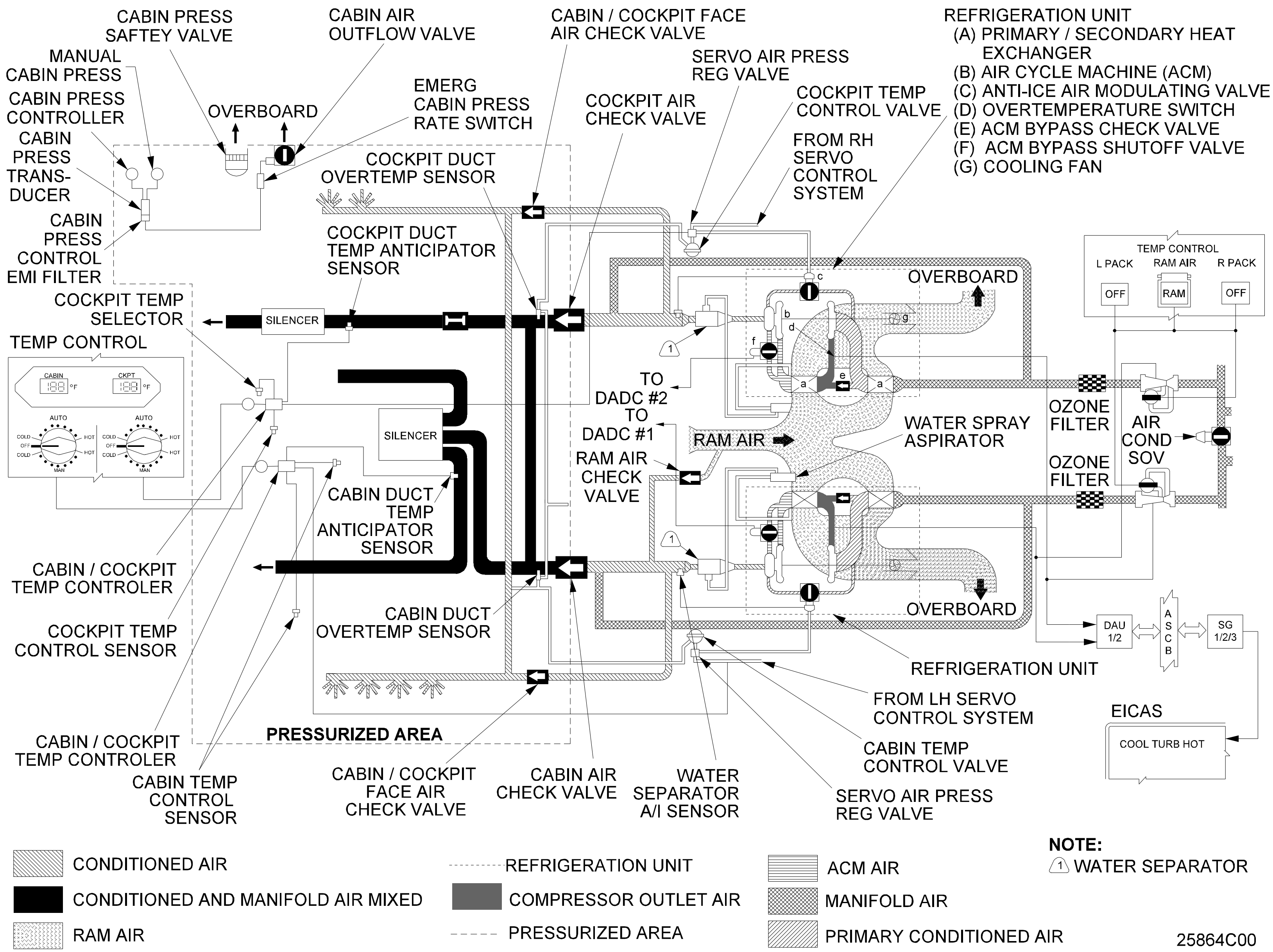
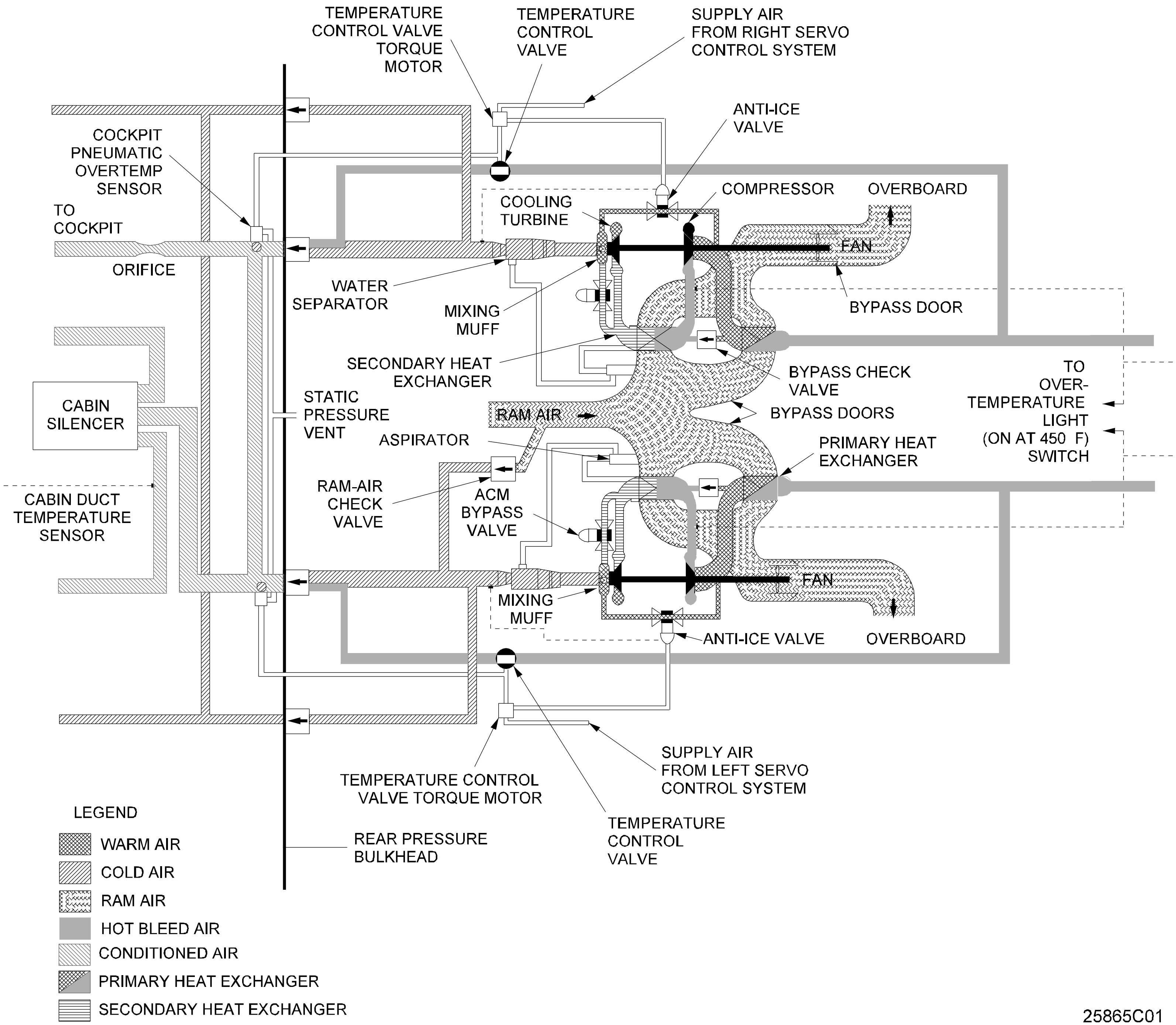


Airflow And Temperature Control System Block Diagram
Figure 6

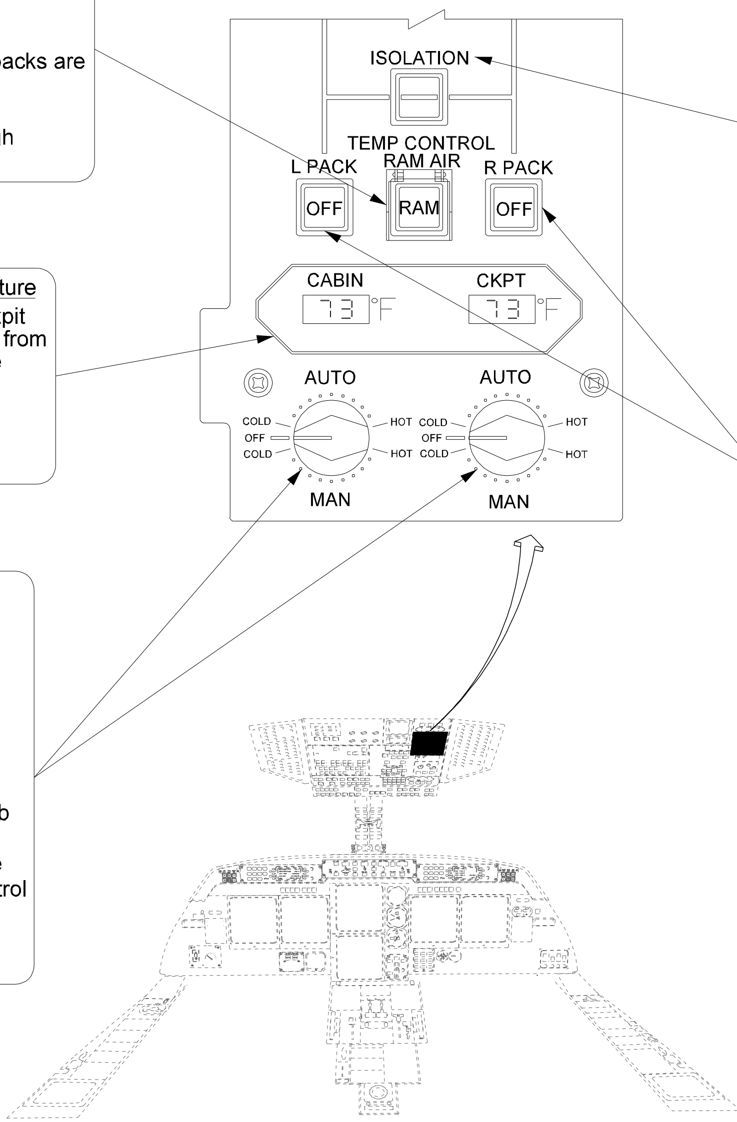


Refrigeration System
Block Diagram
Figure 7

RAM AIR
- OFF (Normal position, RAM legend distinguished): Ram air is inhibited due to ECS pack airflow.
- RAM (Amber): Both ECS packs are shut off. Ram air from dorsal fin duct enters air conditioning system through ram air check valve.

CABIN / CKPT Temperature
- Display cabin and cockpit temperatures obtained from respective temperature sensor.
- Display in degrees Fahrenheit, with a maximum of 199° F.

CABIN / CKPT AUTO / MAN
- OFF: No signal is sent to respective temperature control valve.
- AUTO: Temperature controller provides signal to temperature control valve. Range is COLD (60° F) to HOT (80° F).
- MAN: Manual selector knob provides signal to temperature control valve. Range is COLD (temperature control valve fully closed) to HOT (temperature control valve fully open).



ISOLATION
When selected open (on ground):
- White bar in switch capsule illuminates.
- Left and right bleed air manifolds are combined.
- Crossbleed air from opposite engine is available.
- APU air is available for ECS packs and engine starting.

When selected open (in air):
- Same conditions listed above are present except APU air is available for engine starting only.

When selected to OFF:
- White bar in switch capsule extinguishes.
- Left and right bleed air manifolds are isolated.
- Crossbleed air from opposite engine is inhibited.
- APU air is available for R ENG bleed air manifold only.

L PACK / R PACK
- ON (Normal position; OFF legend extinguished): Air conditioning shutoff and flow regulating valve is de-energized, allowing air conditioning air flow.
- OFF (Amber): Air conditioning shutoff and flow regulating valve is energized closed; air conditioning airflow ceases.

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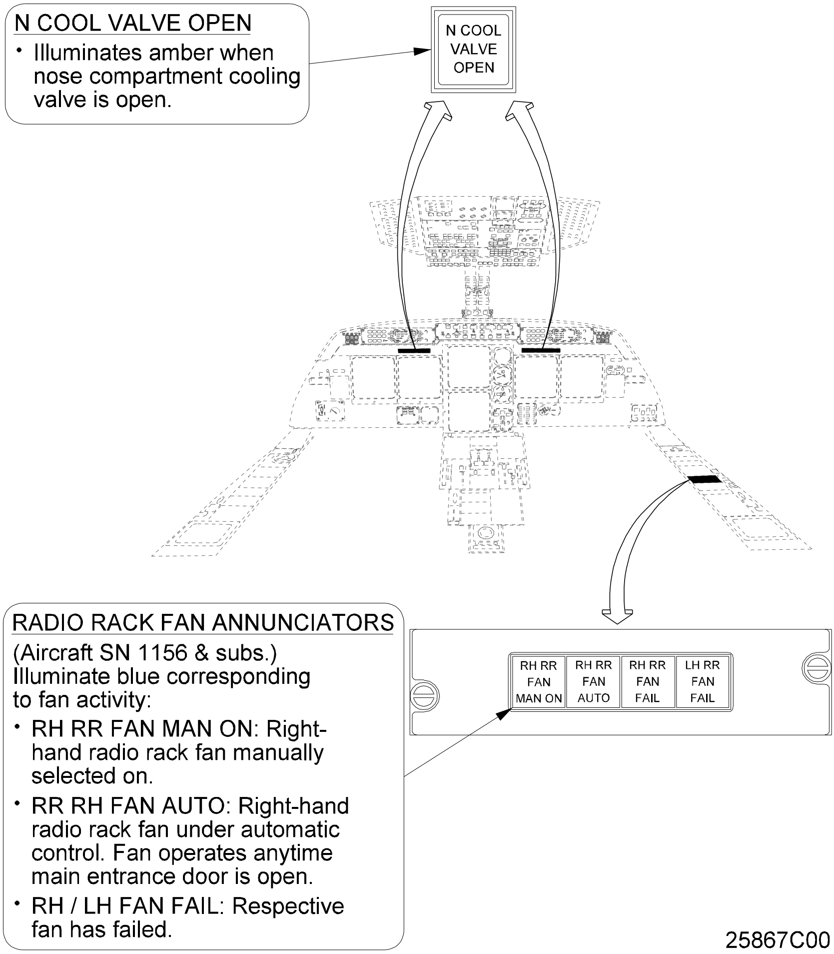
Cabin / Cockpit TEMP
CONTROL Panel
Figure 8



Cockpit Annunciators
Figure 9

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2A-23-10: General

The communications system for the Gulfstream IV, shown in Figure 1, provides the flight crew with a means of communicating with ground stations and other aircraft using Very High Frequency (VHF) and High Frequency (HF) communications equipment. Provisions are also included for intercommunication with others aboard the aircraft and on the ground. Additionally, the most recent thirty (30) minutes of all audio signals transmitted and received by cockpit crew members is recorded by the cockpit voice recorder system in the event it should be needed for investigatory purposes. An ARTEX 406 MHz Emergency Locator Transmitter (ELT) system provides triple frequency homing transmissions when activated.

The communications system is divided into the following subsystems:

- 2A-23-20: VHF Communications System (Collins VHF-422 Transceivers with Gables Digital Frequency Control Units: Aircraft SN 1000 - 1309)
- 2A-23-30: HF Communications System (Collins HF-190 and HF-9000 Series Transceivers)
- 2A-23-40: Integrated Automatic Tuning System (Collins RTU-4200 Series Radio Tuning Unit: Aircraft SN 1310 and subsequent)
- 2A-23-50: Intercommunications System
- 2A-23-60: Cockpit Voice Recorder System
- 2A-23-70: Emergency Locator Transmitter System

Note To Operators: Because of the numerous possible communication system configurations, modifications and software levels, the data contained in these sections is limited to equipment description, controls and indications. For further details and specific operational procedures, the latest approved version of the applicable vendor-supplied pilot’s manuals should be consulted. Where possible, these manuals will be identified in the appropriate section.

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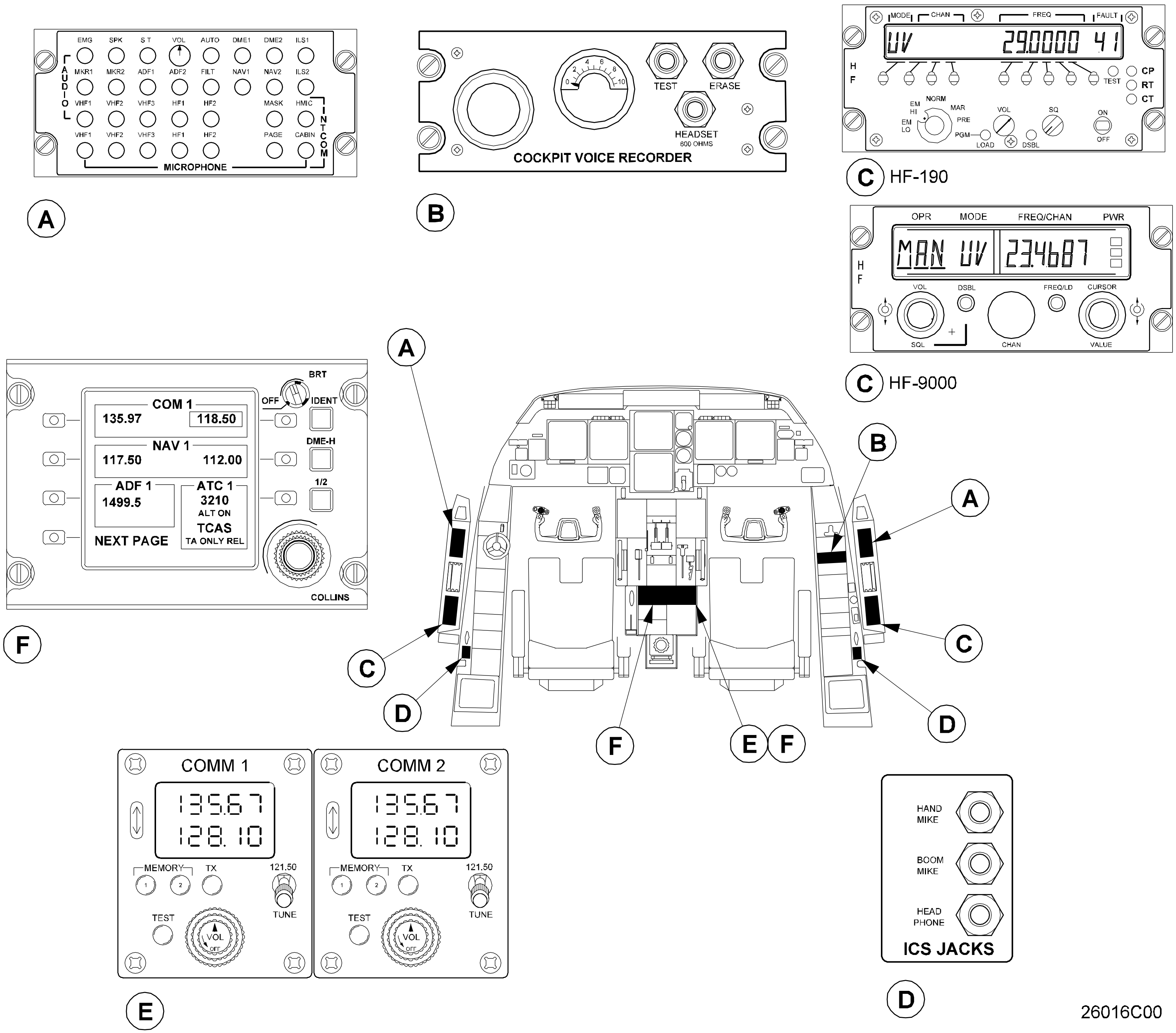


Figure 1. GIV Communications System Components

2A-23-20: VHF Communications System

THIS SECTION APPLIES TO AIRCRAFT SN 1000 - 1309

1. **General Description:**

   The VHF communications system provides the flight crew with two-way VHF communications in the AM frequency range of 118.000 to 151.975 MHz. Channel spacing of 25 kHz provides 1,360 discrete channels.

   For aircraft with 8.33 kHz channel spacing capability, available discrete channels are increased three-fold to 4,080 channels.

   The VHF communications system is further divided into the VHF No. 1 system and the VHF No. 2 system. System design is such that the No. 1 and No. 2 VHF systems are isolated from each other and independent in operation. This helps to prevent any interference problems.

   A typical installation consists of two Collins VHF-422 transceivers installed on a shelf inside the radome, two Gables digital frequency control units and two VHF antennas. In addition to normal system control using the digital frequency control units, the FMS is capable of controlling both VHF systems via the RTE/RTI switch on the Marker Beacon (MKR BCN) controller.

2. **Description of Subsystems, Units and Components:**

   A. **VHF Transceivers:**

   Two Collins VHF-422 transceivers, referred to as the VHF No. 1 transceiver and VHF No. 1 transceiver, are installed on a shelf inside the radome.

   B. **Digital Frequency Control Units:**

   Two Gables digital frequency control units, commonly referred to as the “COMM 1 and COMM 2 control heads”, are installed on the cockpit center pedestal. The COMM 1 and COMM 2 control heads form the communications portion of what is commonly known as the “six pack” configuration. Four other Gables digital frequency control units, two for navigation control (NAV 1 and NAV 2) and two for automatic direction finding control (ADF 1 and ADF 2), complete the six pack concept.

   C. **VHF Antennas:**

   The two VHF antennas (No. 1 and No. 2) are blade-type antennas that operate throughout the frequency range. The VHF No. 1 antenna is located on the bottom of the fuselage along the centerline. It provides full hemispherical coverage. The VHF No. 2 antenna is located on the upper portion of the fuselage. It provides coverage for communication on the ground.

3. **Controls and Indications:**

   (See Figure 2.)

   A. **Display Window:**

   Each control head has a two-section display window. The upper section display consists of a five digit display indication referred to as the **active** display. The lower section display consists of a five digit display indication of the preselected frequency that is available for interchange with active display. It is referred to as the **preselect** display.

B. Power ON Switch / Volume Control:

A rotary switch is located within the frequency selector knobs that is used to apply power to the control head and control volume.

C. Frequency Selection Knobs:

Dual concentric control knobs provide control of the preselect display and frequency / code information encoded into data sent to the radio equipment. A feature of the knob is variable rate tuning, which makes the tuning process much faster. When a single control knob has a wide range to control, the more rapidly the knob is rotated, the more range per detent is covered. Clockwise rotation increases the frequency while counterclockwise rotation decreases the frequency.

The tunable range is from 118.00(0) to 151.97(5) MHz (inclusive) in 25 kHz steps. The 0.001 MHz digit is not displayed.

The 0.1 and 0.01 MHz digits are controlled by the smaller frequency selector knob. Each detent of the knob will vary the display in 0.02(5) MHz increments. The digits range from 0.00(0) to 0.97(5).

The 100 MHz digit is a constant 1. The 10 and 1 MHz digits are controlled by the larger knob with each detent altering the display by 1 MHz. The digits range from 118 through 151 MHz.

D. Active / Preselect Frequency Interchange Switch:

A push button switch located on the left of the display window provides a means of exchanging active and preselected frequencies.

E. MEMORY 1 / 2 Switches:

Push button MEMORY switches provide additional preselected frequency storage. Activation of these switches will illuminate / extinguish the MEMORY 1 / 2 indicators. When depressed from an extinguished state, the previously stored frequency will be transferred to the preselect display. This frequency is now available for ready interchange with the active display. Depressing the switch from an illuminated state causes storing of the current preselect frequency and extinguishing of the indicator. The indicators (white) are located in the push button switches and indicate which memory location is active and displayed on the lower display.

With one MEMORY switch active (illuminated), depressing the other MEMORY switch or the frequency interchange switch will extinguish the currently illuminated indicator and store the displayed frequency. This frequency is stored in the memory location that corresponds to the indicator just extinguished.

F. 121.5 Switch:

A two position recessed toggle switch provides direct access to the VHF guard frequency of 121.50 MHz when placed in the 121.50 position. Provided a COMM control head failure has not occurred, the active display will indicate a frequency of 121.50 MHz. This access discrete is also used by the VHF transceiver to cause direct radio channeling of 121.50 in the event of a COMM control head failure.

When the 121.5 switch is placed back in the TUNE position, the COMM control head will again transmit the previous active frequency displayed in the upper display before 121.5 was selected.

For aircraft with 8.33 kHz channel spacing capability, the TUNE / 121.50 switch is relabeled and reprogrammed to enable either 25 kHz spacing or 8.33 kHz spacing tuning capability. To switch between either kHz spacing, the flight crew should:

(1) Record the current active frequency.

(2) Turn the radio OFF.

(3) Reposition the kHz spacing switch to the desired spacing, e.g., from 25 kHz to 8.33 kHz, or vice versa.

(4) Turn the radio ON.

With 8.33 kHz spacing enabled for tuning and an 8.33 kHz frequency tuned, the actual frequency is not displayed. Rather, a “channel” or “code” is displayed that coincides with the frequency change as received from ATC. Additionally, the 100 MHz digit (the “constant 1”) is not displayed when 8.33 kHz spacing is enabled. An example of actual frequency versus displayed frequency is shown on the table that follows.

Because of the nature of the bandwidth differences between 25 kHz spacing and 8.33 kHz spacing, ATC may assign a frequency channel for a 25 kHz frequency when 8.33 kHz is selected for tuning. The result is that the displayed frequency may differ slightly from the actual frequency, e.g., the displayed frequencies of either 132.000 and 132.005 result in the same actual frequency of 132.0000. See the following table:

8.33 kHz SPACING: GABLES DIGITAL FREQUENCY CONTROL UNIT		
Actual Frequency	**Tuning Spacing**	**Displayed Frequency**
132.0000	**25 or 8.33**	**32.000 or 32.005**
132.0083	8.33	32.010
132.0166	8.33	32.015
132.0250	**25 or 8.33**	**32.025 or 32.030**
132.0333	8.33	32.035
132.0416	8.33	32.040
132.0500	**25 or 8.33**	**32.050 or 32.055**
132.0583	8.33	32.060
132.0666	8.33	32.065
132.0750	**25 or 8.33**	**32.075 or 32.080**

G. TX Indicator:

Green in color, the TX indicator provides an indication that the COMM Push-To-Talk (PTT) line is active.

H. TEST Switch:

The TEST switch has no effect on any other output conditions. When depressed, the following actions occur:

- Functional test status on the control head is transmitted
- Test indicator located in the TEST switch is illuminated
- All segments and decimal points of the active and preselect displays are illuminated

I. Circuit Breakers (CBs):

The VHF communications system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
VHF COMM CONT #1	CP	H-10	FWD EMER BATT Bus
VHF COMM CONT #2	CP	I-10	R Main DC Bus
VHF COMM #1	CP	H-9	FWD EMER BATT Bus
VHF COMM #2	CP	I-9	R Main DC Bus

J. Advisory (Blue) CAS Messages:

CAS Message:	Cause or Meaning:
VHF COMM 1-2-3 FAIL (1)	Indicated VHF communications radio has failed.

(1) A third system may be installed as an outfitting option.

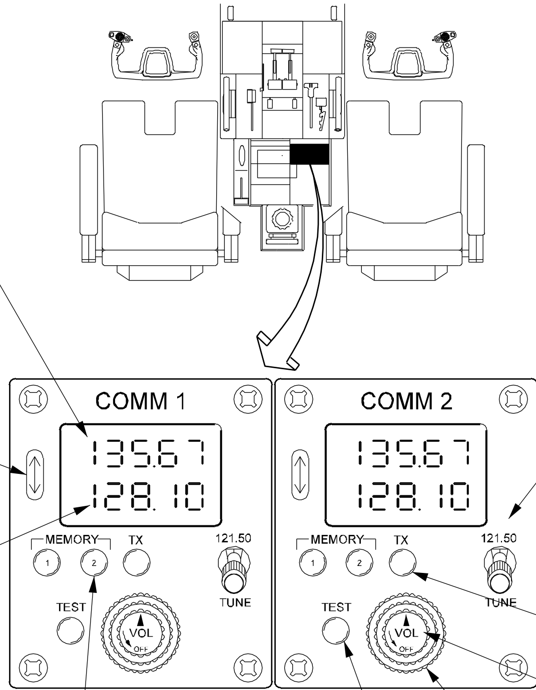
4. Limitations:

A. Flight Manual Limitations:

There are no limitations for the VHF communications system at the time of this revision.

B. Pilot's Manuals:

Consult the Collins RTU-4200 Series Pilot's Guide, Collins Publication Number 523-0777-900, Revision 2, dated May 1, 1996 (including Addendum 2, dated June 16, 1998 or later approved revision) for general operating procedures of the VHF-422 transceivers.



Active Display

Interchange Swtich
Exchanges active and preselected frequencies.

Preselect Display
Interchangeable with active display.

121.5 / TUNE
- 121.5:
  - • Provides direct access to VHF guard frequency of 121.50 MHz.
  - • If COMM control head failure has not occurred, active display will indicate 121.50 MHz.
  - • If COMM control head failure has occurred, also causes direct radio channeling of 121.50 MHz.
- TUNE:
  - • COMM control head transmits active frequency displayed in active display.
- For aircraft with 8.33 kHz channel spacing capability:
  - • Switch is relabeled and reprogrammed to enable either 25 kHz spacing or 8.33 kHz spacing tuning capability.
  - • With 8.33 kHz spacing enabled and 8.33 kHz frequency tuned, actual frequency is not displayed. A "channel" or "code" is displayed coinciding with frequency change as received from ATC.
  - • The 100 MHz digit is displayed.

TX
(Green) indicates COMM control head transmitting.

ON / OFF / VOL
Applies power to the control head and controls volume.

MEMORY 1/2
- Provides additional preselected frequency storage and indicates which memory location is active and displayed on preselect display.
- Activation illuminates / extinguishes MEMORY 1/2 indicators (white).
- When depressed from extinguished state, previously stored frequency is transferred to preselect display, making it available for interchange with the active display.
- Depressing switch from illuminated state causes storing of current preselect frequency and extinguishing of indicator.
- With one MEMORY switch active (illuminated), depressing other MEMORY switch or frequency interchange switch extinguishes currently illuminated indicator and stores displayed frequency in memory location just extinguished.

Frequency Selection Knobs
- Dual concentric control knobs.
- Provides control of preselect display.
- Clockwise rotation increases frequency; counterclockwise rotation decreases frequency.
- Tunable range is from 118.00(0) to 151.97(5) MHz in 25 kHz steps.
- The .001 MHz digit is not displayed.
- The .1 and .01 MHz digits are controlled by smaller frequency knob. Each detent of knob will vary display in .02(5) MHz increments. Digits range from .00(0) to .97(5).
- The 100 MHz digit is a constant 1. The 10 and 1 MHz digits are controlled by larger knob with each detent altering display by 1 MHz. Digits range from 118 through 151 MHz.

TEST
When depressed, the following occurs:
- Functional test status on control head is transmitted.
- TEST switch is illuminated.
- All segments and decimal points of active and preselect displays are illuminated.

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Figure 2. Gables Digital Frequency Control Unit

2A-23-30: HF Communications System

1. **Type of System Installed:**

The type of High Frequency (HF) communications systems installed on GIV aircraft is dependent upon aircraft production number. The two types of systems are:

- Collins HF-190 Series (Aircraft SN 1000 through 1167, excluding 1034)
- Collins HF-9000 Series (Aircraft SN 1034, 1168 and subsequent)

2. **General Description:**

A. **Collins HF-190 System:**

The Collins HF-190 series communications system is a dualized full frequency system designed to provide the flight crew with very long range airborne voice and data communication. Operating in the 2.0000 to 29.9999 MHz frequency range with channel spacing of 100 Hz, it provides 280,000 discrete channels.

HF modes of operation are:

- EM LO: Emergency frequency selection - low
- EM HI: Emergency frequency selection - high
- NORM: Normal frequency selection
- MAR: Maritime channel selection
- PRE: User programmed channel selection
- PGM: Programs user selected channels

HF emission modes include:

- UV: Upper Side Band - Voice
- LV: Lower Side Band - Voice
- UD: Upper Side Band - Data
- LD: Lower Side Band - Data
- CW: Continuous Wave
- AM: Amplitude Modulation

NOTE:

The use of lower side band is legal for some international and offshore communications (most commonly in Australia and the Far East), but is not authorized in the United States and most European countries.

The HF-190 series communications system can operate to 40,000 feet altitude with continuous key, 40,000 to 55,000 feet altitude with one minute key / one minute unkey, and 55,000 to 70,000 feet altitude with half minute key / three and one-half minute unkey, at temperatures from -54°C to +71°C.

The system contains an extensive self-test function to detect faults to the lowest Line Replaceable Unit (LRU). It displays the faults on the FAULT readout in code form and by illuminating appropriate fault light (CP / RT / CT). During self-test, all segments of the display and CP / RT / CT fault

lights will be illuminated. Any system fault detected will be displayed in the FAULT readout and in most cases, the transmit mode will be inhibited.

B. Collins HF-9000 System:

The Collins HF-9000 series communications system is a dualized full frequency system designed to provide the flight crew with very long range airborne voice and data communication. It consists of two single systems sharing a common HF antenna.

The HF-9000 system employs the use of fiber-optic cables to serially transmit all control and status information between units. Transmit and receive RF signals are supported by the RF coaxial cable. The system operates on the frequency band of 2.0000 through 29.9999 MHz, which allows 280,000 channels spaced in 100 Hz increments.

HF operating modes (either simplex or half duplex) are:

- MAN: Manual discrete frequency mode
- CHN: User-programmed preset channel mode
- SCN: User-programmed preset channel receive scan mode
- MAR: Maritime preprogrammed preset channel mode
- TST: Built-In Test (BIT) mode
- PGM: User-programmable preset channel program mode
- EMR: Emergency preprogrammed preset channel mode
- EXT: External system control mode
- SBY: Standby control mode

Emission modes include:

- UV: Upper Side Band - Voice
- LV: Lower Side Band - Voice
- UD: Upper Side Band - Data
- LD: Lower Side Band - Data
- CW: Continuous Wave
- AM: Amplitude Modulation
- FM: Frequency Modulation (Optional)

NOTE:

The use of lower side band is legal for some international and offshore communications (most commonly in Australia and the Far East), but is not authorized in the United States and most European countries.

Transmitter power output is selectable to three levels:

- Low - 10 Watts ±1db Peak Envelope Power (PEP) average
- Medium - 50 Watts ±1db PEP average
- High - 175 Watts ±1db PEP 50-watt average

The system has preset capabilities that include 99 user-programmable frequencies, 176 pre-programmed International Telecommunications Union (ITU) maritime frequencies and two preprogrammed emergency

international distress frequencies. Receiving is continuous. For transmitting, the system is capable of continuous voice operation at temperatures -55°C to +71°C.

The HF-9000 system employs an extensive self-test function to determine which unit has a fault and to display it on the control unit by illuminating the following two-digit codes:

- RT - Receiver / Transmitter
- CU - Antenna Coupler
- R - Remote Control Unit
- FO - Fiber-optic Link
- _ _ (Two Blanks) - External System Malfunction

3. Description of Subsystems, Units and Components:

A. Components Common To Both Series:

(1) Receiver / Transmitters:

Two receiver / transmitters are installed in the tail compartment. Each can operate independent of one another. Since there is only one HF antenna, both systems can receive information simultaneously. With one of the systems in transmit mode, however, the other system is disconnected from the antenna and is unable to receive or transmit.

(2) Antenna Couplers:

Two antenna couplers are installed in the tail compartment. They are pressurized tuning assemblies with a sealed control compartment. Pressurization is done to the tuner assembly to prevent possible high voltage arcing at high altitudes, to provide a predictable cooling medium and to prevent corrosive elements from entering the unit. Output from the couplers is provided to a common antenna adapter.

(3) HF Antenna:

A single HF antenna is incorporated into the leading edge structure of the vertical stabilizer. It is used by both HF systems.

4. Controls and Indications:

A. HF-190 Series Control Panel:

(See Figure 3 and Figure 4.)

(1) General:

The HF-190 series control panel is used to control and display the modes, channels, frequency selected and faults. It contains all function switches necessary to operate the HF system.

(2) ON / OFF Switch:

This switch selects the HF communications system ON or OFF. Upon selection to OFF, data in the display will be stored in non-volatile memory and will be restored to the display when the system is again selected ON.

(3) SQ (Squelch) Knob:

This knob is adjusted to mute undesired background noise when

voice communication is not present.

(4) DSBL (Squelch Disable) Pushbutton:

Depressing the DSBL pushbutton disables the squelch circuit, enabling the user to determine if there is traffic on the selected frequency without adjusting the SQL control.

(5) VOL (Volume) Knob:

The volume knob adjusts the audio level as desired by the operator. Clockwise rotation increases the volume level, while counterclockwise rotation decreases the volume level.

(6) LOAD Pushbutton:

This pushbutton stores the desired preset frequency, channel and emission mode in the non-volatile memory of the receiver / transmitter, ready for instant recall in PRE (preset) operation mode.

(7) Mode Select Knob:

The mode select knob is used to select the following modes:

(a) EM LO (Emergency Frequency Selection - Low): Manually selectable international distress channel that operates on 2.1820 MHz.

(b) EM HI (Emergency Frequency Selection - High): Manually selectable international distress channel that operates on 8.3640 MHz.

(c) NORM (Normal Frequency Selection): Manually selectable mode allowing direct tune of any one of 280,000 frequencies and any of the 6 emission modes (UV, LV, UD, LD, CW and AM).

(d) MAR (Maritime Channel Selection): Manually selectable mode allowing access to the 176 ITU Public Correspondence Channels (and their receive / transmit frequencies) in the maritime radio telephone network. All channels operate on half duplex in the upper side band voice (UV) mode only. All 176 ITU channels are permanently programmed in the non-volatile memory. The last 10 channels used are also stored in non-volatile memory to allow quick retrieval.

(e) PRE (User Programmed Channel Selection): Manually selectable mode allowing access to one of 20 user programmed channels, rather than manually selecting the frequencies and emission mode.

(f) PGM (Program User Selected Channels): Manually selectable mode used to program user channels.

(8) TEST Pushbutton:

The TEST pushbutton will initiate the self-test diagnostic routine. All LCD displays and the three fault indicators will illuminate. If the “receive” self-test detects no faults, the “transmit” self-test may be initiated by momentarily keying the system. A tuning tone will be heard during the “transmit” self-test. No frequency or modes will be displayed unless the system is in MAR mode, in which case the

selected maritime channel transmit frequency will be displayed while the system is keyed.

NOTE:

After the test is initiated, the system must be keyed, channel changed or selected OFF to exit the test.

(9) Emission Mode Switch / MODE Display:

Positioning this switch up or down when in NORM or PGM mode will cycle the emission MODE display and the system between UV, LV, LD, CW or AM emission mode.

(10) Channel Select Switches / CHAN Display:

Positioning these three switches up or down when in MAR, PRE or PGM mode allows selection and display of all 176 ITU channels when in MAR mode, and user-programmed channels when in PRE and PGM modes. The selected channel will appear in the CHAN display. The associated frequency and emission mode will appear in their respective displays. The function of each of the three switches is further explained as follows:

(a) The left channel select switch selects the maritime ITU most significant digit (4 through 22).

(b) The center channel select switch selects the most significant digit for user preset channel 10 through 19 and the 10s digit (0 through 9) for maritime ITU channels.

(c) The right channel select switch selects the user preset channels 0 through 9 as well as the least significant digit of PRE set channels 10 through 19 and the MAR 176 ITU channels.

(11) Frequency Select Switches / FREQ Display:

These 5 switches allow selection and display of discrete frequencies from 2.0000 through 29.9999 MHz in 100 Hz steps in the NORM mode and to select user preset frequencies / channels in the PGM mode for loading and for later use in PRE mode. All switches are momentary in both directions to select digits up or down, in any order. The function of each of the 5 switches is further explained as follows:

(a) The left most switch selects the MHz digit, 2 through 29.

(b) The second from left switch selects the 100 kHz digit, 0 through 9.

(c) The center switch selects the 10 kHz digit, 0 through 9.

(d) The second from right switch selects the 1 kHz digit, 0 through 9.

(e) The right most switch selects the 100 Hz digit, 0 through 9.

(12) FAULT Display / Indicators:

Any system fault detected during power on, normal operation or self-test is indicated by a fault code displayed in the FAULT display and, if appropriate, an illuminated CP / RT / CT annunciator.

B. **HF-9000 Series Control Panel:**

(See Figure 5 and Figure 6.)

(1) General:

The HF-9000 series control panel is used to control and display the modes, channels, frequency selected and faults. It contains all function switches necessary to operate the HF system.

(2) VOL (Volume) Knob:

The volume knob adjusts the audio level as desired by the operator. Clockwise rotation increases the volume level, while counterclockwise rotation decreases the volume level. Whenever the volume control is moved, the volume level is momentarily displayed in the FREQ / CHAN display.

(3) SQL (Squelch) Knob:

The squelch control is adjusted to mute undesired background noise when voice communication is not present. The proper squelch setting is obtained by rotating the SQL knob counterclockwise to disable squelch (SQL 0) and then clockwise one click (SQL 1). The receiver will mute after a short delay. If intermittent noise persists, the SQL knob can be further rotated clockwise one click (SQL 2) or two clicks (SQL 3).

The SQL control has no effect when in TST, PGM, EXT or SBY mode. The squelch level is momentarily displayed in FREQ / CHAN function after each change of SQL control.

(4) DSBL (Disable) Pushbutton:

Depressing the DSBL pushbutton disables the squelch circuit, enabling the user to determine if there is traffic on the selected frequency without adjusting the SQL control.

(5) CHAN (Channel) Knob:

The CHAN knob provides selection of all maritime and emergency preprogrammed preset channels and user-programmed preset channels. Using the CHAN knob, the channel will increase or decrease by one (i.e., 2236, 2237 or 2236, 2235, etc.) regardless of where the cursor is positioned. To increase or decrease channels by more than one (i.e., 2236, 2246, 2336, 3636, etc.), the CURSOR knob is used to position the cursor under the desired channel digit and the VALUE knob is used to set the digit.

(6) FREQ / LD (Frequency / Load) Pushbutton:

When in PGM mode, depressing this switch loads the desired receive-transmit emission mode, frequency and channel data into the receiver / transmitter's non-volatile memory for the 99 user-programmable preset channels for simplex operation. For half-duplex operation, the FREQ / LD switch and microphone key must be depressed to load transmit data. By depressing the switch, the channel and frequency display recycles in CHN mode; when operating in EMR or MAR mode, the emergency (receive and transmit) or maritime (receive) frequency will be displayed. To view the maritime transmit frequency, depress the FREQ / LD switch while keying the microphone. When in the TST mode and the

HF-9000 system has failed, depressing this switch will sequence through the fault diagnostic codes.

(7) CURSOR Knob:

The CURSOR knob moves the cursor left or right for selecting the display section to be changed.

(8) VALUE Knob:

The VALUE knob increases or decreases the value in the display section (OPR, MODE, FREQ / CHAN or PWR) selected by the cursor.

(9) Cursor Display:

The cursor, a segmented line display, is positioned (by using the CURSOR knob) under the display section (OPR, MODE, FREQ / CHAN and PWR) to be changed. After selection of the desired display section, the value displayed may be increased or decreased from its current value by turning the appropriate control: CHAN or VALUE knobs for CHAN function and VALUE knob for all other functions.

(10) OPR (Operating Mode) Display Section:

Three alpha-numeric characters display the HF-9000 system operating mode selected by VALUE knob. The operating modes are:

- MAN: Manual discrete frequency mode
- CHN: User-programmed preset channel mode
- SCN: User-programmed preset channel receive scan mode
- MAR: Maritime preprogrammed preset channel mode
- TST-BIT: Built-in test mode
- PGM: User-programmable preset channel program mode
- EMR: Emergency preprogrammed preset channel mode
- EXT: External system control mode
- SBY: Standby control mode

The OPR section will display FLT (fault) or WRN (warning) if a fault or warning occurs in the HF-9000 system. The OPR function will momentarily display VOL or SQL whenever the VOL (volume) or SQL (squelch) control setting has been changed. The VOL or SQL level will also be displayed at the same time in FREQ / CHAN section.

(11) MODE (Emission Mode) Display Section:

Two alpha-numeric characters display the RF emission mode selected by the VALUE knob. The following modes are available:

- UV: Upper Side Band - Voice
- LV: Lower Side Band - Voice
- UD: Upper Side Band - Data
- LD: Lower Side Band - Data
- CW: Continuous Wave
- AM: Amplitude Modulation

- FM: Frequency Modulation (Optional)

If a fault or warning occurs in the system, the MODE function characters will indicate the unit in which the fault or warning occurred: CU for antenna coupler, RT for receiver / transmitter, R for remote control unit, FO for fiber-optic system and _ _ (two blanks) for an external system malfunction.

(12) FREQ / CHAN (Frequency / Channel) Display Section:

Up to six numeric characters display frequency data and channel number for normal operation. The frequency is either increased or decreased by using the VALUE knob and is displayed in all six digits plus a decimal point. Channel selection is accomplished with the CHAN or VALUE knob. During the TST (built-in test) mode, the unit, module and circuit card failure are displayed.

(13) PWR (Power) Display Section:

This is a three-level bar indicator for selectable output power levels of low power (bottom bar), medium power (bottom two bars) and high power (all three bars). The output power level is selected by the VALUE control.

C. Circuit Breakers (CBs):

The HF-190 communications system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
HF COMM #1	CP	C-10	L Main AC Bus, ϕA
HF COMM #1	CP	C-11	L Main AC Bus, ϕB
HF COMM #1	CP	C-12	L Main AC Bus, ϕC
HF COMM #2	CP	E-10	R Main AC Bus, ϕA
HF COMM #2	CP	E-11	R Main AC Bus, ϕB
HF COMM #2	CP	E-12	R Main AC Bus, ϕC

The HF-9000 communications system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
HF RT CPLR #1	CP	D-11	L Main DC Bus
HF CONT #1	CP	D-12	L Main DC Bus
HF RT CPLR #2	CP	E-11	R Main DC Bus
HF CONT #2	CP	E-12	R Main DC Bus

5. Limitations:

A. Flight Manual Limitations:

There are no limitations for the HF communications system at the time of this revision.

B. Pilot's Manuals:

(1) HF-190 Series:

Consult the Collins High Frequency Communications System Pilot's Guide, Collins Publication Number 523-0774-209, for operating procedures of the HF-190 series system.

(2) HF-9000 Series:

Consult the Collins High Frequency Communications System Pilot's Guide, Collins Publication Number 523-0774-344, Revision 6, dated February 1, 1994 (or later approved revision) for operating procedures of the HF-9000 series system.

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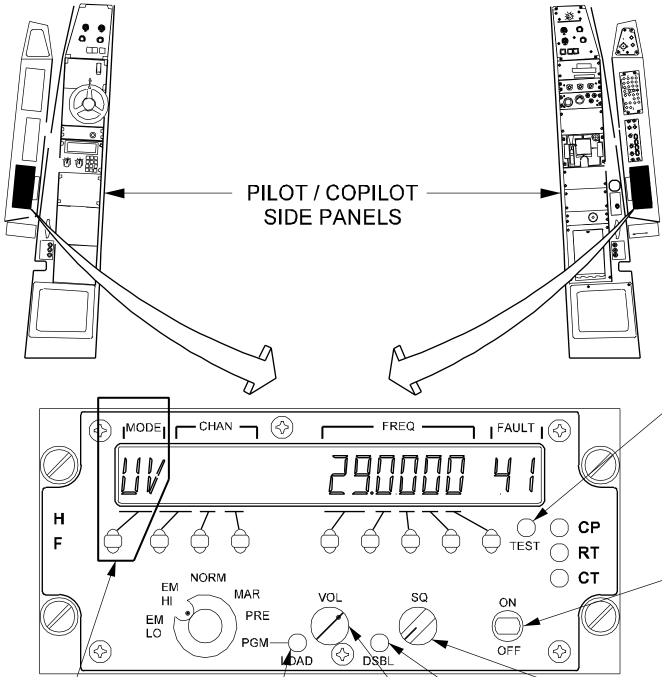

TEST
Initiates self test diagnostic routine.
- All LCD displays and the three fault indicators illuminate.
- If "receive" self test detects no faults, "transmit" self test may be initiated by momentarily keying system. A tuning tone will be heard during self test.
- No frequency or modes will be displayed unless system is in MAR mode, in which case selected maritime channel transmit frequency will be displayed while system is keyed.

NOTE: After test is initiated, system must be keyed, channel changed or selected OFF to exit test.

ON / OFF
- Selects system ON or OFF.
- Upon selection to OFF, data in display is stored in non-volatile memory and will be restored to display when system is again selected ON.

SQ
(Squelch) Mutes undesired background noise when voice communication not present.

LOAD
Stores desired preset frequency, channel and emission mode in non-volatile memory of receiver / transmitter, ready for instant recall in PRE (preset) operation mode.

DSBL
(Squelch Disable) disables squelch circuit without adjusting SQL.

Emission Mode Switch / MODE display
Cycles emission MODE display and system between UV, LV, UD, LD, CW or AM emission modes.

VOL
(Volume) Adjusts the audio volume level as desired by the operator.
- Clockwise rotation increases volume level.
- Counterclockwise rotation decreases volume level.

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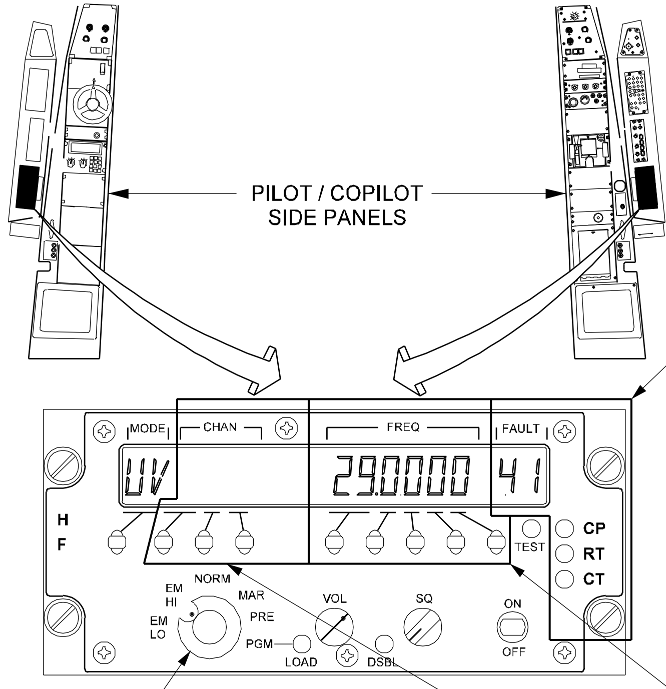
Figure 3. HF-190 Series Control Panel (Sheet 1 of 2)



Fault Display / Indicators
Any system fault detected during power on, normal operation or self-test is indicated by a fault code displayed in FAULT display and, if appropriate, an illuminated CP / RT / CT annunciator.

Frequency Select Switches / FREQ Display
Allows selection and display of discrete frequencies from 2.0000 through 29.9999 MHz in 100 Hz steps in NORM mode. Also used to select user preset frequencies / channels in the PGM mode for loading and for later use in PRE mode.

- Left most switch selects the MHz digit, 2 through 29.
- Second from left switch selects the 100 Hz digit, 0 through 9.
- Center switch selects the 10 kHz digit, 0 through 9.
- Second from right switch selects the 1 kHz digit, 0 through 9.
- Right most switch selects the 100 Hz digit, 0 through 9.

Mode Select Knob

- EM LO (Emergency Frequency Selection - Low): Manually selectable international distress channel operating on 2.1820 MHz.
- EM HI (Emergency Frequency Selection - High): Manually selectable international distress channel operating on 8.3640 MHz.
- NORM (Normal Frequency Selection): Manually selectable mode allowing direct tune of any one of 280,000 frequencies and any of the 6 emission modes (UV, LV, UD, LD, CW and AM).
- MAR (Maritime Channel Selection): Manually selectable mode allowing access to the 176 ITU channels on maritime radio telephone network. All channels operate on half duplex in UV mode only, and are permanently programmed in non-volatile memory. The last 10 channels used are also stored in non-volatile memory to allow quick retrieval.
- PRE (User Programmed Channel Selection): Manually selected mode allowing access to one of 20 user programmed channels rather than manually selecting the frequencies and emission mode.
- PGM (User Programmed Channel Selection): Manually selectable mode used to program user channels.

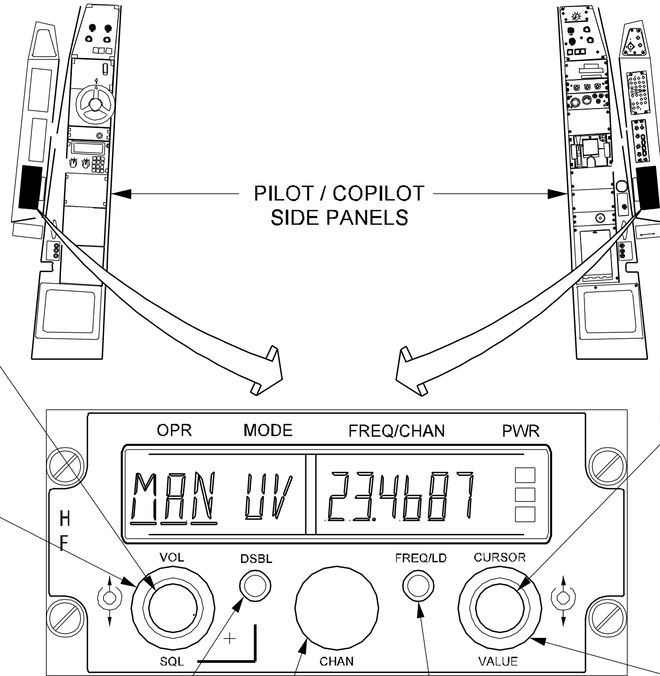
Channel Select Switches / CHAN Display
Selection up or down when in MAR, PRE or PGM mode allows selection and display of all 176 ITU channels when in MAR mode, and user programmed channels when in PRE and PGM modes. Selected channel will appear in CHAN display. Associated frequency and emission mode will appear in respective displays.

- Left channel select switch selects maritime ITU most significant digit (4 through 22).
- Center channel select switch selects most significant digit for user preset channel 10 through 19 and 10s digit (0 through 9) for maritime ITU channels.
- Right channel select switch selects user preset channels 0 through 9 as well as least significant digit of PRE set channels 10 through 19 and MAR 176 ITU channels.

26019C00

Figure 4. HF-190 Series Control Panel (Sheet 2 of 2)

VOL
(Volume) Adjusts the audio volume level as desired by the operator.
- Clockwise rotation increases volume level.
- Counterclockwise rotation decreases volume level.
- Level momentarily displayed in FREQ / CHAN display.



CURSOR
Moves cursor left or right for selecting display section to be changed.

SQL
(Squelch) Mutes undesired background noise when voice communication not present.
- Level is momentarily displayed in FREQ / CHAN function after each change.
- Has no effect in TST, PGM, EXT or SBY mode.

VALUE
Increases or decreases value in display section (OPR, MODE, FREQ / CHAN or PWR) selected by cursor.

DSBL
(Squelch Disable) Disables squelch circuit without adjusting SQL.

CHAN
(Channel)
- Provides selection of all maritime and emergency preprogrammed preset channels and user programmed preset channels.
- When using CHAN knob, channel will increase or decrease by one (i.e. 2236, 2237 or 2236, 2235, etc.) regardless of where cursor is positioned.
- To increase or decrease channels by more than one (i.e. 2236, 2246, 2336, 3636, etc.) CURSOR knob is used to position cursor under desired channel digit and VALUE knob is used to set digit.

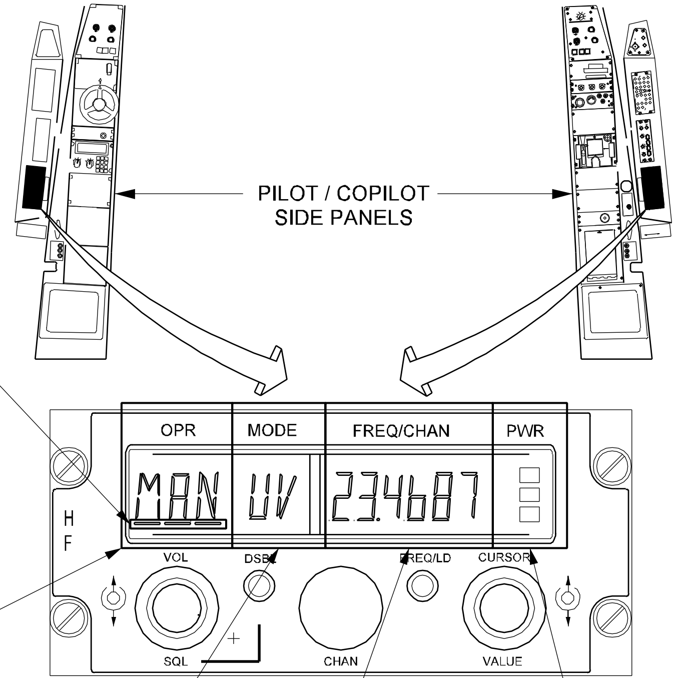
FREQ / LD
(Frequency / Load)
- In PGM mode, depressing switch loads desired receive transmit emission mode, frequency and channel data into receiver transmitter's non-volatile memory as one of 99 user programmable preset channels for simplex operation.
- For half duplex operation, FREQ / LD switch and microphone key must be depressed to load transmit data. By depressing switch, channel and frequency display recycles in CHN mode; when operating in EMR or MAR mode, emergency (receive and transmit) or maritime (receive) frequency is displayed. To view maritime transmit frequency, depress FREQ / LD switch while keying microphone.
- When in TST mode and HF-9000 system has failed, depressing switch will sequence through fault diagnostic codes.

26020C00

Figure 5. HF-9000 Series Control Panel (Sheet 1 of 2)

Title Page | Prev Page | Next Page | TOC

Cursor Display
A segmented line display positioned (by using CURSOR knob) under display section (OPR, MODE, FREQ / CHAN and PWR) to be changed. After selection of desired display section, value displayed may be increased or decreased from current value by turning appropriate control: CHAN or VALUE knobs for CHAN function and VALUE knob for all other functions.



OPR
(Operating Mode)

- Three alpha-numeric characters display operating mode selected by VALUE knob. Operating modes are:
  - MAN: Manual discrete frequency mode
  - CHN: User programmed preset channel mode
  - SCN: User programmed preset channel receive scan mode
  - MAR: Maritime preprogrammed preset channel mode
  - TST-BIT: Built in test mode
  - PGM: User programmable preset channel program mode
  - EMR: Emergency preprogrammed preset channel mode
  - EXT: External system control mode
  - SBY: Standby control mode
- Will display FLT (fault) or WRN (warning) if a fault or warning occurs in the HF-9000 system.
- Will momentarily display VOL or SQL whenever VOL or SQL setting has been changed. VOL or SQL level is also displayed at the same time in FREQ / CHAN section.

PWR
(Power) Three bar indicator displaying output power as selected by VALUE knob.

- Low power - bottom bar only
- Medium power - bottom two bars
- High power - all three bars

MODE
(Emission Mode)

- Two alpha-numeric characters display RF emission mode selected by VALUE knob. Modes available are:
  - UV: Upper Side Band - Voice
  - LV: Lower Side Band - Voice
  - UD: Upper Side Band - Data
  - LD: Lower Side Band - Data
  - CW: Continuous Wave
  - AM: Amplitude Modulation
  - FM: Frequency Modulation (Optional)
- If fault or warning occurs in system, MODE function characters indicate unit in which fault or warning occured:
  - CU for antenna coupler
  - RT for receiver / transmitter
  - R for remote control unit
  - FO for fiber optic system
  - _ _ (two blanks) for external system malfunction.

FREQ / CHAN
(Frequency / Channel)

- Up to six numeric characters display frequency data and channel number for normal operation.
- Frequency increased or decreased by using VALUE knob and is displayed in all six digits plus a decimal point.
- Channel selection accomplished with CHAN or VALUE knob.
- During TST mode, unit, module and circuit card failure are displayed.

26021C00

Figure 6. HF-9000 Series Control Panel (Sheet 2 of 2)

2A-23-40: Integrated Automatic Tuning System

THIS SECTION APPLIES TO AIRCRAFT SN 1310 AND SUBSEQUENT

1. General Description:

In Aircraft 1310 and subsequent, integrated control and display of aircraft communications and navigation systems is provided by Collins RTU-4200 series Radio Tuning Units (RTUs). The integrated control includes the setting of radio frequencies / channels and modes as well as optional control of radio volume. The RTUs provide single point control of both on-side and cross-side systems from either the pilot's or copilot's positions.

RTU-4200 standard features include the following:

- COM (Communications) Control Function
- NAV (Navigation) Control Function
- DME (Distance Measuring Equipment) Control Function
- ATC (Air Traffic Control Transponder) Control Function
- ADF (Automatic Direction Finder) Control Function

On the RTU-4210, TCAS (Traffic Collision Avoidance System) provisions are included. The RTU-4220 includes all features of the RTU-4200 and RTU-4210, plus the capability to display an HSI (Horizontal Situation Indicator).

The RTU is designed to be installed in pairs, with each RTU normally controlling the on-side systems. The RTU is capable, however, of controlling cross-side systems by selection of cross-side tuning or reversionary tuning is selected. Cross-side tuning obviates the need for reaching across the pedestal when both RTUs are functioning normally. Reversionary tuning allows control of both on-side and cross-side systems in the event that one RTU should fail.

In addition to on-side and cross-side RTU tuning, systems may be remotely tuned through the RTU. Manual (keyboard) and FMS tuning commands may tune both the on-side and cross-side RTUs. Remote tuning commands from a CTL (remote controller) may also tune the pilot's RTU COM and NAV functions.

All RTUs have the ability to display RTU / system / radio diagnostic data. Additionally, some RTUs provide tuning capability of a third VHF communications transceiver and one or two High Frequency (HF) communication transceivers.

2. RTU Displays:

The RTU display structure is made up of three tiers: top level displays, main display pages and preset pages. Two error pages also exist that can be shown from any level: a cross-side radio tuning inoperative page and a configuration error page. Depending on configuration, an HSI display may be accessible from the top level displays.

A. Top Level Displays:

The top level display consists of the top level page and, if installed, a second top level page. The top level page is shown at power-up and when no RTU control has been selected for a period of time. If installed, the second page can be accessed by pressing the NEXT PAGE line select key. The top level displays show subdisplays for each of the systems controlled by the RTU and provide access to the second tier of RTU display structure, the main display pages.

Subdisplays are normally shown on the RTU with which the system is paired (i.e., COM 1 subdisplay is shown on pilot's RTU, NAV 2 is shown on

copilot's RTU, etc.). Subdisplays for single systems (TCAS, etc.) are shown on both the on-side and cross-side RTUs.

The top level display normally shows subdisplays for the COM, NAV, ATC and TCAS systems. If TCAS is not incorporated, the ADF system subdisplay is shown. The second top level page, if present, shows subdisplays for items not shown on the top level page and may include ADF, HF, HSI and a third COM.

The information contained on top level subdisplays includes radio frequencies, presets, codes and mode annunciators.

Labels (i.e., COM1, ATC1, TCAS, etc.) are shown at top of each subdisplay. These labels identify the systems and indicate the RTU with which they are associated. Labels for single systems do not show a side number.

B. Main Display Pages:

Main display pages are accessed from the associated system subdisplay on the top level display. Information shown for an individual system on the top level display is also shown on that system's main display page, along with additional controls. Individual main display pages also provide access to preset pages. To return to the top level page from a main display page, the RETURN line select key is depressed.

C. HSI Page:

If incorporated, the HSI page is accessed from either the top level page or the second top level page. The HSI page shows the familiar HSI display with compass card, course pointer, TO-FROM pointer, lateral and vertical deviation display, DME distance, and marker beacon annunciators. In addition, the HSI page provides control for the COM, NAV and ATC systems. To return to the top level page from the HSI page, the RETURN line select key is depressed.

D. Preset Pages:

Preset pages for a particular system are accessed by pushing the PRESET PAGE line select key on the desired system's main display page. Active frequencies, presets and mode annunciators are shown on the preset pages, along with controls for preset programming and tune mode selection. Unless otherwise specified, four of the maximum presets available are shown on each preset page. Twenty presets are normally available for each of the COM, NAV, ADF and HF systems. To return to the main display page from a preset page, the RETURN line select key is depressed.

3. Controls and Indications:

(See Figure 7.)

A. ON / OFF Switch:

An ON / OFF switch may or may not be present, depending on the installation. If present, the ON / OFF switch is incorporated into the brightness (BRT) knob. The RTU is selected ON by rotation of the BRT knob clockwise out of the OFF detent, and OFF by rotation of the BRT knob fully counterclockwise into the OFF detent.

B. Brightness (BRT) Knob:

In installations where RTU brightness is controlled by the aircraft's master brightness control, the RTU BRT knob is used in conjunction with the aircraft's master brightness control to achieve the desired display brightness. In installations where RTU brightness is not controlled by the aircraft's master brightness control, the RTU BRT knob is solely adjusted to achieve the desired display brightness.

NOTE:

The RTU uses a fluorescent bulb for display backlighting. The performance of fluorescent bulbs and LCDs at extremely low temperatures can be degraded. Within the temperature range of -20°C to +70°C, operation will be normal after a 4 second power-up delay. Below -20°C, there will be a power-up delay of approximately 1 minute for each degree below -20°C, up to 10 minutes. After the display is ON, full display brightness may not be available for an additional 10 minutes.

C. Line Select Keys:

The RTU has seven panel-mounted line select keys surrounding the display window. These keys are used to select control individual radio frequencies, presets, codes and modes. The tune window is shown around the value selected for control.

D. Tuning Window:

The tuning window surrounds ("boxes") the frequency, preset or code selected for control. The tuning knobs are then used to change values shown inside the tuning window. The default position for the tuning window is around the COM recall (top right) frequency on the top level page.

In some installations, the tuning window automatically returns to the default position after twenty seconds of inactivity.

E. Tuning Knobs:

Tuning knobs are used to set the value shown in the tune window. When a frequency, code or mode is shown in the tune window, the large knob controls the most significant digits and the small knob controls the least significant digits.

F. Volume Knob:

A volume knob may or may not be present, depending on the installation. If present, the volume knob is the third (smallest diameter) knob on the tuning knob cluster. Individual radio volume may be set from a top level display, described as follows:

(1) The line select key adjacent to the desired radios subdisplay (on the top level display) is depressed to show the tuning window.

(2) The selected radio's volume is then set with the volume knob. A volume scale will be displayed in the recall display after the first click of the volume knob.

To set a volume level from a radio mode or preset page, the main display page or preset page is selected and the volume knob is turned. A volume

scale will be displayed after the first click of the volume knob.

DME Volume: Normally, DME volume is set to the same level as NAV volume. In installations with independent DME tuning, the DME volume level may be independently controlled from the NAV main display page when the DME Hold (DME-H) function key is depressed. The process is described as follows:

(3) The line select key adjacent to the DME hold frequency is depressed, resulting in the tuning window surrounding the DME frequency.

(4) The DME volume is then set with the volume knob. A volume scale will be displayed after the first click of the volume knob.

G. DME Hold (DME-H) Function Key:

The DME-H function key is used to hold the currently tuned DME frequency. When DME hold is active, the DME hold frequency is shown in green followed by a yellow “H” on the top level page, NAV main display page and preset page.

H. IDENT Function Key:

The IDENT function key is used to transmit an ATC identification pulse. When the IDENT feature is active, “ID” is annunciated in cyan on the ATC subdisplay and on the ATC main display page.

I. 1 / 2 (Cross-side Tuning) Function Key:

To select control of the cross-side system, the 1 / 2 function key on the on-side RTU is depressed. When selected, cross-side labels are shown in yellow. When both RTUs attempt to command the same system simultaneously, the pilot’s RTU takes precedence.

If an RTU had failed, selection of the 1 / 2 function key shows the Cross-side Radio Tuning Inoperative page. This page can be cleared by again depressing the 1 / 2 function key or any line select key. To tune the cross-side systems with a failed RTU (reversionary tuning mode), the failed RTU must first be shut off.

NOTE:

Cross-side tuning removes the associated preset from the active display.

J. 8.33 kHz Channel Spacing:

The ability to select between 25 kHz and 8.33 kHz tuning is found on the first top level page for VHF COMM tuning. This page is accessed by depressing the 1L line select key, adjacent to the active frequency. The top level page then displays the option of selecting 25 kHz or 8.33 kHz tuning using the 1R line select key. The active choice is displayed in large font, cyan in color. Continuing to depress the 1R line select key toggles between the choices. After the choice is made, the RETURN prompt adjacent to the 4L line select key is depressed. Note that when 25 kHz spacing is selected, a “25” icon is present under the preset frequency position at line select key 1R. When 8.33 kHz spacing is selected, there is no icon displayed.

With 8.33 kHz spacing selected for tuning and an 8.33 kHz frequency

tuned, the actual frequency is not displayed. Rather, a “channel” or “code” is displayed that coincides with the frequency change as received from ATC. An example of actual frequency versus displayed frequency is shown on the table that follows.

Because of the nature of the bandwidth differences between 25 kHz spacing and 8.33 kHz spacing, ATC may assign a frequency channel for a 25 kHz frequency when 8.33 kHz is selected for tuning. The result is that the displayed frequency may differ slightly from the actual frequency, e.g., the displayed frequencies of either 132.000 and 132.005 result in the same actual frequency of 132.0000. See the following table:

8.33 kHz SPACING: COLLINS RTU-4200 SERIES RADIO TUNING UNIT		
Actual Frequency	**Tuning Spacing**	**Displayed Frequency**
132.0000	**25 or 8.33**	**132.000 or 132.005**
132.0083	8.33	132.010
132.0166	8.33	132.015
132.0250	**25 or 8.33**	**132.025 or 132.030**
132.0333	8.33	132.035
132.0416	8.33	132.040
132.0500	**25 or 8.33**	**132.050 or 132.055**
132.0583	8.33	132.060
132.0666	8.33	132.065
132.0750	**25 or 8.33**	**132.075 or 132.080**

K. RTU Color Convention:

The RTU uses a color Liquid Crystal Display (LCD) to display data. For the purposes of this discussion, the RTU can be classified into two groups: those with white on-side and single system labels, and those with green on-side and single system labels. Thus, RTU color convention is shown in the following table:

DISPLAY PARAMETER	ON-SIDE / SINGLE SYSTEM LABEL COLOR	
	WHITE	**GREEN**
On-side and Single System Labels	White	Green
Cross-side System Labels	Yellow	
Active Frequency / Channel / Code Data	Green	
Active ATC Code - Standby Selected	White	
Unverified Frequency / Channel / Code Data	Magenta	Cyan
Dashed Unverified Frequency / Channel / Code Data	White	
Recall Freq / Channel / Code / Preset Data	White	Cyan
Active Mode Control Selections	Cyan	
DME Hold (H) Annunciator	Yellow	
ATC Reply (RPLY) Annunciator	Cyan	Green
Transponder Fail (XPDR FAIL) Annunciator	Yellow	
No Remote Tuning (NO RMT TUNE) Annunciator	Cyan	Yellow
CTL standby (STBY) Annunciator	Yellow	

L. **Circuit Breakers (CBs):**

The integrated automatic tuning system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
RTU #1	CP	F-7	EMER DC Bus
RTU #2	CP	F-8	ESS DC Bus

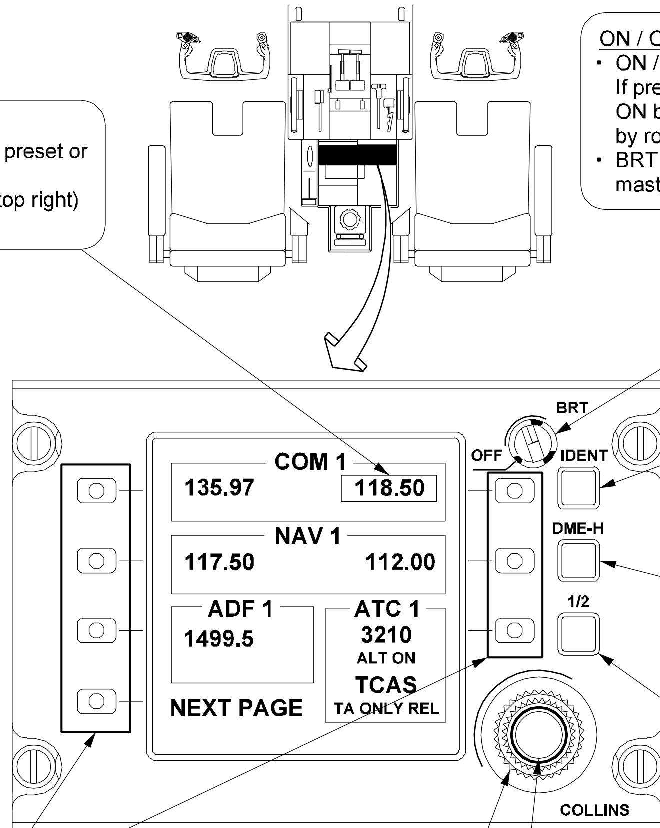
4. Limitations:

A. Flight Manual Limitations:

There are no limitations for the integrated automatic tuning system at the time of this revision.

B. Pilot's Manuals:

Consult the Collins RTU-4200 Series Pilot's Guide, Collins Publication Number 523-0777-900, Revision 2, dated May 1, 1996 (including Addendum 2, dated June 16, 1998) for operating procedures of the RTU-4200 series system.



Tuning Window
- Surrounds ("boxes") frequency, preset or code selected for control.
- Default position is COM recall (top right) frequency on top level page.

ON / OFF / BRT
- ON / OFF: May or may not be present, depending on installation. If present, switch is incorporated into BRT knob. RTU is selected ON by rotation of BRT knob clockwise out of OFF detent, and OFF by rotation or BRT knob fully counterclockwise into OFF detent.
- BRT (Brighthess): Used alone or in conjunction with aircraft's master brightness control to achieve desired display brightness.

IDENT
(ATC Identification)
Used to transmit an ATC identification pulse.

DME-H
(DME Hold)
Used to hold the currently tuned DME frequency.

Line Select Keys
- Provides control of individual radio frequencies, presets, codes and modes.
- Tuning window is shown around the value selected for control.

1/2
(Cross-side Tuning)
- 1/2 function key on the on-side RTU is depressed to select control of the cross-side system.
- If a RTU has failed, selection of 1/2 function key shows Cross-side Radio Tuning Inoperative page. Page can be cleared by again depressing 1/2 function key or any line select key.
- To tune the cross-side systems with a failed RTU (revisionary tuning mode), the failed RTU must first be shut off.

Tuning Knob
Used to set the value shown in tuning window. When a frequency, code or mode is shown in the tuning window, the large knob controls the most significant digits and the small knob controls the least significant digits.

Volume Knob
May or may not be present, depending on installation. If present, the volume knob is the third (smallest diameter) knob on the tuning knob cluster.

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Figure 7. Collins RTU-4200 Series Radio Tuning Unit

2A-23-50: Intercommunications System

1. General Description:

The Intercommunications System (ICS) is an audio integrating system that ties in the communication and navigation receiver audio inputs and outputs to the cockpit speakers, ICS jacks, headsets and microphones.

A typical ICS system consists of:

- Two Baker audio control panels, one located on each upper side console
- Two communications jacks panels (labeled COMM JACKS), one located on each side console
- Two ICS receptacles (labeled ICS JACK), one located in the forward external switch panel, the other located in the tail compartment

2. Description of Subsystems, Units and Components:

A. Baker Audio Control Panel:

A dualized Baker Audio Control System (No. 1 system for the pilot and No. 2 system for the copilot) is installed in the aircraft. The ICS audio control panel and audio amplifier are housed within the same unit, providing an integrated audio system capable of selection and volume control of all installed radio equipment. Selection can be made of all communications, receive and microphone functions including onboard communications (i.e., passenger address, cabin and cockpit interphone).

Two types of pushbutton switches are used. Microphone selection is accomplished by an interlocking switch permitting only one button to lock down at one time. When another button in the microphone selector group is depressed, the previously depressed button is released automatically. The second type of pushbutton switch is known as an alternate action or push-push switch (i.e., the switch is on when depressed and may be released to the off position by depressing again). The push-push switch type of switch is used for all selections of receivers and other functions except the microphone selector.

The audio control panels have a self-contained transistor isolation and speaker amplifier with audio leveling. The system includes use of a filter network to remove the 1020 Hz identifier from the navigation or ADF receiver if desired.

In case of amplifier failure or complete power loss to the system, an emergency operation mode allows receiver selection and direct pass-through of all normal inputs to the headphones and use of the hand microphone for transmission purposes.

B. COMM JACKS Panels:

A communications jack panels (labeled COMM JACKS) is located on each side console. Each panel contains a receptacle for a hand-held microphone (HAND MIKE), a boom microphone (BOOM MIKE) and a set of headphones (HEAD PHONE).

C. Cockpit Microphones:

Hand, boom or mask microphones may be used. The desired output (radio transmitter, paging or cabin telephone) is first selected. The hand microphone, if used, is keyed in the usual way. If the pilot is wearing the oxygen mask, the ICS key on the control wheel must be used to operate

the mask microphone. A boom microphone may be used if desired by connecting to the BOOM MIC jack. Push-To-Talk (PTT) is activated using the ICS MIC switch.

D. ICS Receptacles:

Two ICS receptacles (labeled ICS JACK), one located in the forward external switch panel, the other located in the tail compartment, are provided for intercommunication with ground personnel. Both receptacles are the "single plug" (combined microphone and headset plug) type.

3. Controls and Indications:

(See Figure 8.)

A. EMG Switch:

Selection of the red EMG switch enables the emergency operation mode. This permits emergency operation of the audio system, bypassing the isolation amplifier. In the EMG position, the pilot may select the receivers he desires by using the audio selector switch but without isolation or speaker operation. The headset must be used in this mode.

B. SPK Switch:

The SPK switch controls use of the speaker or headset. Depressing the SPK switch connects the speakers to the system (headphones will remain operational). Releasing the SPK switch (switch extended) disconnects the speakers from the system.

C. S.T. Switch:

The S.T. switch controls transmitter sidetone. When the S.T. switch is depressed, sidetone is heard regardless of which pilot makes the transmission. When the switch is released (extended), no sidetone is heard.

D. VOL Control Knob:

The volume (VOL) control knob selects an audio level for all receivers feeding into the audio system and automatically maintains the selected level on all receivers with one exception: ADF receivers in the unfiltered position (voice and identifier) are neither automatically level-controlled nor affected by the VOL knob.

In the filtered position (voice remaining only), ADF receivers are automatically audio level controlled and under the control of the VOL knob. The marker receiver, however, is neither automatically level-controlled nor regulated by the VOL knob. This permits the pilot to detect audio build or fade as approach to, or departure from, the marker occurs.

E. AUTO Switch:

The AUTO switch enables the auto select feature when depressed. When depressed, each receiver's AUDIO is automatically selected when the receiver's MICROPHONE switch is selected. When not depressed, each receiver's AUDIO and MICROPHONE switch must be selected individually.

F. FILT Switch:

When depressed, the FILT switch filters out or removes the 1020 Hz code identifier signal of its associated NAV or ADF receiver leaving only the voice signal. In the released position, identifier and voice are both heard on

selected receiver.

G. AUDIO Switches:

The NAV1, NAV2, DME1, DME2, VHF1, VHF2, VHF3, HF1, HF2, ADF1, ADF2, MKR1, MKR2, ILS1 and ILS2 pushbutton switches select their respective receiver audio when in the depressed position.

H. VHF1, VHF2, VHF3, HF1 and HF2 MICROPHONE Switches:

These switches connect both the hand and boom or hand and mask microphones to the desired transmitter for communications. Microphone selection is accomplished by interlocking switches, permitting only one button at a time to be depressed.

I. H'MIC Switch:

The H'MIC switch is an orange push-push type switch and is not interlocked with the other MICROPHONE selector switches. When depressed, the pilot and copilot may communicate on the intercom system through the boom microphone circuit without depressing the ICS key on the control yoke. The desired transmitter may remain selected on the MICROPHONE section while the H'MIC switch remains depressed. In this configuration, it is only necessary to depress the RADIO key on the control yoke to make a transmission. Upon releasing the RADIO key, crew intercom operation immediately returns without further keying. With the H'MIC switch released, interphone communications between pilot and copilot will only take place when the ICS button on the control yoke is depressed.

J. MASK Switch:

The MASK switch is an orange push-push type switch and is not interlocked with the other MICROPHONE selector switches. When depressed, the pilot and copilot may communicate on the intercom system through the mask microphone circuit. The desired transmitter may remain selected on the MICROPHONE section while the MASK switch remains depressed.

K. PAGE Switch:

The PAGE switch connects the microphones to the paging system in order to page personnel or make announcements on the cabin speaker system. When the PAGE switch is depressed, all audio from the radio receiver to the cockpit speakers or headphones is silenced to prevent crosstalk or receiver audio going to the cabin.

L. CABIN Switch:

The CABIN switch connects the audio system to the cabin two-way hand telephone for communications with the cabin. When the CABIN switch is depressed, all audio from the radio receiver to the cockpit speakers or headphones is silenced to prevent crosstalk or receiver audio going to the cabin.

M. Circuit Breakers:

The audio control / amplifier No. 1 is powered from the 28 Vdc emergency bus via CKPT AUDIO #1 circuit breaker. The audio control / amplifier No. 2 is powered from the 28 Vdc main bus via CKPT AUDIO #2 circuit breaker. Both circuit breakers are located on the copilot circuit breaker panel.

The intercommunications system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
CKPT AUDIO #1	CP	H-8	FWD EMER BATT
CKPT AUDIO #2	CP	I-8	L MAIN DC Bus

4. Limitations:

A. Headsets:

The flight crew shall wear headsets with acoustical protection when operating the airplane in the “green” configuration.

ST
(Side Tone)
- Controls transmitter side tone.
- When switch is depressed, side tone is heard regardless of which pilot makes transmission.
- When switch is released, no side tone is heard.
- Airplane SN 1485 and subsequent has an adjustable knob.

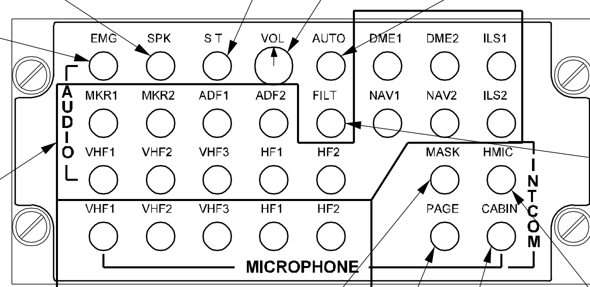
VOL
(Volume)
- Selects audio level for all receivers feeding into audio system.
- Automatically maintains selected level on all receivers except for ADF receivers in unfiltered position.

SPK
(Speaker)
- Controls use of cockpit speakers with system.
- Depressing switch connects speakers to system (headphones will remain operational).
- Releasing switch disconnects speakers from system.

AUTO
(Automatic Select)
- Controls auto select feature.
- When depressed, each receiver's AUDIO is automatically selected when receiver's MICROPHONE switch selected.
- When not depressed, each receiver's AUDIO and MICROPHONE switch must be selected individually.

EMG
(Emergency Mode, Red In Color)
- Enables the emergency operation mode.
- Bypasses isolation and speaker operation.
- Headset must be used in this mode.

FILT
(Filter)
- When depressed, filters out or removes 1020 Hz code identifier signal of associated NAV or ADF receiver, leaving only voice signal.
- In released position, identifier and voice are both heard on selected receiver.



AUDIO Switches
Selects respective receiver audio when in depressed position.

MICROPHONE Switches
- Connects both hand and boom or hand and mask microphones to desired transmitter for communications.
- Interlocking-type switches: permits only one button at a time to be depressed.

MASK
(Mask Microphone, Orange In Color)
- Not interlocked with other MICROPHONE selector switches.
- With switch depressed, pilot and copilot may communicate on ICS through mask microphone circuit.
- Desired transmitter may remain selected on MICROPHONE section while MASK switch remains depressed.

H'MIC
(Hot Microphone, Orange In Color)
- Not interlocked with other MICROPHONE selector switches.
- With switch depressed, pilot and copilot may communicate on ICS through boom microphone circuit without depressing ICS key on control yoke.
- Desired transmitter may remain selected on MICROPHONE section while H'MIC switch remains depressed making it only necessary to depress RADIO key on the control yoke to transmit. Upon releasing RADIO key, ICS operation immediately returns without futher keying.
- With switch released, ICS operation between pilot and copilot only takes place when ICS button on control yoke is depressed.

PAGE
Connects microphones to paging system in order to page personnel or make announcements on cabin speaker system.

CABIN
Connects audio system to cabin two-way hand telephone for communications with cabin.

26023C01

Figure 8. Baker Audio Control Panel

2A-23-60: Cockpit Voice Recorder System

1. General Description:

The Cockpit Voice Recorder (CVR) system provides a means for recording the most recent thirty (30) minutes of all audio signals transmitted and received by aircraft crew members. This allows the recorded data to be retrieved for investigatory purposes. The CVR system is composed of the following units and components:

- CVR Unit
- CVR Control Unit
- Cockpit Area Microphone
- Impact Switch

2. Description of Subsystems, Units and Components:

A. CVR Unit:

The CVR is mounted in the right side of the tail compartment. It provides four (4) separate channels of voice recording of either transmitted or received signals that originate at the pilot's or copilot's stations and in the cockpit area. The fourth channel is a spare. Signals are recorded on an endless-loop magnetic tape for a maximum period of 30 minutes. The recorder medium is contained within a highly-protective enclosure to guard against potential damage resulting from an aircraft accident.

An underwater acoustic beacon is physically attached to CVR unit to aid in location of CVR unit if lost in a body of water. The acoustic beacon is automatically triggered upon contact with water and operates from internal battery power.

B. Voice Recorder Control Unit (VRCU):

The VRCU is located on the copilot's side console. It accepts voice inputs for the pilot, copilot and cockpit area microphone and then outputs the audio signals to the CVR unit. Pilot and copilot receiver and transmitter audios are routed to the CVR through selection on the Audio Control Panel (ACP) amplifiers, while cockpit area microphone audio signals are introduced into an internal preamplifier and then output to the CVR.

The VRCU provides the capability to perform internal testing of the CVR system using the TEST switch. The cockpit area microphone can also be tested by connecting a headset in the HEADSET jack.

Bulk erasing of the CVR tape can be accomplished using the VRCU's ERASE switch or using a BULK ERASE switch located on the right-hand radio rack test panel. Erasing capability, however, is interlocked with the nutcracker system and is only possible on the ground.

The CVR contains an hourmeter that indicates total hours that CVR has been used. The hourmeter has a scale 0 to 5000 hours.

C. Cockpit Area Microphone:

The cockpit area microphone is located on the windshield center post between the angle-of-attack indexer lights. It is positioned to maximize the recording of human conversation and other ambient sound in the cockpit area while at the same time suppressing engine noise.

D. **Impact Switch:**

The CVR impact switch is located adjacent to the CVR in the tail compartment. Activation of the switch is automatic upon an impact severe enough to overcome its set force limit. Activation of the switch signals the CVR to cease recording, saving the previously-recorded data, also triggering an annunciation lamp which illuminates continuously. A reset button is located on the impact switch to reset the system.

E. **CVR “Quick Test”:**

The following guidance is recommended as a suitable “quick test” of the CVR:

(1) Ensure a COMM radio is tuned to a continuous output frequency, such as ATIS.

(2) Plug headset into CVR headset jack. Ensure continuous output frequency can be heard through the CVR headset jack.

(3) Snap your fingers or clap your hands near the cockpit area microphone. Ensure cockpit area microphone audio can be heard through the CVR headset jack. (Some delay will be present.)

(4) Push the CVR TEST button. Ensure a tone can be heard through the CVR headset jack. Ensure the meter/needle jumps.

3. **Controls and Indications:**

(See Figure 9 through Figure 11.)

A. **Circuit Breakers (CBs):**

The cockpit voice recorder system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
CKPT VOICE RECORDER	CP	D-13	ESS AC Bus, ϕA [1]

[1] Applies to Aircraft 1096 and subsequent. For Aircraft 1000 and Aircraft 1002 through 1095 (excluding 1034), not having ASC 49 / 49A but having ASC 186, power source is the ESS DC bus.

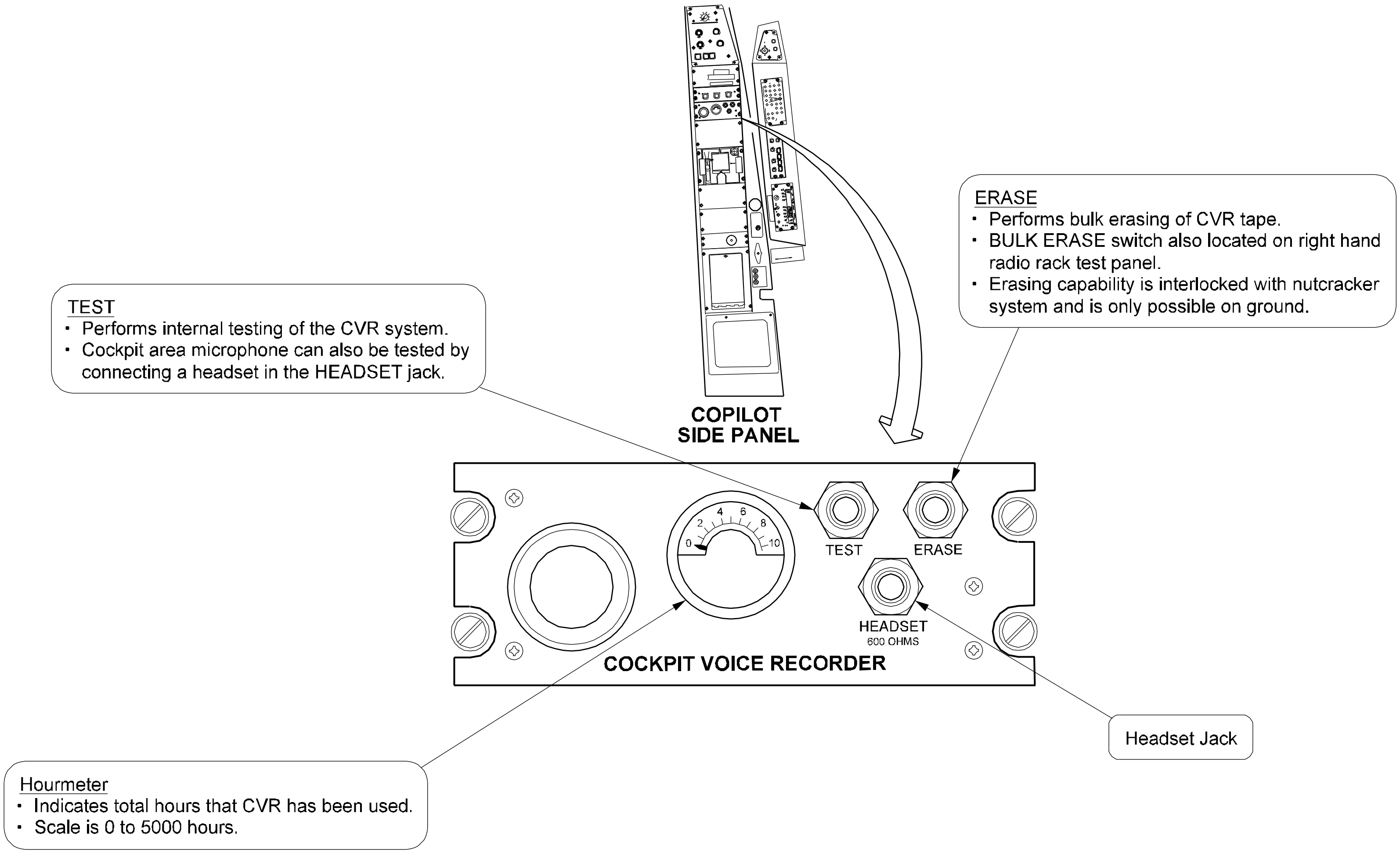
B. **Advisory (Blue) CAS Messages:**

CAS messages provided by the cockpit voice recorder system are:

CAS Message:	Cause or Meaning:
VOICE REC FAIL	Cockpit Voice Recorder has failed.

4. **Limitations:**

There are no limitations established for the cockpit voice recorder system at the time of this revision.



26024C00

Figure 9. Voice Recorder Control Unit

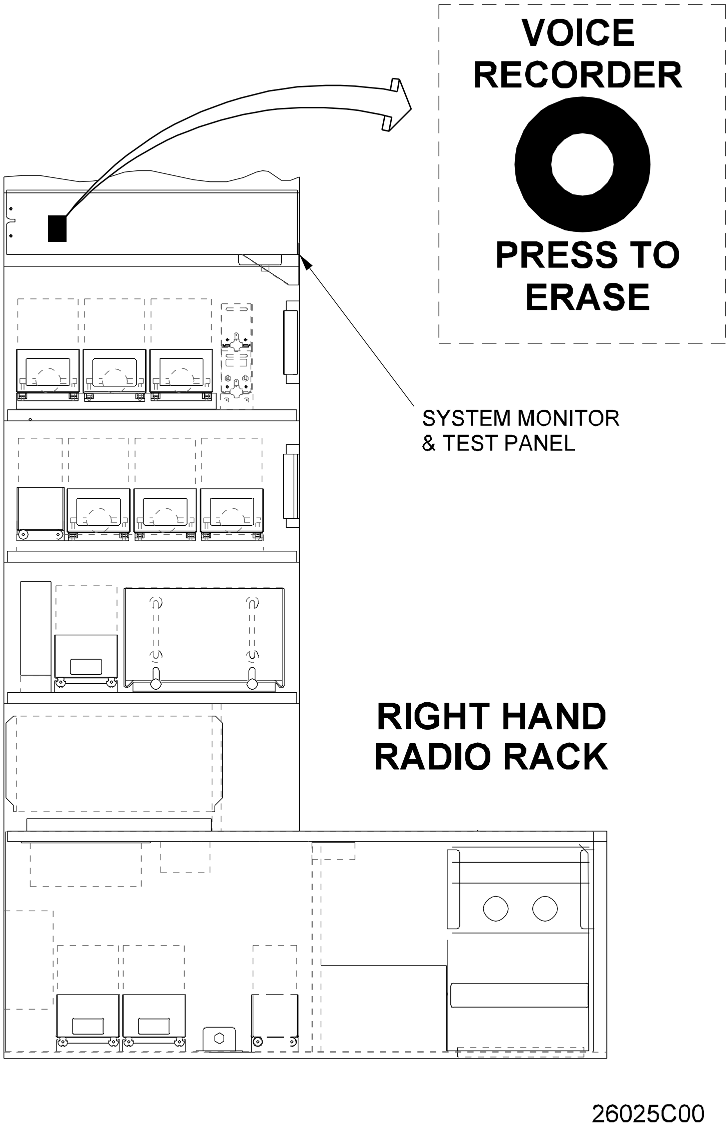


Figure 10. Voice Recorder BULK ERASE Switch

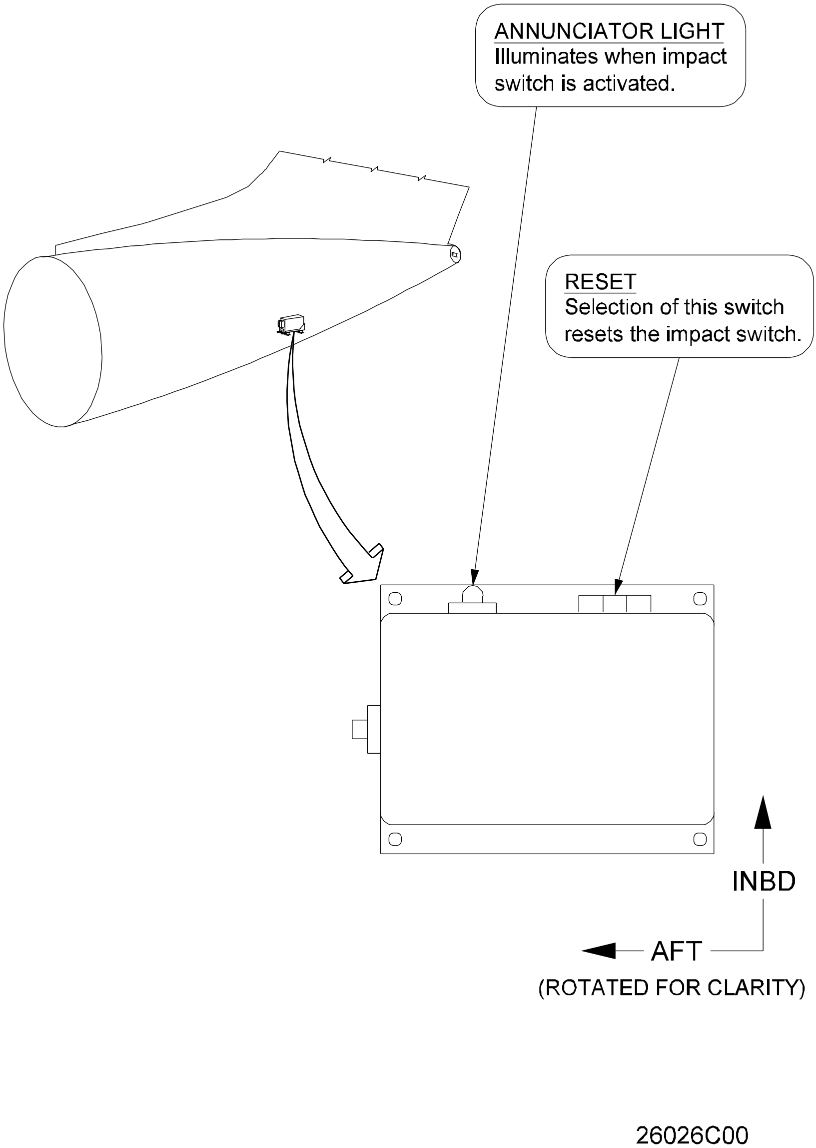


Figure 11. Voice Recorder Impact Switch

2A-23-70: Emergency Locator Transmitter System:

1. General Description:

The ARTEX 406 MHz Emergency Locator Transmitter (ELT) system provides triple frequency homing transmissions when activated. The system is certified under the requirements of TSO-C126. The ELT system transmits distress signals at frequencies of 121.5 MHz, 243.0 MHz, and a satellite frequency of 406.025 MHz. The transmitter will activate automatically under emergency conditions ('G' switch activation inside the ELT transmitter unit) or may be operated manually from the flight deck to summon assistance in an emergency situation.

The ELT system is composed of the following components:

- ELT Transmitter Unit
- ELT Navigation Interface Unit (Optional)
- Remote Switch
- Antenna
- Aural Monitor

2. Description of Subsystems, Units and Components:

A. ELT Transmitter Unit:

The ELT system automatically activates under emergency conditions ('G' switch activation) and simultaneously transmits a standard swept tone on 121.5 and 243.0 MHz. Typically, the 121.5 and 243.0 MHz transmitter will continue to operate for at least 72 hours until the transmitter has exhausted battery power. The 406.025 MHz transmitter operates for the first 24 hours of the 72 total hours and then shuts down automatically.

The 406.025 MHz satellite transmitter transmits every 50 seconds for a period of 520 milliseconds. During that time, an encoded digital message is sent to the receiving satellite. The information contained in that message is as follows:

- Country of registration
- Serial number of the ELT
- Latitude and longitude position coordinates (when coupled to the optional Navigation Interface Unit)

The ELT transmitter unit has a front panel containing, among other items, a control switch and a red LED indicator. See Figure 12. The control switch provides manual selection of ON, OFF, and RESET modes. When the ELT is transmitting, the red LED indicator on the front panel blinks. The light is not illuminated when the ELT is in the OFF (armed) mode. In addition to selection of ON and OFF, the control switch can be used to reset the ELT system from operating to armed by selection from OFF to ON for one second, then returning the switch to OFF. This is confirmed by the red LED indicator extinguishing. It should be noted that reset cannot take place if either the ELT transmitter unit control switch or the remote switch is selected to ON.

The ELT transmitter unit operates on a self-contained battery and requires no other power source. It is located in the baggage compartment, usually on the lower right side. The actual location may vary, however, depending on outfitting requirements.

B. ELT Navigation Interface Unit (Optional):

The ELT navigation interface unit provides continuous latitude and longitude position updates from the Flight Management System or Global Positioning Satellite system and translates the information to the proper format for use by the ELT. Operating without the navigation interface unit, the ELT transmitter unit provides position accuracy with 1 to 2 kilometers of its location. With the navigation interface unit, position accuracy is improved to within 100 meters.

The ELT navigation interface unit receives power from the Forward Emergency Battery Bus. It is installed in the baggage compartment adjacent to the ELT Transmitter Unit.

C. Remote Switch:

The ELT system remote switch is located on the right side of the copilot's flight panel. Two types of remote switches are installed on GIV airplanes, as shown in Figure 13. The functions of each switch are identical, regardless of the type.

The remote switch provides manual selection of ON, ARM, and RESET modes. When the ELT is transmitting, a red LED indicator blinks above the switch. The light is not illuminated when the ELT is in the ARM (armed) mode. In addition to selection of ON and ARM, the remote switch can be used to reset the ELT system from operating to armed by selection from ARM to ON for one second, then returning the switch to ARM. This is confirmed by the red LED indicator extinguishing. It should be noted that reset cannot take place if either the ELT transmitter unit control switch or the remote switch is selected to ON.

D. Antenna:

The ELT system uses a high performance, externally mounted antenna. It is located on the top aft portion of the fuselage adjacent to the left engine.

E. Aural Monitor:

An aural monitor provides a distinct signal enabling an airplane with a transmitting ELT to be located in an area with a large number of aircraft (e.g., an airport). A beeper, powered by the ELT transmitter is installed to provide this function. The beeper is not designed to operate continuously, but to sound at predetermined intervals. The beeper will sound for shorter periods toward the end of the battery life.

The beeper is loud enough to be heard outside the airplane in areas with low ambient noise. In areas with high ambient noise, the airplane will have to be physically checked for a blinking red LED indicator on either the ELT transmitter unit or remote switch panel.

The aural monitor beeper is usually located in the tail compartment. The actual location may vary, however, depending on outfitting requirements.

3. Controls and Indications:

(See Figure 12 and Figure 13).

4. Limitations:

A. Flight Manual Limitations:

There are no limitations established for this system as of this revision.

B. System Notes:

WARNING

DO NOT ALLOW OPERATIONAL TEST DURATION TO EXCEED FIFTEEN (15) SECONDS. TRANSMISSIONS THAT EXCEED 15 SECONDS MAY BE INTERPRETED BY THE SATELLITE SYSTEM AS A VALID DISTRESS SIGNAL. UNIT TESTING SHOULD BE COORDINATED WITH THE LOCAL FAA ATC TOWER AND BE CONDUCTED WITHIN FIVE (5) MINUTES AFTER THE HOUR.

(1) Reset of a transmitting ELT system cannot take place if either the ELT transmitter unit control switch or the cockpit remote switch is selected to ON.

(2) The ARTEX 406 MHz ELT battery must be replaced on or before the replacement date noted on the battery pack. Tests to verify proper operation of the unit must be accomplished after battery replacement.

(3) Operational tests, including ’G’ switch testing, of the ARTEX 406 MHz ELT must be accomplished every twelve (12) calendar months unless operational regulatory requirements dictate a more frequent schedule.

(4) The cockpit remote switch is required under TSO-C126 and may not be disabled or rendered inoperative.

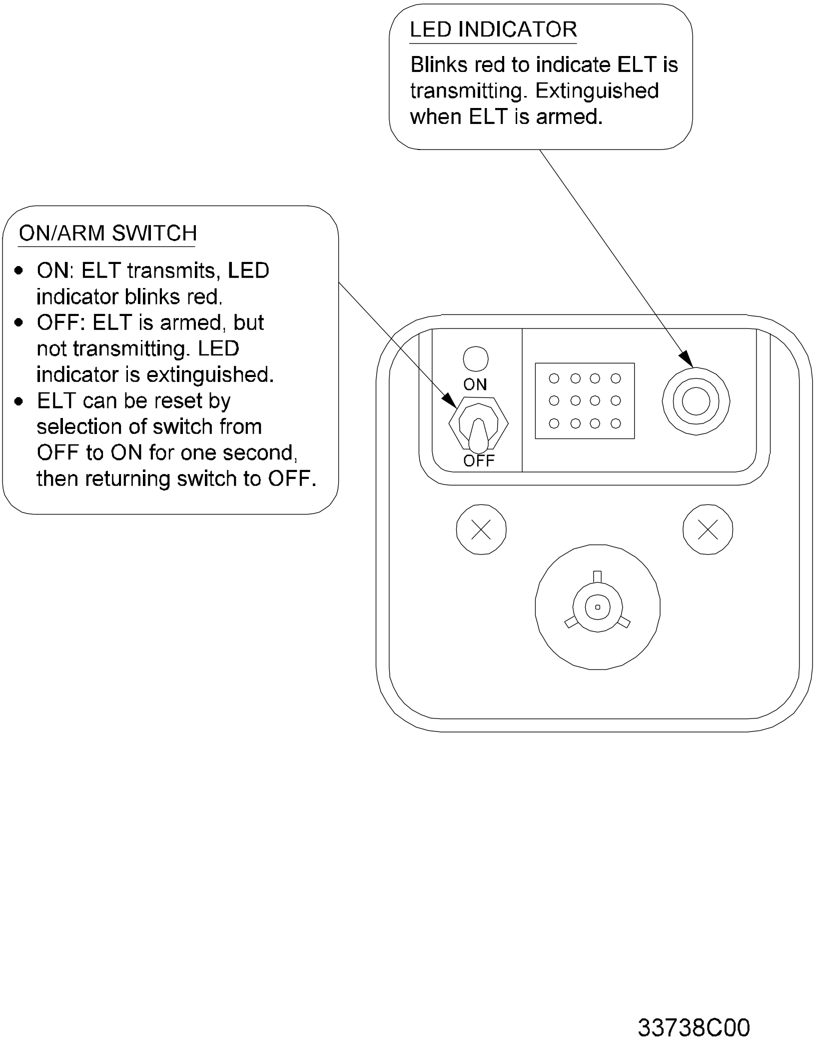


Figure 12. ELT Transmitter Unit Front Panel

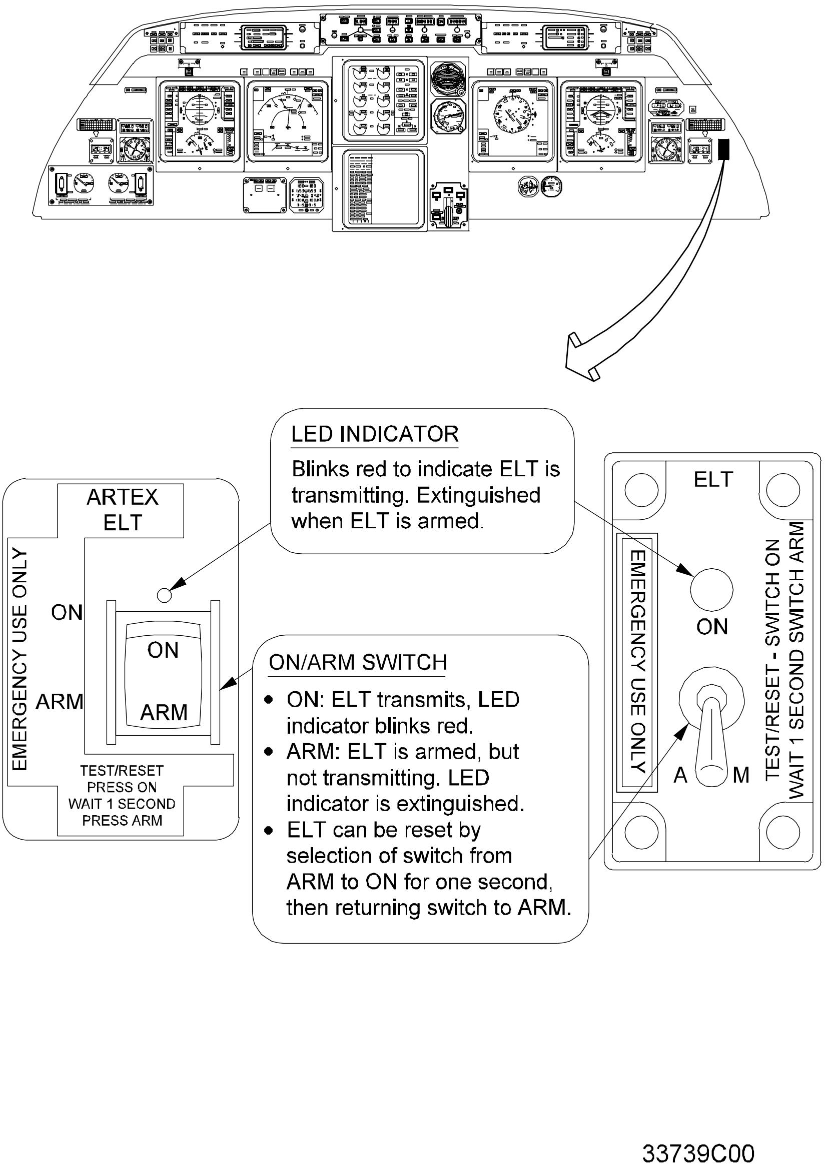


Figure 13. ELT Remote Switch

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2A-24-10: General

1. Purpose:

The purpose of the electrical power system is to provide the aircraft with Alternating Current (AC) and Direct Current (DC) power, and to provide a means of control, protection and distribution of electrical power required for ground and flight operations. The Gulfstream IV meets these needs through the use of its Variable-Speed, Constant-Frequency (VSCF) electrical power system, based primarily on AC power. The AC power is frequency regulated and converted to DC power for use by the standard aircraft systems. The AC and DC power is distributed by separate bus systems. Provisions to convert DC power back to AC through the use of an emergency inverter are incorporated.

2. General Description:

A. AC Electrical Power System:

(See Figure 1, Figure 2 and Figure 7.)

The primary source of electrical power for the Gulfstream IV is the AC electrical power system. AC electrical power can be provided to the aircraft under normal conditions in the following manner:

- Internally by two (2) engine-driven alternators, one mounted on each engine-driven gearbox, and the respective converter
- Internally through the use of an alternator mounted on the Auxiliary Power Unit (APU). Normally used for ground operations, the APU alternator can also serve as an alternate source of power in flight if one or both alternators fail.
- Externally through use of an external AC power supply

AC electrical power can be provided to the aircraft under abnormal/ emergency conditions through the use of an Emergency Inverter (commonly referred to as the E-Inverter) and/or a Standby Electrical Power System for Transformer Rectifier Unit (TRU) operation.

An AC Bus Power Control Unit (ACBPCU) controls distribution of AC power from its various sources to the AC distribution buses.

The ELECTRIC POWER MONITOR Panel (EPMP), located on the cockpit overhead panel, contains the majority of the electrical control and monitoring functions. In addition, some electrical system annunciators are incorporated on the overhead annunciator panel.

B. DC Electrical Power System:

(See Figure 1, Figure 2 and Figure 16.)

DC electrical power can be provided to the aircraft under normal conditions in the following manner:

- Internally by two (2) VSCF converters
- Internally by a Transformer Rectifier Unit (TRU), where fixed frequency AC power is rectified by the TRU to provide a 28 VDC nominal output
- Internally by two (2) nickel cadmium batteries, which are provided to start the APU, run the Auxiliary Hydraulic System (AUX) pump, and

to supply power to the Essential DC Bus when not powered by the TRU. Each battery has a dedicated battery charger to provide battery charging.

- Externally through use of an external DC power supply

DC electrical power can be provided to the aircraft under abnormal/ emergency conditions through the use of a battery-operated Emergency Power System and/or a Standby Electrical Power System to provide 28 VDC for Emergency Inverter operation.

Like the AC electrical power system, a DC Bus Power Control Unit (DCBPCU) controls distribution of DC power from its various sources to the DC distribution buses.

Also like the AC electrical power system, the EPMP contains the majority of the electrical control and monitoring functions. In addition, some electrical system annunciators are incorporated on the overhead annunciator panel.

C. ELECTRIC POWER MONITOR Panel (EPMP):

Shown in Figure 3, the EPMP is the major switching and indication control panel for the electrical power system. Located on the cockpit overhead panel, the EPMP provides the automatic (AUTO) mode of system operation. Provisions exist, however, to manually override AUTO operation and select any and all buses to any one of the possible desired power sources. (This provision is called hard selection.) Each channel of the system is provided with metering to read voltage, frequency and percent of loading.

Out-of-limit range markings consist of a red bar to the left of the indications for voltage, frequency and percent of loading. It should be noted here that out-of-limit ranges do not “latch”, rather, when normal ranges return, the out-of-limit range indicator extinguishes.

D. Remote Power Supply:

A remote power supply, located in the right avionics bay (Figure 2), is dedicated to the EPMP for switchlight, digit and out-of-limit indicators illumination. It contains redundant power supplies in the form of two identical boards having individual inputs and outputs; these boards being crosstied for reliability. All remote power supply outputs are fed directly to the EPMP through a dedicated interface.

If a power supply failure occurs or a single battery switchlight capsule is selected OFF, the remote power supply has an alarm circuit that will prompt an amber EPMP POWER FAIL message for display on the Crew Alerting System (CAS).

Input power sources to the remote power supply include two sources from the Essential DC bus and one source from each battery. In addition to the

two power supply boards, one additional card provides EPMP light dimmer control and failure detection circuitry power.

NOTE:

If all EPMP indications are lost and the amber EPMP POWER FAIL CAS message is displayed, it is possible that the overhead panel dimming rheostat (OVHD PNL knob) is in a position between fully OFF (which reverts the EPMP to full bright mode) and the beginning of the brightness controlling feature. Rotation of the OVHD PNL knob may remedy this anomaly.

3. Subsystems of the Electrical Power System:

The electrical power system is divided into the following subsystems:

- 2A-24-20: AC Electrical Power System
- 2A-24-30: DC Electrical Power System

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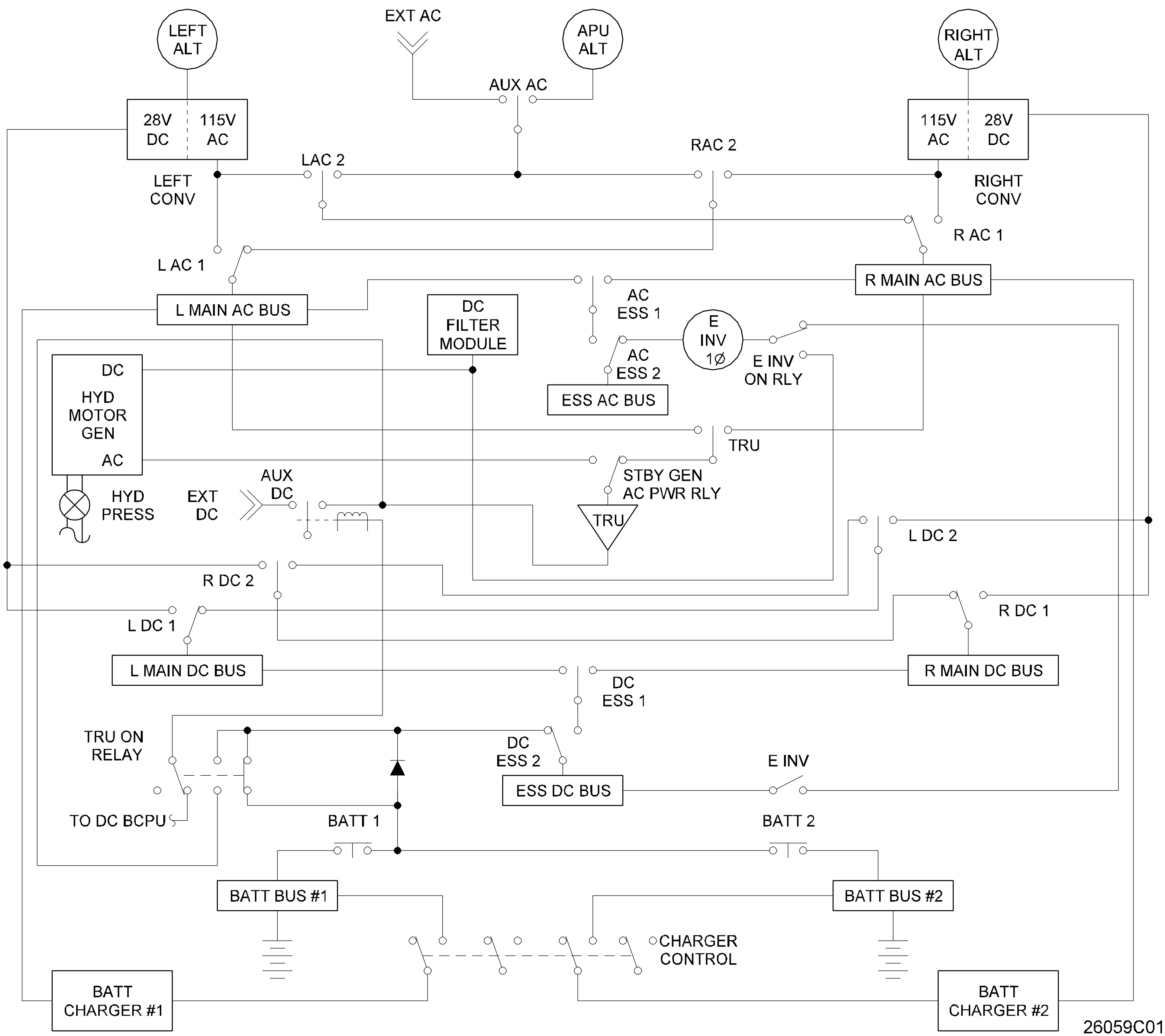
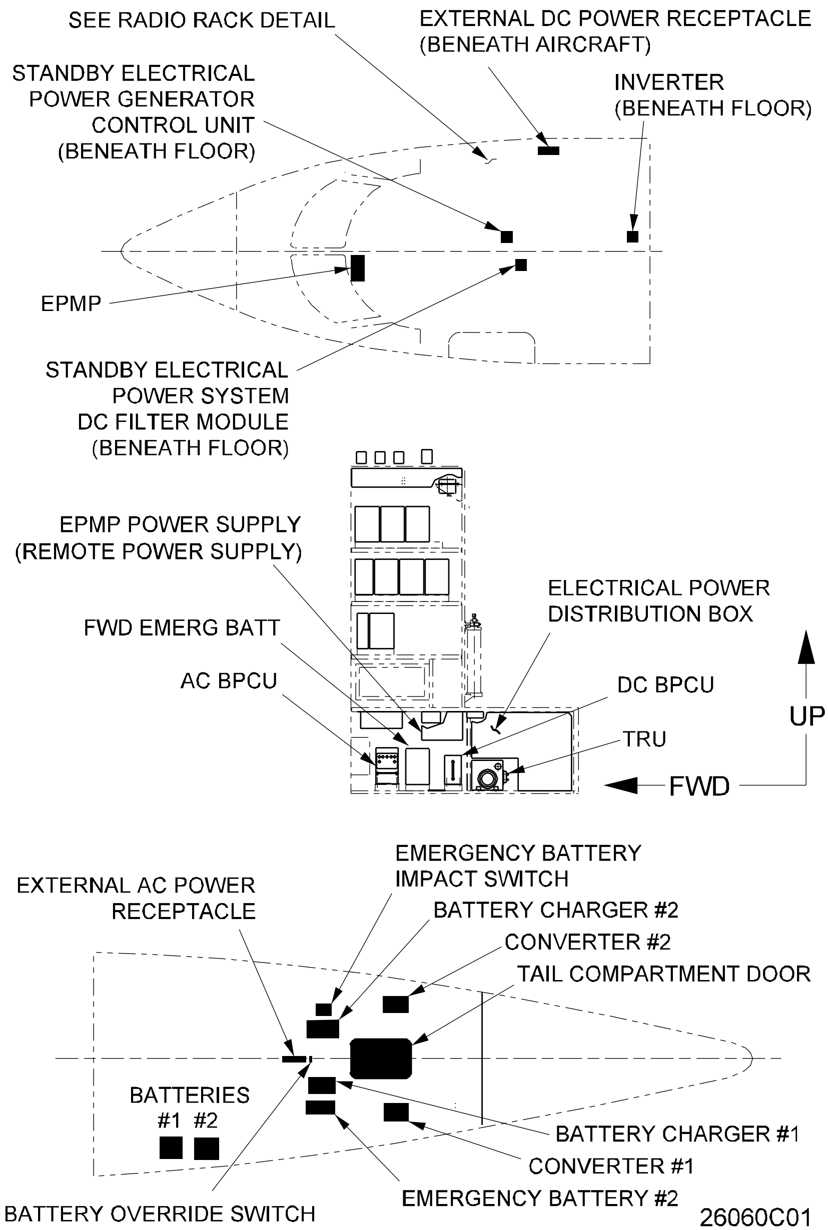
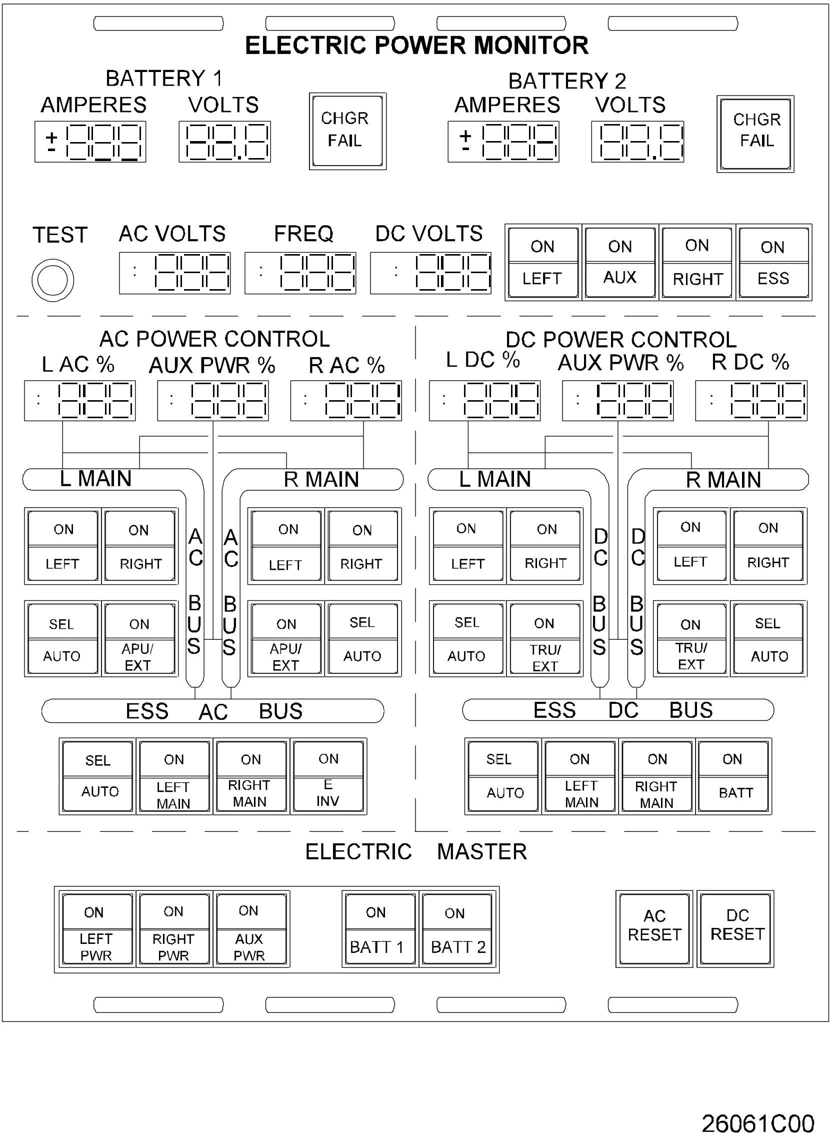


GIV Electrical Power System Simplified Block Diagram
Figure 1



GIV Electrical Power System Component Locations
Figure 2



GIV ELECTRIC POWER MONITOR Panel
Figure 3

2A-24-20: AC Electrical Power System

1. General Description:

The AC electrical power system supplies power to the AC electrical buses for distribution to AC systems and equipment. It is the primary source of electrical power for the GIV electrical power system. The AC electrical power system also supplies power to be converted to DC electrical power by the Transformer-Rectifier Unit (TRU).

Figure 1 shows the entire GIV electrical power system in simplified form; Figure 7 shows only the AC electrical power system in simplified form. Units and components for the AC electrical power system are shown in Figure 2. They are as follows:

- Engine-Driven Alternators
- APU Alternator
- Converters
- AC Bus Power Control Unit
- Emergency Inverter
- Power Distribution Box
- AC External Power System
- APU AC Power System
- Standby Electrical Power System
- Electrical Load Warning System
- Distribution and Control System

2. Description of Subsystems, Units and Components:

A. Engine-Driven Alternators:

The AC alternators used in GIV aircraft are air-cooled, brushless, permanent magnet type units that provide an alternating current power output. There are three identical AC alternators: one mounted on each engine-driven high speed gearbox and one mounted on the Auxiliary Power Unit (APU) in the tail compartment.

The engine-driven alternators are rated at 30 kVA and provide three-phase output. Cooling is accomplished by means of air entering an air inlet and circulating through the unit. Being air-cooled in this manner, the units are rated up to 50,000 feet.

Auxiliary bearings are mounted adjacent to the main bearings within the alternator. The auxiliary bearings take over support of the rotor in the event of main bearing malfunction. They act to prevent contact between the rotor and the stator, which could result in major damage.

Being engine-driven alternators, output frequency will vary with engine High Pressure (HP) RPM. Thus, in order to provide AC power at a constant 400 Hz frequency to the AC distribution buses, alternator output is first supplied to a Variable-Speed, Constant-Frequency (VSCF) converter. The converter then supplies 115/200V AC, 400 Hz power to the AC distribution buses.

Failure of an engine-driven alternator will be indicated by an amber L-R AC POWER FAIL or L-R DC POWER FAIL message displayed on CAS. This message is displayed whenever power is available from the Essential 28

VDC bus engine-driven alternator output or converter output has failed, or the ELECTRIC MASTER LEFT PWR or RIGHT PWR switch (Figure 6 Sheet 8) is OFF.

An amber L-R ALT HOT message will be displayed on CAS when an engine-driven alternator temperature exceeds 250° F (121° C).

An amber L-R ALT BRG FAIL message displayed on CAS indicates that an engine-driven alternator main bearing has failed, and the auxiliary bearing is operating. This condition is also indicated by an amber L-R ALT FAILED BRG light on the overhead annunciator panel illuminating, as shown in Figure 10. These messages indicate the alternator is operating ″on condition″ and further actions shall be with reference to the latest approved revision of the GIV Airplane Flight Manual.

B. APU Alternator:

The APU alternator is identical to the two engine-driven alternators. It is rated at 30 kVA, provides three-phase output and is also air-cooled. Internal bearing redundancy is identical to that of the engine-driven alternators.

Because the APU runs at a constant 100% RPM, the APU alternator turns at a constant speed of approximately 8,000 RPM. At this operating speed, the alternator provides 115 VAC power at a constant 400 Hz. Thus, the APU alternator output can be directly supplied to the AC distribution buses without first being output to a converter.

An amber APU ALT BRG FAIL message displayed on CAS indicates that the APU alternator main bearing has failed, and the auxiliary bearing is operating. This condition is also indicated by an amber APU ALT FAILED BRG light on the overhead annunciator panel illuminating, as shown in Figure 10. These messages indicate the alternator is operating ″on condition″ and further actions shall be with reference to the latest approved revision of the GIV Airplane Flight Manual.

Failure of the APU alternator will be indicated by an amber AUX AC POWER FAIL message displayed on CAS. This message is displayed whenever output power from the APU has dropped off line. If the APU alternator is running and the AUX PWR switch is ON, a blue ALTNTR OFF light located on the APU control panel (Figure 11) will be illuminated. A blue APU ALT OFF message will be displayed on CAS if the APU alternator is otherwise operational but has not been selected for use through the AUX PWR switch.

An amber APU ALT HOT message will be displayed on CAS when the APU alternator temperature exceeds 300° F (149° C).

C. Converters:

Two 30 kVA solid state converters are installed in the tail compartment: one to protect and control each engine-driven alternator. Each converter rectifies the AC alternator output into DC voltage, then converts it to 23 kVA of 115/200V AC, 400 Hz, three-phase power. In addition, the converters provide 250 amperes of DC power output regulated to 28 VDC. The output of the APU alternator does not require a converter since its frequency is fixed due to APU speed control.

Each converter contains extensive protection and fault sensing circuitry for the engine-driven alternator and AC power system. Among the anomalies

for which protection is provided are:

- AC Bus Faults
- Alternator Underspeed, Overfrequency, Underfrequency and Overvoltage
- Failed Converter Fans and Converter Overtemperature
- Converter Overvoltage, Undervoltage and Overcurrent

Normally, five high speed fans provide forced cooling air for each converter. If one of the five cooling fans should fail, an amber L-R CONV FAN FAIL message is displayed on CAS and the normally gray/black FAILED FAN flag (on the converter itself, shown in Figure 8) changes to white/black. These annunciations indicate the converter is operating ″on condition″ and further actions shall be with reference to the latest approved revision of the GIV Airplane Flight Manual. This is because if a second fan should fail, related CAS message(s) may be cleared, leading the flight crew to believe that the problem has corrected itself when, in fact, automatic converter shutdown may occur. With ASC 285 incorporated, all messages will remain illuminated.

When a converter exceeds 220° F (104° C), an amber L-R CONV HOT message is displayed on CAS. Continued operation with an overheated converter could result in automatic converter shutdown. The L-R CONV HOT message will clear when converter temperature falls to 190° F (88° C) and below.

An amber L-R AC POWER FAIL or L-R DC POWER FAIL message will be displayed on CAS if AC output from the converter drops off line.

D. AC Bus Power Control Unit:

The AC Bus Power Control Unit (ACBPCU), located in the right avionics rack, controls the connection of the AC power sources to the AC buses. By providing automatic control capability over the AC power relays and contactors, it controls the AC buses by switching AC bus power to an alternate source should a malfunction occur. Manual control over the relays and contactors is also available through the ACBPCU.

The ACBPCU provides regulation, excitation and electrical fault protection for the APU alternator, and electrical fault protection for the external AC power system and AC buses. It protects the APU alternator against overvoltage, undervoltage, undercurrent, underspeed and feeder faults. External power protection includes overvoltage, undervoltage, overfrequency, underfrequency, phase rotation, overcurrent and feeder fault protection. The AC buses are protected against overcurrent, undervoltage and feeder faults.

The ACBPCU also provides 28V DC (through an internal TRU) for the BATTERY OVERRIDE switch (sometimes referred to as the “dead battery switch”).

E. Emergency Inverter:

An Emergency Inverter (commonly referred to as the E-inverter) is installed under the forward floor. In the event of Essential AC bus failure, it provides power to Phase A of the Essential AC bus. Considered primarily an emergency power source, sufficient power is supplied to operate only equipment regarded as essential for flight. Among the items available:

- Bleed Air Isolation Shutoff Valve
- Cabin Pressurization
- Cockpit Voice Recorder
- Electric Pitch Trim
- Engine Pressure Ratio
- Engine Oil Pressure
- Pitot Heat: Left and Standby
- No. 1 and No. 2 26V AC Transformers
- Power Lever Angle
- Temperature Control System

Power for the Essential AC bus is normally supplied by from the Left or Right Main AC buses. Should a failure of all three primary power sources (left, right and APU alternators) occur, the E-inverter will automatically receive 28 VDC power from the Essential DC bus. The E-inverter in turn will automatically supply single-phase (Phase A only), 115 VAC, 400 Hz power to the Essential AC bus. The E-inverter can also be powered by the Standby Electrical Power system (commonly referred to as the ABEX generator) by manual selection as shown in Figure 4.

E-inverter operation is accomplished by ESS switchlight selection and verified by observing the VOLTS and FREQ meters. The ESS switchlight is located immediately below the BATTERY 2 CHGR FAIL annunciator on the EPMP (Figure 6 Sheet 1). Illumination of the amber ESS AC BUS E INV switchlight (AC POWER CONTROL section of the EPMP, Figure 6 Sheet 4) indicates a request for service only, not actual E-inverter operation.

F. Power Distribution Box:

The Power Distribution Box (PDB), located in the right avionics rack, is the load center of the electrical system. It serves as the terminal for all input power and output to the buses and other loads in the aircraft. Located on the front surface of the PDB are the load center circuit breakers, each protecting a separate channel of power.

G. AC External Power System:

The AC power system includes provisions for connecting an external source of regulated, three-phase, 115/200V AC, 400 Hz power. An external AC power receptacle (Figure 9) is located forward of the tail compartment door on the exterior of fuselage. With an external AC power supply connected and operating, a blue AC EXT PWR annunciator (Figure 10) on the overhead annunciator panel illuminates, indicating that external AC power is available. Selecting the AUX PWR switch and at least one BATT switch to ON (both shown in Figure 6 Sheet 8) will close the appropriate relays to energize the AC buses. The TRU will then be activated to power the DC buses.

If the main aircraft batteries are discharged or removed, a BATTERY OVERRIDE SWITCH, located within the external AC receptacle, may be used to initiate connection of AC external power by routing AC external power (Phase A) directly to the ACBPCU. The ACBPCU then changes this voltage to DC and routes it to the AUX PWR switchlight. When connection is made through switch selection, the circuit from the ACBPCU holds the

power until disconnect is selected. Circuit protection from improper AC external power is also provided by the ACBPCU.

H. APU AC Power System:

The APU alternator supplies AC power be distributed for electrical system use. With the APU operating and the APU generator ready, a blue ALTNR OFF annunciator on the APU control panel (Figure 11) is illuminated and a blue APU ALT OFF message is displayed on CAS. Depressing the AUX PWR switchlight on the EPMP (Figure 6 Sheet 8) routes APU alternator power to the AC buses and the annunciations are removed.

I. Standby Electrical Power System:

The Standby Electrical Power system consists of a hydraulic motor-driven generator and a Generator Control Unit (GCU). When no AC power is available from either main AC bus or the auxiliary power source, Phase A of the Essential AC bus and the Essential DC bus can receive power from this system.

The Standby Electrical Power system serves to permit continued flight at cruise altitudes after a double alternator/converter failure in order spare heavy loads from the main aircraft batteries. Upon descent through maximum APU starting altitude, the Standby Electrical Power system is normally replaced with APU alternator power (AUX PWR). Although the Standby Electrical Power system is not intended for use during approach and landing (as demands on the hydraulic system could cause it to drop off line), it may be used according to the limitations provided in the latest approved revision of the GIV Airplane Flight Manual. Also, battery charging when using the Standby Electrical Power system is not possible.

The constant-speed hydraulic motor-driven generator, located in the main wheel well, is commonly referred to as the ABEX generator. The hydraulic motor receives power from Combined hydraulic system pressure through a normally open shutoff valve. With Essential DC bus power available, the shutoff valve is energized closed. Loss of Essential DC bus power allows the valve to open, allowing approximately 1,000 psi of Combined hydraulic system pressure to spin up the motor-generator to its GCU-regulated maximum speed of 12,000 RPM. When on-speed, the motor-generator provides 5 kVA of 115 VAC, 400 Hz power to the TRU for the Essential DC bus. It also supplies 50 amperes of filtered 28 VDC power to the E-inverter for Phase A of the Essential AC bus.

The GCU serves to regulate motor-generator voltage and provide generator field excitation. Deviations of output voltage and frequency from reference values causes the GCU to apply corrective signals or, if necessary, disconnect generator output altogether.

The STANDBY ELECTRICAL POWER control panel, shown in Figure 4, is located above the EPMP on the cockpit overhead panel. It contains digital displays showing AC voltage and frequency, DC voltage, and AC and DC percent loading. Below the displays are three switchlights: STBY ELEC, TRU and E INVERT. Manual selection of the STBY ELEC switchlight to ON removes the Essential DC bus power from the shutoff valve, allowing the motor-generator to operate. (The UTILITY PUMP, if previously selected to ARM, will also operate and should be selected OFF if not needed.)

With the motor-generator operating, selection of the E INVERT switchlight

to ON supplies 28 VDC power to the E-inverter. The E-inverter in turn supplies AC power for Phase A of the Essential AC bus.

Selection of the TRU switchlight to ON supplies AC power from the motor-generator to the TRU. The TRU in turn supplies DC power to the Essential DC bus.

With the STANDBY ELECTRICAL POWER system operating, loads are normally shed by the flight crew to reduce demand on the system. AC loads are reduced by shedding DC-powered equipment and DC loads are reduced by shedding AC-powered equipment.

J. Electrical Load Warning System:

(Aircraft SN 1156 Through 1429 and SN 1000 Through 1155 with APU LOAD Meter Not Having ASC 420)

On Aircraft SN 1156 through 1429 and SN 1000 through 1155 with APU LOAD Meter but not having ASC 420, an Electrical Load Warning System (ELWS) is installed. The ELWS is designed to provide electrical load management for the APU electrical system when the APU alternator is used to replace a failed alternator/converter during operations above 30,000 feet Pressure Altitude (PA). It is not intended for daily operational use.

The ELWS consists of three Line Replaceable Units (LRUs) that interface with the aircraft's electrical system, air data computers (ADCs), and annunciation systems. The three LRUs are a three-phase current transformer, an electrical load warning computer (also referred to as an electrical load processing unit) and an electrical load warning indicator (more commonly referred to as the APU LOAD meter). The APU LOAD meter is located on the lower left side of the pilot's flight panel and is shown in Figure 5.

The current transformer receives three-phase AC inputs from the APU alternator and provides it to the electrical load warning computer. The electrical load warning computer rectifies the AC input, sums the resulting DC voltage and determines the total APU load in kVA. The electrical load warning computer then outputs the APU load to the APU LOAD meter for display.

The APU LOAD meter consists of an analog meter, an amber TEST light and an amber ADC light. Amber and green bands on the meter's face denote the normal and caution ranges of APU alternator loading in kVA. High and low values of the amber and green bands are altitude-dependent as follows:

ALTITUDE	WARNING LOW RANGE	CAUTION LOW RANGE	NORMAL RANGE
(FT PA)	(RED BAND)	(AMBER BAND)	(GREEN BAND)
≤ 32,000	< 2.1 kVA	2.1 - 3 kVA	3 - 16 kVA
≥ 34,000	< 2.1 kVA	2.1 - 3 kVA	3 - 14 kVA

ALTITUDE (FT PA)	CAUTION HIGH RANGE (AMBER BAND)	WARNING HIGH RANGE (RED BAND)
≤ 32,000	16 - 17 kVA	> 17 kVA
≥ 34,000	14 - 15 kVA	> 15 kVA

In addition to monitoring APU load, the electrical load warning computer is capable of automatically shedding excess APU loads to protect generating capabilities by monitoring Pressure Altitude (PA) through the ADCs. At or above 34,000 feet PA, the electrical load warning computer will automatically shed windshield heat based on the bus powered by the APU alternator, i.e., left windshield heat controller if APU alternator is powering the Left Main AC bus, right windshield heat controller if APU alternator is powering the Right Main AC bus. If the operating alternator/converter should fail, the electrical load warning computer will automatically shed galley power. Once at or below 32,000 feet PA, windshield heat and galley power are automatically restored.

In order for ELWS to become active when selected ON, the following selections must first occur: windshield heat must be selected ON, aircraft electrical power sources must be selected ON and proper EPMP configuration must be established. If these conditions are not met, an amber ELWS FAIL annunciator is illuminated above the pilot's and copilot's navigation display. The system can be reset and the annunciation extinguished by ensuring windshield heat is selected ON and cycling the APU LOAD meter OFF and back to ON.

The electrical load warning computer continuously performs self-monitoring of its hardware and software. It also monitors annunciator outputs for excessive current conditions during normal system operation. When operating, the ELWS provides outputs for the following annunciations described here and shown in Figure 5 and Figure 12.

(1) An amber TEST light illuminates on the APU LOAD meter when ELWS detects a warm or cold start with the aircraft on the ground. It also illuminates briefly in flight as part of the self-test.

(2) An amber ADC light illuminates on the APU LOAD meter when ELWS determines that one or both ADC inputs is invalid with the aircraft on the ground. It also illuminates in flight when ELWS determines that one ADC input is invalid. If both ADC inputs are invalid, an amber ELWS FAIL annunciator is illuminated above the pilot's and copilot's navigation display.

(3) An amber APU LOAD annunciator is illuminated above the pilot's and copilot's navigation display when APU electrical load for operation above 30,000 feet PA is out of limits. The APU electrical load should then be adjusted as required.

(4) A red APU LOAD annunciator is illuminated above the pilot's and copilot's navigation display when APU electrical load for operation above 30,000 feet PA is out of limits and should be adjusted as soon as possible. With this annunciator illuminated, the CABIN MASTER is normally selected OFF.

(5) An amber ELWS CONFIG annunciator is illuminated above the pilot's and copilot's navigation display when ELWS determines that the EPMP is improperly configured for ELWS operation. On the ground, illumination occurs simultaneously with illumination of the amber ADC light on the APU LOAD meter when one air data source is invalid.

(6) An amber ELWS FAIL annunciator is illuminated above the pilot's and copilot's navigation display when ELWS determines the system is unreliable for operation above 30,000 feet PA. Normally, the system can be reset and the annunciation extinguished by ensuring windshield heat is selected ON and cycling the APU LOAD meter OFF and back to ON.

(7) A white ON light illuminates on the APU LOAD meter when ELWS is powered and operational. If ELWS detects a fault and is unable to illuminate the ELWS FAIL annunciator, the ON light will flash twice per second to alert the flight crew of the ELWS failure.

Airplanes with ASC 420 (Removal of APU LOAD Meter), and SN 1430 and subs: Airplanes SN 1000 through SN 1155 with both APU LOAD Meter and ASC 420 incorporated, SN 1156 through 1429 with ASC 420 incorporated, and SN 1430 and subs have a ram air scoop installed over the APU inlet door in order to direct more ram air into the APU. This modification improves the high altitude performance of the APU, making the APU LOAD meter unnecessary and thus it is removed by the ASC.

K. Distribution and Control System:

AC power control and distribution to the AC buses takes place through the EPMP, PDB and ACBPCU. The bus system then distributes power to the various aircraft systems. The AC bus system consists of the Left Main AC bus, Right Main AC bus and Essential AC bus.

Normal system load during alternator operation consists of the left converter powering the Left Main and Essential AC buses, and the right converter powering the Right Main AC bus. If an alternator fails, the remaining alternator automatically assumes the total electrical load.

Transfer relays allow either the Left or Right Main AC bus to receive power from either the left or right converters. In addition, either or both main AC buses can receive power from the APU or, if on the ground, external AC power. The transfer relays also prevent the inadvertent connection of two power sources to one system. The Essential AC bus provides power for essential inflight loads. It can receive three-phase power from either of the main AC buses or single-phase power from the E-inverter powered by the Essential DC bus.

The ACBPCU automatically connects the AC buses to the appropriate power source in a set order. If more than one power source is available, the system automatically assigns power to the:

(1) Left Main AC bus from:

  (a) Left alternator/converter

  (b) APU alternator or external AC power

  (c) Right alternator/converter

(2) Right Main AC bus from:

(a) Right alternator/converter

(b) APU alternator or external AC power

(c) Left alternator/converter

(3) Essential AC bus from:

(a) Left Main AC bus

(b) Right Main AC bus

(c) E-inverter

Circuit breakers and overload sensors protect the AC bus distribution system from shorts and overloads. If a circuit breaker's rated amperage is exceeded, the circuit breaker will automatically open to protect the component. Circuit breakers may also be manually opened as necessary by the flight crew. Regardless of how circuit breakers are opened, they must always be manually closed.

Overload sensors are used in large current applications to disconnect a system when an overload condition exists. The sensors are operated thermally and although the sensor itself is self-resetting, the circuit breakers that the sensors may open as a result of overload must be manually closed.

3. AC Electrical Power System Operation:

(See Figure 6.)

The flight crew controls the electrical power system through the EPMP. It consists of 13 digital displays and 33 switchlights that allow the selection of electrical power sources and display of voltages, amperages, frequencies and percentages of total load.

When normally configured (L MAIN, R MAIN and ESS AC BUS switches selected to AUTO), the EPMP allows automatic operation and selection of electrical power sources in order to minimize flight crew workload. Manual operation is always available, however, in order to hard-select or deactivate power sources.

With external AC power available and the ELECTRIC MASTER AUX PWR switch selected ON, external AC power is supplied to the Left Main, Right Main and Essential AC buses. Power is also available to both battery chargers and the TRU.

With the alternators operating and converters supplying power, selection of the ELECTRIC MASTER LEFT PWR and RIGHT PWR switches to ON allows left converter output to the Left Main AC bus and Essential AC bus, and right converter output to the Right Main AC bus. Simultaneous bus connection to two power sources is impossible, thus no interlock is incorporated.

If the APU is the only source of power and the AUX PWR switch is selected to ON, the APU alternator powers the Left Main AC bus, Right Main AC bus and Essential AC bus (through the Left Main AC bus).

The Essential AC bus provides power to equipment essential for safe flight. Normally, the Left Main AC bus supplies the Essential AC bus, but the Right Main AC bus, E-inverter (Phase A only) or batteries (through the E-inverter) are also capable of powering the bus.

Illumination of the AC RESET amber switchlight indicates a transient problem, bus fault or logic problem. If the problem is a Left Main or Right Main AC bus fault or overcurrent condition, depressing the switchlight resets the system. If the

switchlight extinguishes, the fault has been cleared. If it remains illuminated, the fault remains. If the problem is an Essential AC bus fault or overcurrent condition, the ACBPCU will automatically transfer the bus to another power source in an attempt to remain powered.

4. Controls and Indications:

A. Circuit Breakers:

Circuit breakers controlling the AC bus distribution system are shown in bold print in the tables that follow. Circuit breakers receiving power from the controlling circuit breaker are shown immediately following that circuit breaker.

Example: The L TEMP CONT AC circuit breaker receives power from the PILOT ϕA section of the Essential AC bus. Thus the PILOT ϕA circuit breaker is the controlling circuit breaker.

Essential AC Bus CBs
Table 1

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT ϕ A	**PDB**	**AC ESS**
L TEMP CONT AC	P	H-11
R TEMP CONT AC	P	I-11
BLEED AIR ISO S/O V	PO	D-12
CABIN PRESS 115V	PO	D-11
ESS AC ϕA VM	P	A-13
BOT A/C LT (ASC 10)	P	D-6
PILOT ϕ B	**PDB**	**AC ESS**
ESS AC ϕB VM	P	B-13
PILOT ϕ C	**PDB**	**AC ESS**
ESS AC ϕC VM	P	C-13
COPILOT ϕ A	**PDB**	**AC ESS**
L EPR 115V	CP	A-9
R EPR 115V	CP	B-9
L PITOT HT HTR	CP	L-11
STBY PITOT HT PWR	CP	M-13
#1 26VAC XFMR	CP	A-10
L ENG OIL PRESS	CP	A-12
ELEC TRIM EXC #1	CPO	C-1
PLA EXC #1	CPO	C-2
#2 26VAC XFMR	CP	B-10
R ENG OIL PRESS	CP	B-12
ELEC TRIM EXC #2	CPO	D-1
PLA EXC #2	CPO	D-2
CKPT VOICE RECORDER (1096 & Subs.)	CP	D-13
POSN SNSRS	CP	C-10
FDR/FDAU	CP	C-7
DISPLAYS FAN #1	CP	D-5
COPILOT ϕ B	**PDB**	**AC ESS**
ILS #1 (1000-1095)	CP	I-3
COPILOT ϕ C	**PDB**	**AC ESS**
CKPT VOICE RECORDER (1000-1095)	CP	D-13
ACCESS	**PDB**	**AC ESS**

Left Main AC Bus CBs
Table 2

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT ϕ A	**PDB**	**AC LEFT**
NAV/INSP LTS XFMR	P	C-8
NAV LITES	P	E-8
WING INSP LTS	P	E-9
L LDG LT PWR	P	C-7
L MAIN AC ϕA VM	P	A-11
PILOT ϕ B	**PDB**	**AC LEFT**
L MAIN AC ϕB VM	P	C-11
PILOT ϕ C	**PDB**	**AC LEFT**
L MAIN AC ϕC VM	P	E-11
COPILOT ϕ A	**PDB**	**AC LEFT**
ENG VIB MONITOR	CP	B-8
L SIDE WSHLD	CP	J-10
R FRONT PWR	CP	J-8
#1 HF COMM	CP	C-10
COPILOT ϕ B	**PDB**	**AC LEFT**
AHRS AC	CP	M-6
W RDR R/T	CP	E-14
R FRONT PWR	CP	J-8
#1 HF COMM	CP	C-11
COPILOT ϕ C	**PDB**	**AC LEFT**
IRU #1 AC	CP	K-6
#1 HF COMM	CP	C-12
ACCESS ϕ A	**PDB**	**AC LEFT**
ACCESS ϕ B	**PDB**	**AC LEFT**
ACCESS ϕ C	**PDB**	**AC LEFT**
BATTERY CHARGER	**PDB**	**AC LEFT**

Right Main AC Bus CBs
Table 3

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT ϕ A	**PDB**	**AC RIGHT**
STROBE LTS	P	F-9
R MAIN AC ϕA VM	P	B-11
PILOT ϕ B	**PDB**	**AC RIGHT**
TAXI LTS XFMR	P	F-10
L TAXI LT PWR	P	C-10
R TAXI LT PWR	P	E-10
CTR TAXI LT PWR	P	D-10
R LDG LT PWR	P	E-7
R MAIN AC ϕB VM	P	D-11
PILOT ϕ C	**PDB**	**AC RIGHT**
BCN LTS (ASC 10)	P	C-6
TOP A/C LT (Pre-ASC 10)	P	C-6
R MAIN AC ϕC VM	P	F-11
COPILOT ϕ A	**PDB**	**AC RIGHT**
R PITOT HT PWR	CP	M-11
#2 HF COMM	CP	E-10
COPILOT ϕ B	**PDB**	**AC RIGHT**
ILS #2 (1000 - 1095)	CP	J-3
TOTAL TEMP PROBE HTR (1096 & Subs)	CP	L-10
L FRONT POWER	CP	L-8
#2 HF COMM	CP	E-11
DISPLAYS FAN #2	CP	D-6
NOSE COMPT COOL FAN	CP	M-1
COPILOT ϕ C	**PDB**	**AC RIGHT**
IRU #2 AC	CP	L-6
R SIDE WSHLD	CP	K-10
RR ENG TEST	CP	G-13
L FRONT POWER	CP	L-8
#2 HF COMM	CP	E-11
ACCESS ϕ A	**PDB**	**AC RIGHT**
ACCESS ϕ B	**PDB**	**AC RIGHT**
ACCESS ϕ C	**PDB**	**AC RIGHT**
BATTERY CHARGER	**PDB**	**AC RIGHT**
TRU	**PDB**	**AC RIGHT**

B. Crew Alerting System (CAS) Messages:

(1) Warning (Red) CAS Messages and Annunciations:

CAS Message	Cause or Meaning
APU LOAD	APU electrical load for operations above 30,000 ft not within limits.

Annunciation	Cause or Meaning
Red APU LOAD light above pilot's/ copilot's NAV display. (SN 1156 - 1252 equipped with SPZ-8000)	APU electrical load for operations above 30,000 ft not in limits.

(2) Caution (Amber) CAS Messages and Annunciations:

CAS Message	Cause or Meaning
L-R AC POWER FAIL	AC output from converter has dropped off line.
L-R ALT BRG FAIL	Alternator main bearing has failed and is operating on auxiliary bearing.
L-R ALT HOT	Alternator temperature is above 250° F (121° C).
APU ALT BRG FAIL	Alternator main bearing has failed and is operating on auxiliary bearing.
APU ALT HOT	APU alternator temperature above 300° F (149° C).
APU LOAD [1]	APU electrical load for operation above 30,000 ft not within limits.
AUX AC POWER FAIL	Power output from APU alternator has failed or dropped off line.
L-R CONV FAN FAIL	A converter cooling fan has failed.
L-R CONV HOT	Converter temperature is above 220° F (104° C).
EPMP POWER FAIL [2]	Either one (1) or all four (4) sources of input power supplied to the EPMP have failed.

[1] For SPZ-8400 equipped aircraft.

[2] If – in addition to this message – all EPMP indications are lost, it is possible that the overhead panel dimming rheostat (OVHD PNL knob) is in a position between fully OFF (which reverts the EPMP to full bright mode) and the beginning of the brightness controlling feature. Rotation of the OVHD PNL knob may remedy this anomaly.

Annunciation	Cause or Meaning
AC RESET light (amber) illuminated on EPMP.	Main AC Bus fault indicated.
L ALT FAILED BRG light (amber) illuminated on overhead panel.	Left alternator main bearing has failed and alternator is operating on auxiliary bearing.
APU ALT FAILED BRG light (amber) illuminated on overhead panel.	APU alternator main bearing has failed and alternator is operating on auxiliary bearing.
R ALT FAILED BRG light (amber) illuminated on overhead panel.	Right alternator main bearing has failed and alternator is operating on auxiliary bearing.
APU LOAD light (amber) illuminated above pilot's/copilot's NAV display. [1]	APU electrical load for operations above 30,000 ft not within limits.

Annunciation	Cause or Meaning
ELWS CONFIG light (amber) illuminated above pilot's/copilot's NAV display. [1]	APU electrical power improperly configured for operations above 30,000 ft.
ELWS FAIL light (amber) illuminated above pilot's/copilot's NAV display. [1]	APU electrical power unreliable for operations above 30,000 ft.

[1] SN 1156 - 1252 equipped with SPZ-8000.

(3) Advisory (Blue) CAS Messages and Annunciations:

CAS Message	Cause or Meaning
AC EXT POWER	AC external power is connected to airplane.
APU ALT OFF	APU alternator is operating but AUX PWR switch is not selected to ON.

Annunciation	Cause or Meaning
AC EXT PWR light (blue) illuminated on overhead panel.	AC external power applied.

5. Limitations:

A. Standby Electrical System:

When the Standby Electrical System is in operation, the following limitations apply:

(1) **Minimum HP RPM:**

Minimum HP RPM when the Standby Electrical System is in operation is 67 percent HP RPM.

(2) **Use of Speed Brakes:**

Speed brakes may be used, however, operation should be slow (approximately five (5) seconds for full range movement).

(3) **Landing with Standby Electrical System Operating:**

Landing is approved provided automatic ground spoilers and thrust reversers are not used for landing. See Section 05-17-50, Landing With Standby Electrical Power System Operating.

B. APU Alternator Electrical Load Envelope:

(1) **The following limitation applies for GIV airplanes SN 1480 and subs, SN 1156 through 1309 with Loadmeter installed but with or without ASC 470, and for SN 1000 through SN 1479 having either:**

- APU Loadmeter installed, **or**:
- ASC 420 (APU Loadmeter Removal)

OR:

ASC 427 (APU Enclosure Sooting Mod) AND ALSO HAVING EITHER:

- APU Loadmeter installed, **or**:
- ASC 420 (APU Loadmeter Removal)

The APU alternator can deliver 100% electrical power (30 kVA) on ground or in flight from Sea Level to 25,000 feet. From 25,000 feet

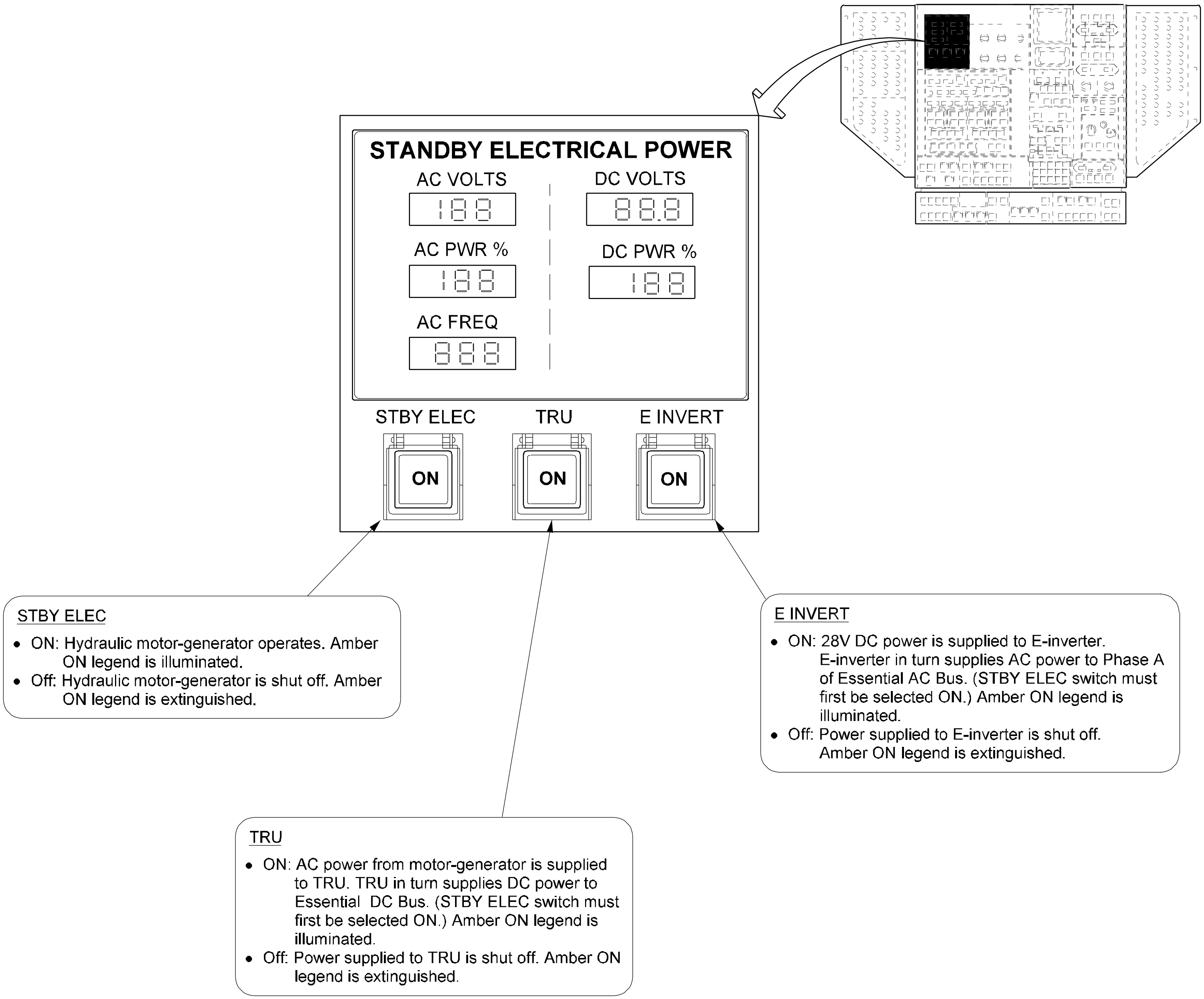
to 30,000 feet, the limit load decreases linearly to 83% (25 kVA). From 30,000 feet to 35,000 feet the limit load decreases to 67% electrical power (20 kVA). Load shedding may be required.

(2) **The following limitation applies for GIV SN 1000 through 1309 without APU Loadmeter and with or without ASC 470:**

The APU alternator can deliver 100% electrical power (30 kVA) on ground or in flight from Sea Level to 22,000 feet. From 22,000 feet to 30,000 feet, the limit load decreases linearly to 50% electrical power (15 kVA). Load shedding may be required.

(3) **The following limitation applies for GIV airplanes having ASC 465 (36-150[G] APU):**

The APU alternator can deliver 100% electrical power (30 kVA) on the ground or in flight from Sea Level to 15,000 feet (20,000 feet if airspeed is maintained below 300 KIAS). From 15,000 feet to 30,000 feet the limit load is 75% (22.5 kVA). From 30,000 feet to 35,000 feet the limit load is 50% (15 kVA). Load shedding may be required.



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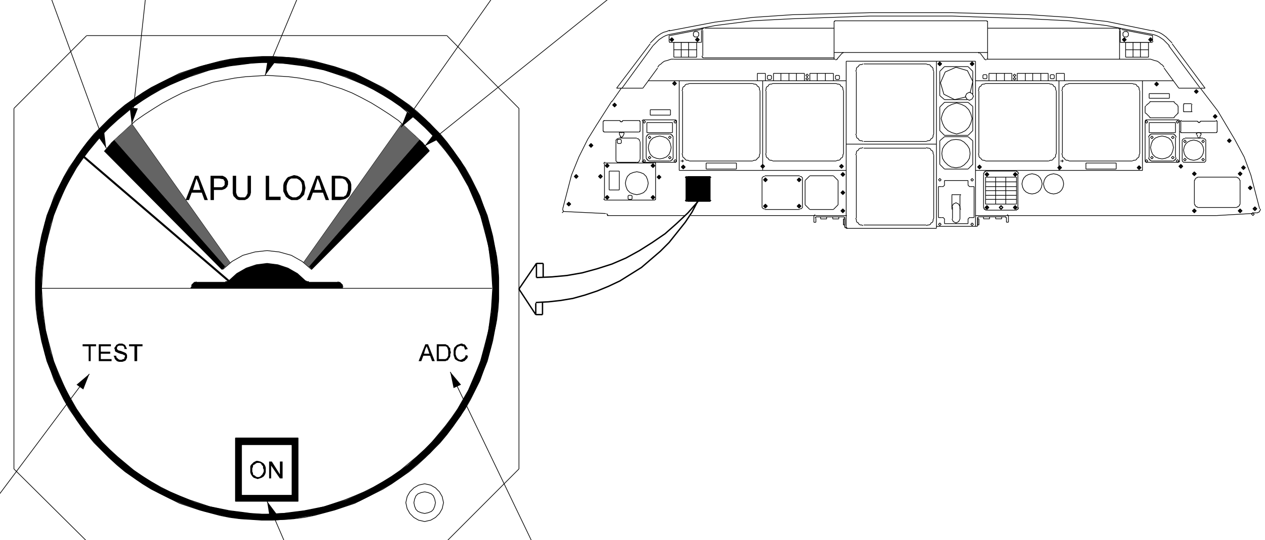
STANDBY ELECTRICAL POWER Control Panel
Figure 4

Altitude (Ft PA)	Warning Low (Red Band)	Caution Low (Amber Band)	Normal (GreenBand)	Caution High (Amber Band)	Warning High (Red Band)
≤ 32,000	< 2.1 kVA	2.1 - 3 kVA	3 - 16 kVA	16 - 17 kVA	> 17 kVA
≥ 34,000	< 2.1 kVA	2.1 - 3 kVA	3 - 14 kVA	14 - 15 kVA	> 15 kVA



TEST
- On Ground: Illuminates amber when ELWS detects a warm or cold start.
- In Flight: Illuminates briefly as part of self-test.

ADC
- On Ground: Illuminates amber when ELWS determines that one or both ADC inputs is invalid.
- In Flight: Illuminates amber when ELWS determines that one ADC input is invalid.

ON
- Illuminates steady (white) when ELWS is powered and operational.
- Flashes twice per second (white) if ELWS detects a fault and is unable to illuminate the ELWS FAIL annunciator, signifying ELWS failure.

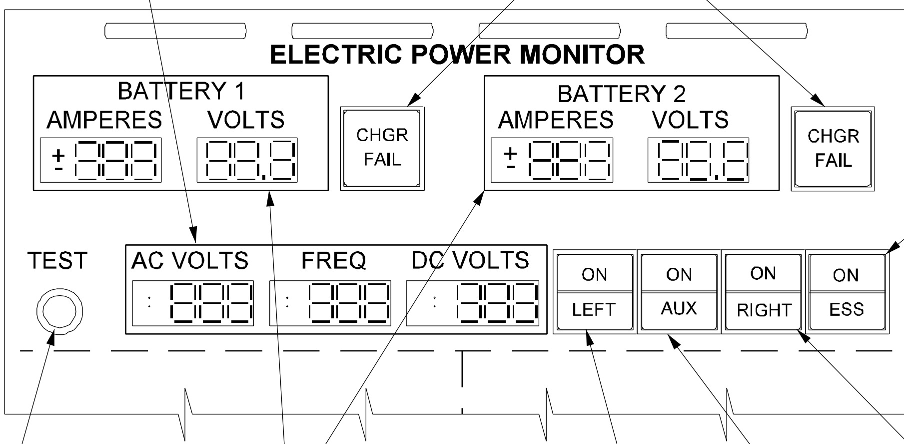
26070C00

APU LOAD Meter
Figure 5

AC VOLTS / FREQ / DC VOLTS
Displays AC voltage and frequency, and DC voltage from source selected by voltmeter monitor point switchlights.

CHGR FAIL
Illuminates amber when:
- BATT switch selected ON and AC power is not available.
- Battery charger failure or shutdown has occurred.

ESS Voltmeter Switchlight
- ON:
  - Switch depressed, ON legend illuminated green.
  - Normal operating position.
  - AC VOLTS / FREQ display values monitored at Essential AC bus.
  - DC VOLTS display values monitored at Essential DC bus.
- Off:
  - Switch extended, ON legend extinguished.
  - AC VOLTS / FREQ and DC VOLTS display values monitored using LEFT, AUX or RIGHT voltmeter switchlights.



RIGHT Voltmeter Switchlight
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - AC VOLTS / FREQ display values monitored at Essential AC bus.
  - DC VOLTS display values monitored at Essential DC bus.
- ON:
  - Switch depressed, ON legend illuminated green.
  - AC VOLTS / FREQ and DC VOLTS display values monitored at right converter.

BATTERY 1/2 AMPERES / VOLTS
Displays amperage and voltage of respective battery and battery bus.

TEST
- Depressing and holding switch illuminates all digital displays and switchlight legends.
- In order for switch to operate, battery power (minimum) must be applied.

AUX Voltmeter Switchlight
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - AC VOLTS / FREQ display values monitored at Essential AC bus.
  - DC VOLTS display values monitored at Essential DC bus.
- ON:
  - Switch depressed, ON legend illuminated green.
  - AC VOLTS / FREQ display values monitored at AUX AC contactor for external AC power voltage or APU alternator voltage.
  - DC VOLTS display values monitored at AUX DC contactor for external DC power voltage or TRU voltage.

LEFT Voltmeter Switchlight
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - AC VOLTS / FREQ display values monitored at Essential AC bus.
  - DC VOLTS display values monitored at Essential DC bus.
- ON:
  - Switch depressed, ON legend illuminated green.
  - AC VOLTS / FREQ and DC VOLTS display values monitored at left converter.

26071C00

ELECTRIC POWER
MONITOR Panel
Figure 6 (Sheet 1 of 8)

L AC % / AUX PWR % / R AC %

Displays percentage of total electrical load supported by the AC electrical power system as follows:

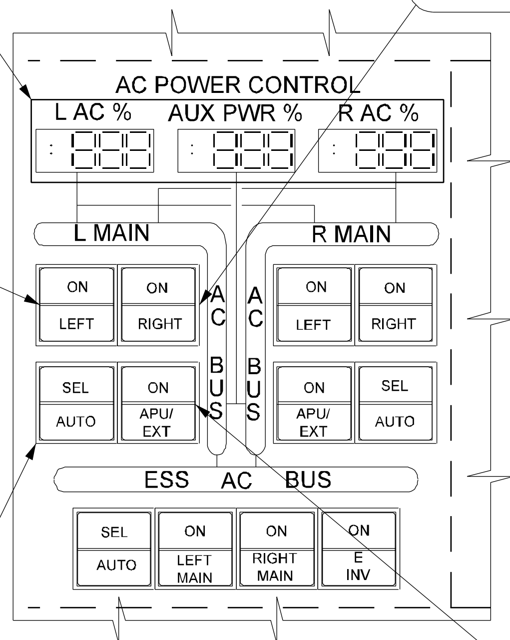
- L AC: Left alternator/converter
- AUX PWR: External AC power or APU alternator
- R AC: Right alternator/converter

RIGHT

- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - BPCU selects power source to be connected to Left Main AC bus.
  - If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when BPCU connects right alternator/converter to Left Main AC bus.
- ON:
  - Switch depressed, ON legend illuminated amber.
  - Right alternator/converter hard-selected for connection to Left Main AC bus.

LEFT

- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - BPCU selects power source to be connected to Left Main AC bus.
  - If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate green when BPCU connects left alternator/converter to Left Main AC bus.
- ON:
  - Switch depressed, ON legend illuminated green.
  - Left alternator/converter hard-selected for connection to Left Main AC bus.



AUTO SEL

- On:
  - Switch depressed, SEL legend illuminated blue.
  - Normal operating position.
  - Automatic selection (auto select) mode.
  - BPCU automatically connects Left Main AC bus to left alternator/converter, APU/external AC power, or right alternator/converter, depending on which is available, in the order listed.
- Off:
  - Switch extended, SEL legend extinguished.
  - Power source connected to Left Main AC bus through LEFT, RIGHT or APU/EXT switches.

APU / EXT

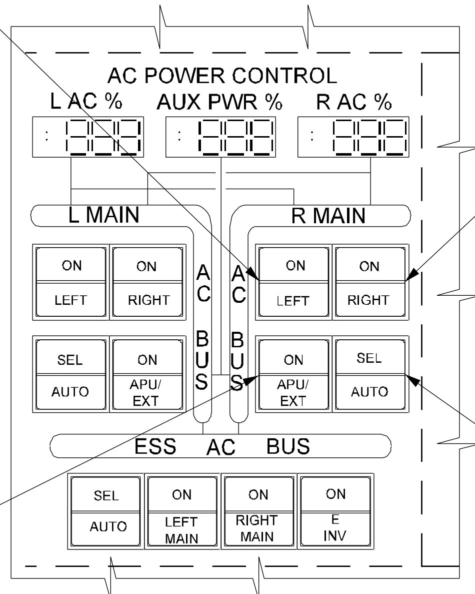
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - BPCU selects power source to be connected to Left Main AC bus.
  - If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when BPCU connects external power or APU alternator to Left Main AC bus.
- ON:
  - Switch depressed, ON legend illuminated amber.
  - External power or APU alternator hard-selected for connection to Left Main AC bus.

26072C00

ELECTRIC POWER
MONITOR Panel
Figure 6 (Sheet 2 of 8)

LEFT
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - BPCU selects power source to be connected to Right Main AC bus.
  - If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when BPCU connects left alternator/converter to Right Main AC bus.
- ON:
  - Switch depressed, ON legend illuminated amber.
  - Left alternator/converter hard-selected for connection to Right Main AC bus.

RIGHT
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - BPCU selects power source to be connected to Right Main AC bus.
  - If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate green when BPCU connects right alternator/converter to Right Main AC bus.
- ON:
  - Switch depressed, ON legend illuminated green.
  - Right alternator/converter hard-selected for connection to Right Main AC bus.



APU / EXT
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - BPCU selects power source to be connected to Right Main AC bus.
  - If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when BPCU connects external power or APU alternator to Right Main AC bus.
- ON:
  - Switch depressed, ON legend illuminated amber.
  - External power or APU alternator hard-selected for connection to Right Main AC bus.

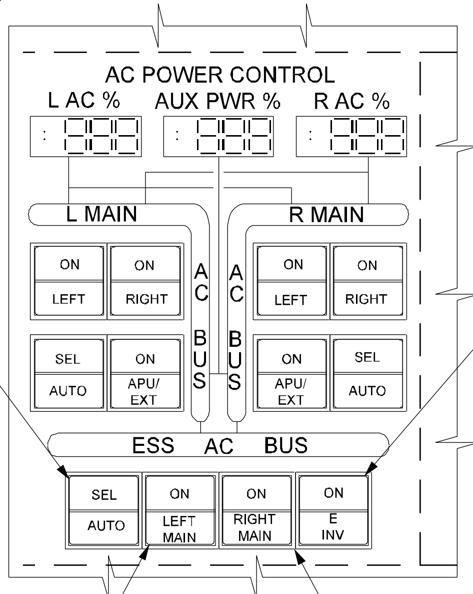
AUTO SEL
- On:
  - Switch depressed, SEL legend illuminated blue.
  - Normal operating position.
  - Automatic selection (auto select) mode.
  - BPCU automatically connects Right Main AC bus to right alternator/converter, APU/external AC power, or left alternator/converter, depending on which is available, in the order listed.
- Off:
  - Switch extended, SEL legend extinguished.
  - Power source connected to Right Main AC bus through LEFT, RIGHT or APU/EXT switches.

26073C00

ELECTRIC POWER
MONITOR Panel
Figure 6 (Sheet 3 of 8)

AUTO SEL
- On:
  - • Switch depressed, SEL legend illuminated blue.
  - • Normal operating position.
  - • Automatic selection (auto select) mode.
  - • ACBPCU automatically connects Essential AC bus to Left Main AC bus, Right Main AC bus, or E-Inverter, depending on which is available, in the order listed.
- Off:
  - • Switch extended, SEL legend extinguished.
  - • Power source connected to Essential AC bus through LEFT MAIN, RIGHT MAIN or E INV switches.

E INV
- Off:
  - • Switch extended, ON legend extinguished.
  - • Normal operating position.
  - • ACBPCU selects power source to be connected to Essential AC bus.
  - • If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when ACBPCU connects E-Inverter to Essential AC bus.
- ON:
  - • Switch depressed, ON legend illuminated amber.
  - • E-Inverter hard-selected for connection to Essential AC bus.



LEFT MAIN
- Off:
  - • Switch extended, ON legend extinguished.
  - • Normal operating position.
  - • ACBPCU selects power source to be connected to Essential AC bus.
  - • If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate green when ACBPCU connects Left Main AC bus to Essential AC bus.
- ON:
  - • Switch depressed, ON legend illuminated green.
  - • Left Main AC bus hard-selected for connection to Essential AC bus.

RIGHT MAIN
- Off:
  - • Switch extended, ON legend extinguished.
  - • Normal operating position.
  - • ACBPCU selects power source to be connected to Essential AC bus.
  - • If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when ACBPCU connects Right Main AC bus to Essential AC bus.
- ON:
  - • Switch depressed, ON legend illuminated amber.
  - • Right Main AC bus hard-selected for connection to Essential AC bus.

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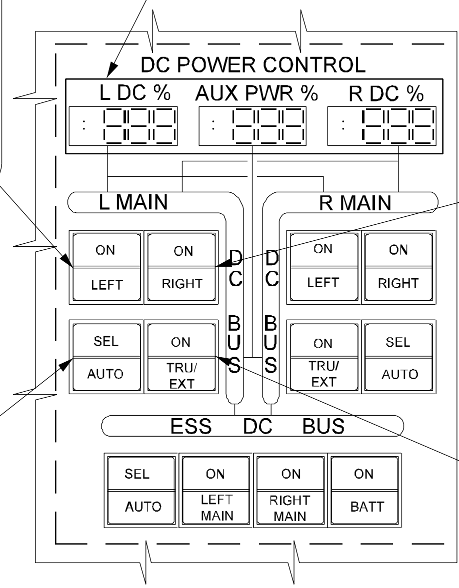
ELECTRIC POWER
MONITOR Panel
Figure 6 (Sheet 4 of 8)

L DC % / AUX PWR % / R AC %
Displays percentage of total electrical load supported by the DC electrical power system as follows:
- L DC: Left alternator/converter
- AUX PWR: External DC power or TRU
- R DC: Right alternator/converter

LEFT
- Off:
  - • • Switch extended, ON legend extinguished.
  - • • Normal operating position.
  - • • BPCU selects power source to be connected to Left Main DC bus.
  - • • If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate green when BPCU connects left alternator/converter to Left Main DC bus.
- ON:
  - • • Switch depressed, ON legend illuminated green.
  - • • Left alternator/converter hard-selected for connection to Left Main DC bus.

RIGHT
- Off:
  - • • Switch extended, ON legend extinguished.
  - • • Normal operating position.
  - • • BPCU selects power source to be connected to Left Main DC bus.
  - • • If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when BPCU connects right alternator/converter to Left Main DC bus.
- ON:
  - • • Switch depressed, ON legend illuminated amber.
  - • • Right alternator/converter hard-selected for connection to Left Main DC bus.



AUTO SEL
- On:
  - • • Switch depressed, SEL legend illuminated blue.
  - • • Normal operating position.
  - • • Automatic selection (auto select) mode.
  - • • BPCU automatically connects Left Main DC bus to left alternator/converter, TRU/external DC power, or right alternator/converter, depending on which is available, in the order listed.
- Off:
  - • • Switch extended, SEL legend extinguished.
  - • • Power source connected to Left Main DC bus through LEFT, RIGHT or TRU/EXT switches.

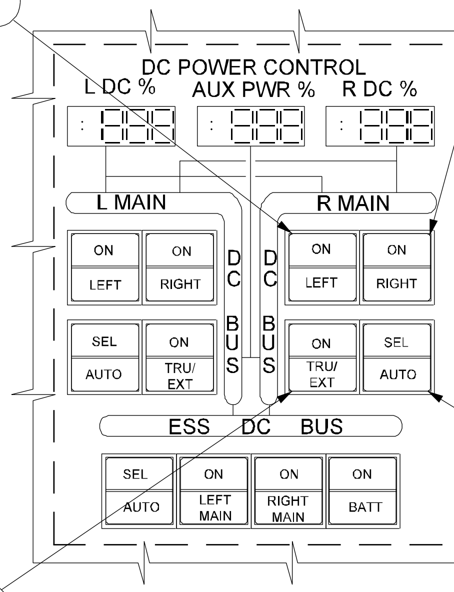
TRU / EXT
- Off:
  - • • Switch extended, ON legend extinguished.
  - • • Normal operating position.
  - • • BPCU selects power source to be connected to Left Main DC bus.
  - • • If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when BPCU connects external power or TRU to Left Main DC bus.
- ON:
  - • • Switch depressed, ON legend illuminated amber.
  - • • External power or TRU hard-selected for connection to Left Main DC bus.

26074C00

ELECTRIC POWER
MONITOR Panel
Figure 6 (Sheet 5 of 8)

LEFT
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - BPCU selects power source to be connected to Right Main DC bus.
  - If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when BPCU connects left alternator/converter to Right Main DC bus.
- ON:
  - Switch depressed, ON legend illuminated amber.
  - Left alternator/converter hard-selected for connection to Right Main DC bus.

RIGHT
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - BPCU selects power source to be connected to Right Main DC bus.
  - If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate green when BPCU connects right alternator/converter to Right Main DC bus.
- ON:
  - Switch depressed, ON legend illuminated green.
  - Right alternator/converter hard-selected for connection to Right Main DC bus.



TRU / EXT
- Off:
  - Switch extended, ON legend extinguished.
  - Normal operating position.
  - BPCU selects power source to be connected to Right Main DC bus.
  - If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when BPCU connects external power or TRU to Right Main DC bus.
- ON:
  - Switch depressed, ON legend illuminated amber.
  - External power or TRU hard-selected for connection to Right Main DC bus.

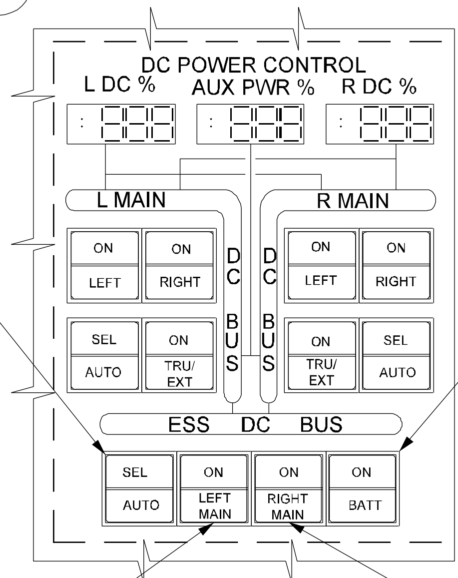
AUTO SEL
- On:
  - Switch depressed, SEL legend illuminated blue.
  - Normal operating position.
  - Automatic selection (auto select) mode.
  - BPCU automatically connects Right Main DC bus to right alternator/converter, TRU/external DC power, or left alternator/converter, depending on which is available, in the order listed.
- Off:
  - Switch extended, SEL legend extinguished.
  - Power source connected to Right Main DC bus through LEFT, RIGHT or TRU/EXT switches.

26075C00

ELECTRIC POWER
MONITOR Panel
Figure 6 (Sheet 6 of 8)

AUTO SEL
- On:
  - • Switch depressed, SEL legend illuminated blue.
  - • Normal operating position.
  - • Automatic selection (auto select) mode.
  - • DCBPCU automatically connects Essential DC bus to Left Main DC bus, Right Main DC bus, or batteries, depending on which is available, in the order listed.
- Off:
  - • Switch extended, SEL legend extinguished.
  - • Power source connected to Essential DC bus through LEFT MAIN, RIGHT MAIN or BATT switches.

BATT
- Off:
  - • Switch extended, ON legend extinguished.
  - • Normal operating position.
  - • DCBPCU selects power source to be connected to Essential DC bus.
  - • If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when DCBPCU connects batteries to Essential DC bus.
- ON:
  - • Switch depressed, ON legend illuminated amber.
  - • Batteries hard-selected for connection to Essential DC bus.



LEFT MAIN
- Off:
  - • Switch extended, ON legend extinguished.
  - • Normal operating position.
  - • DCBPCU selects power source to be connected to Essential DC bus.
  - • If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate green when DCBPCU connects Left Main DC bus to Essential DC bus.
- ON:
  - • Switch depressed, ON legend illuminated green.
  - • Left Main DC bus hard-selected for connection to Essential DC bus.

RIGHT MAIN
- Off:
  - • Switch extended, ON legend extinguished.
  - • Normal operating position.
  - • DCBPCU selects power source to be connected to Essential DC bus.
  - • If AUTO SEL switch is depressed (auto select mode), ON legend will illuminate amber when DCBPCU connects Right Main DC bus to Essential DC bus.
- ON:
  - • Switch depressed, ON legend illuminated amber.
  - • Right Main DC bus hard-selected for connection to Essential DC bus.

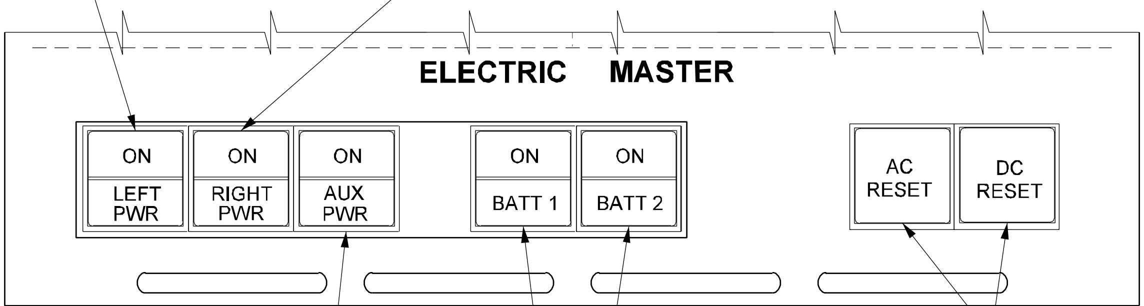
26146C01

ELECTRIC POWER
MONITOR Panel
Figure 6 (Sheet 7 of 8)

LEFT PWR
(Assumes proper EPMP configuration)
- ON (Depressed):
  - • With left alternator operating and left converter supplying power, allows left converter output to Left Main AC bus, Essential AC bus and Left Main DC bus.
  - • ON legend illuminated green.
  - • ON legend extinguishes if left alternator/converter fails with switch depressed.

Off (Extended):
  - • Left converter output to Left Main AC bus and Essential AC bus is inhibited.

RIGHT PWR
(Assumes proper EPMP configuration)
- ON (Depressed):
  - • With right alternator operating and right converter supplying power, allows right converter output to Right Main AC bus and Right Main DC bus.
  - • ON legend illuminated green.
  - • ON legend extinguishes if right alternator/ converter fails with switch depressed.
- Off (Extended):
  - • Right converter output to Right Main AC bus is inhibited.



AC RESET / DC RESET
- Indicates a transient problem, bus fault or logic problem. If problem is a Left Main or Right Main bus fault or overcurrent condition, the BPCU will automatically transfer the bus to another power source.
- Depressing switchlight resets system.
- If switchlight extinguishes, fault has been cleared.
- If switchlight remains illuminated, fault remains.

AUX PWR
(Assumes proper EPMP configuration)
- ON (Depressed):
  - • With external AC power available, external AC power is supplied to the Left Main, Right Main and Essential AC buses. The Left and Right Main DC buses in turn receive power from the TRU.
  - • With external DC power available, external DC power is supplied to the Left Main, Right Main and Essential DC buses. The Essential DC bus in turn supplies power to the Essential AC bus through the E-inverter.
  - • With APU alternator power available, AC power is supplied to Left Main, Right Main and Essential AC buses. The Left and Right Main DC buses in turn receive power from the TRU.
  - • ON legend is illuminated amber.
  - • ON legend extinguishes if power source fails with switch depressed.
- Off (Extended):
  - • Auxiliary power input is inhibited.

BATT 1 / BATT 2
(Assumes no other power source available and EPMP properly configured.)
- ON (Depressed):
  - • Powers the respective battery bus and connects it to Battery Tie bus. Essential DC bus will be powered by Battery Tie bus and Phase A of Essential AC bus will be powered by E-inverter.
  - • Associated battery charger is activated. (If AC power is not available, the associated amber CHGR FAIL light illuminates.)
- Off (Extended):
  - • Battery power is inhibited.

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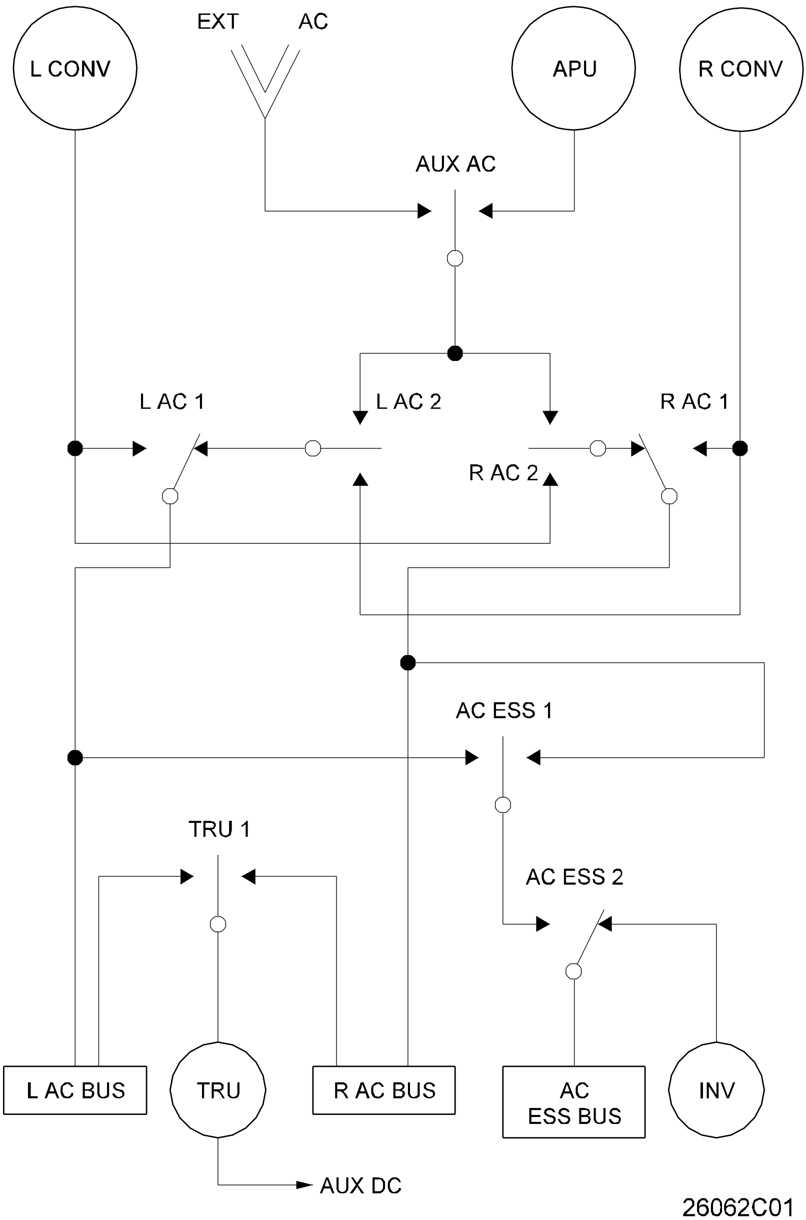
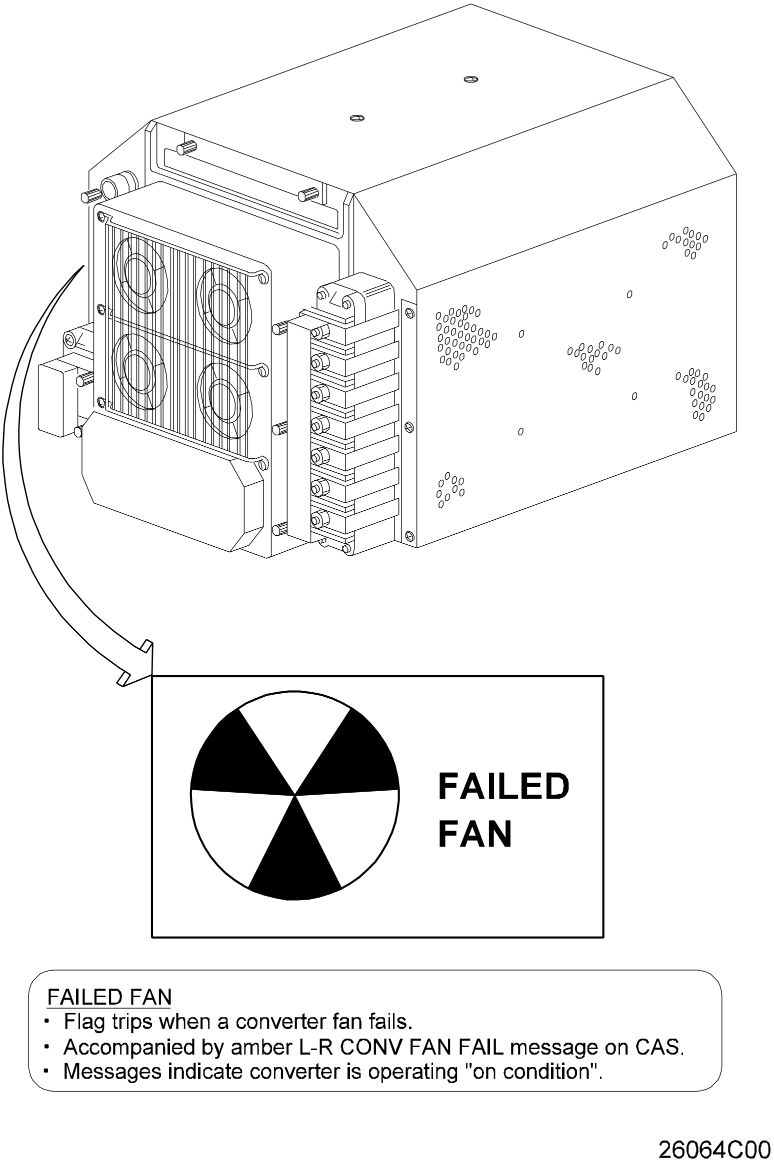
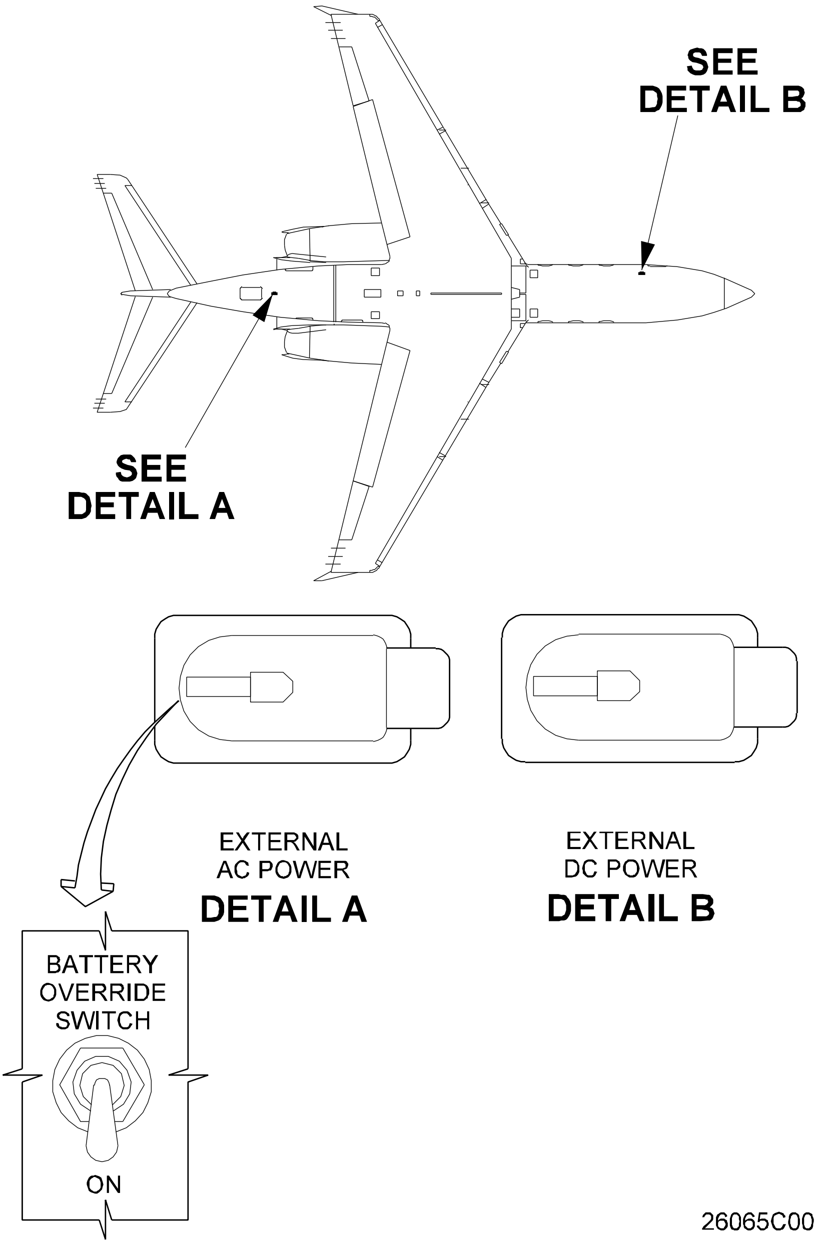
ELECTRIC POWER
MONITOR Panel
Figure 6 (Sheet 8 of 8)



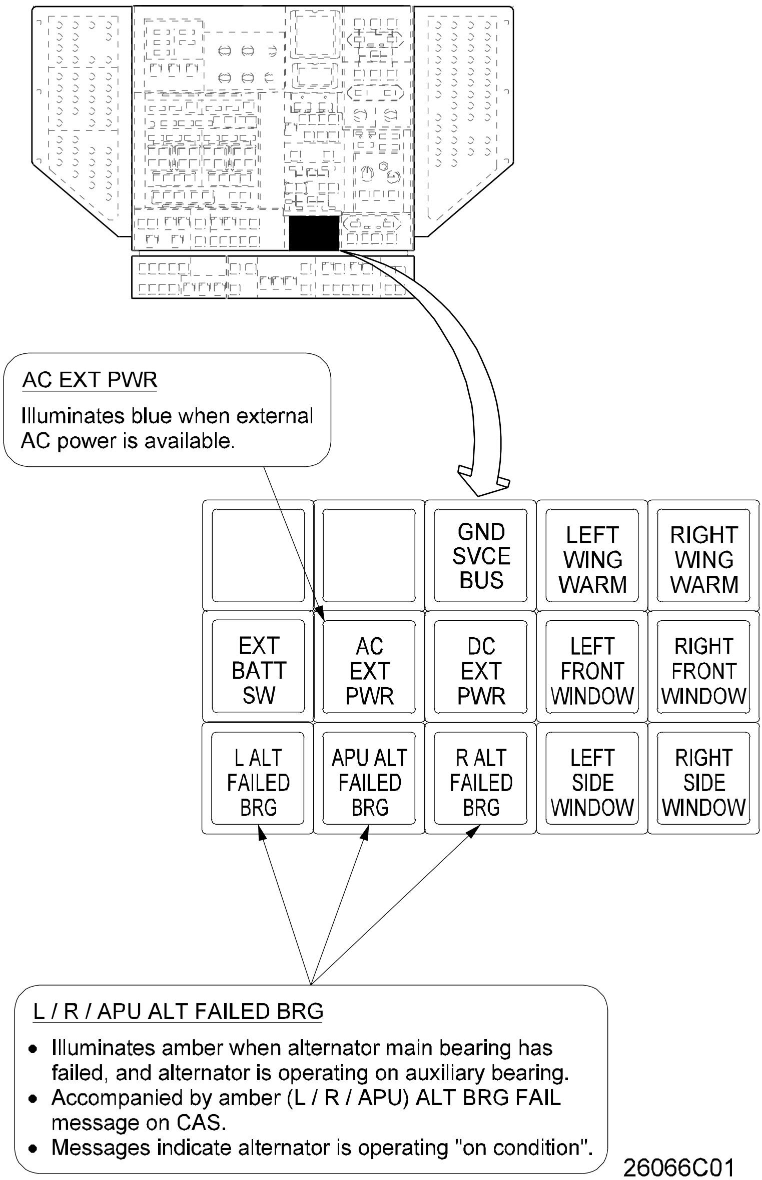
AC Electrical Power System Simplified Block Diagram
Figure 7



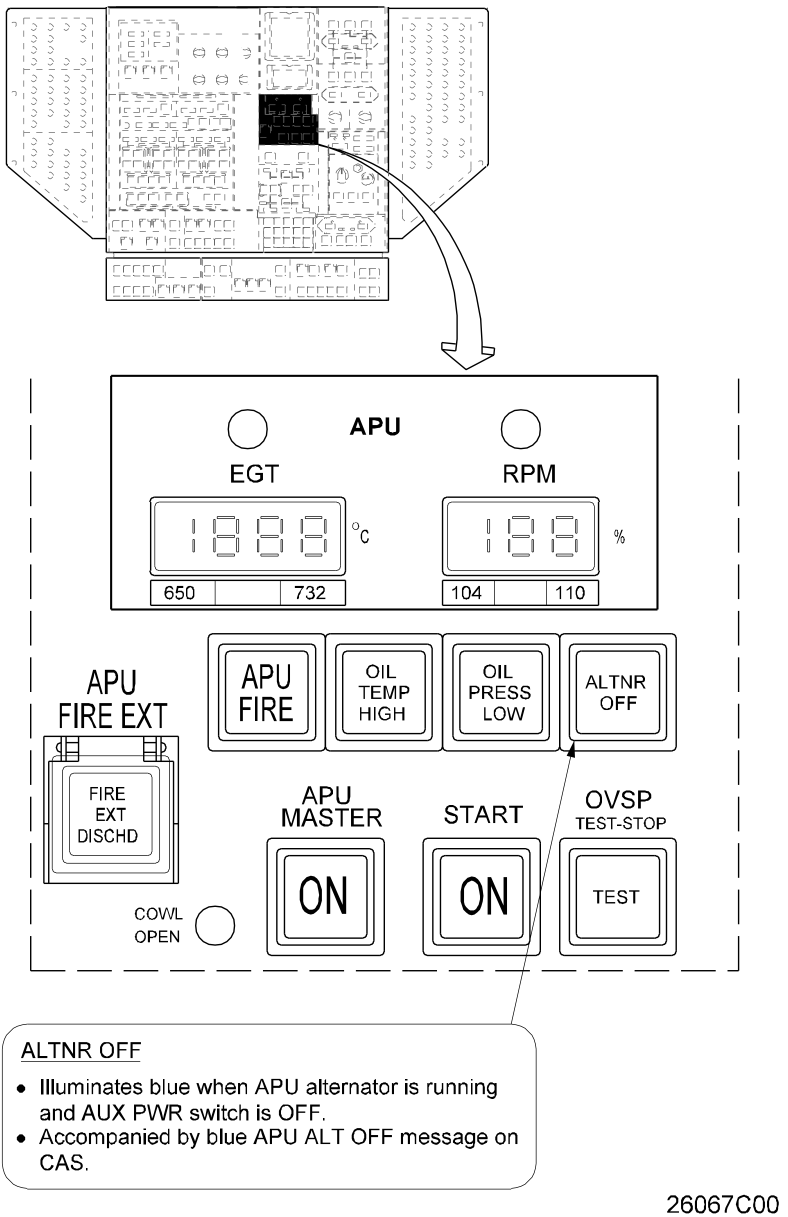
Converter FAILED FAN Flag
Figure 8



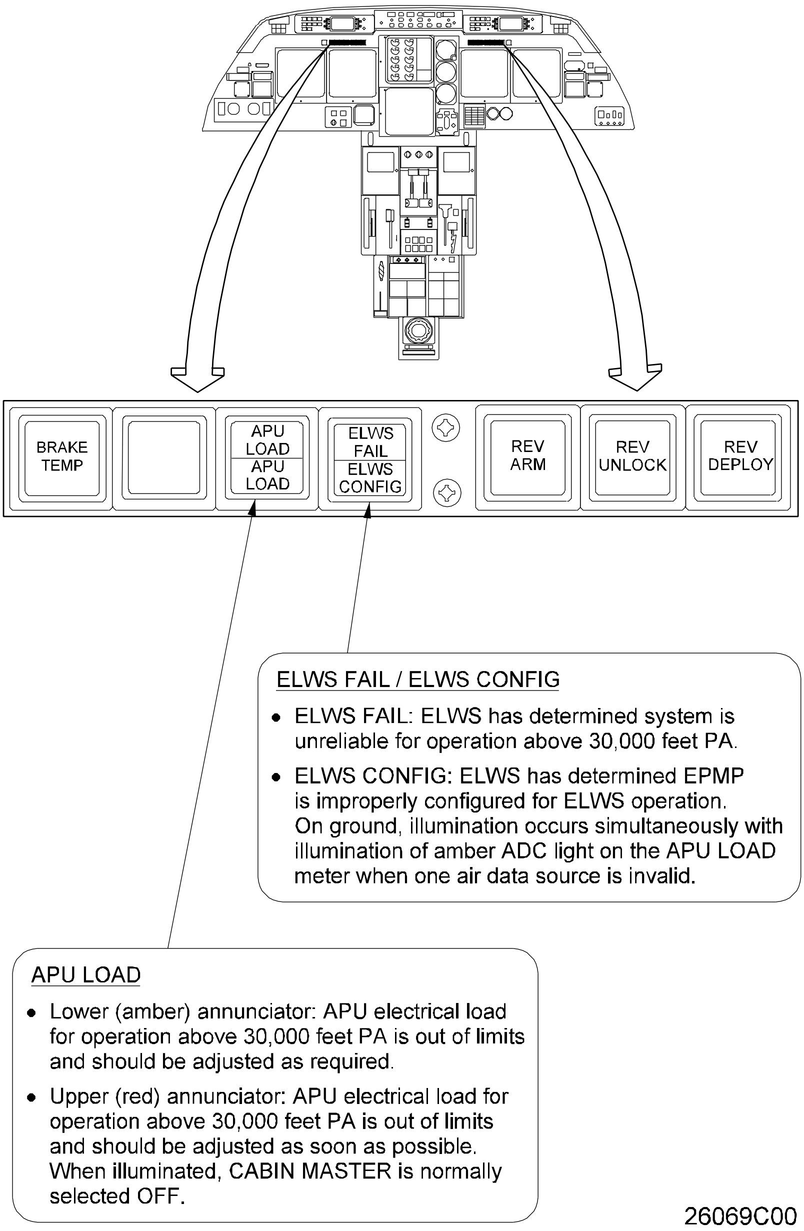
External AC/DC Power Connection Access Panels
Figure 9



Overhead Annunciator Panel
Figure 10



APU Control Panel
Figure 11



APU LOAD/ELWS Annunciators
Figure 12

2A-24-30: DC Electrical Power System

1. General Description:

The GIV DC electrical power system supplies power to the DC electrical buses and feeders for distribution to DC systems and equipment.

Figure 1 shows the entire GIV electrical power system in simplified form; Figure 16 shows only the DC electrical power system in simplified form. Units and components for the DC electrical power system are shown in Figure 2. They are as follows:

- Two Variable-Speed, Constant-Frequency (VSCF) Converters
- Transformer-Rectifier Unit (TRU)
- Two Nickel Cadmium Batteries
- Two Battery Chargers
- DC External Power System
- Standby Electrical Power System
- Emergency Power System (SN 1000 through 1466)
- Emergency Power System (SN 1467 and Subsequent)
- DC Bus System
- Ground Service Bus (SN 1455 and Subsequent)
- DC Bus Power Control Unit (DCBPCU)
- Power Distribution Box (PDB)

Similar to the AC electrical power system, a DC Bus Power Control Unit (DCBPCU) controls DC power application from the various power sources to the DC buses. The DC bus system consists of the Left Main DC bus, Right Main DC bus and an Essential DC bus.

2. Description of Subsystems, Units and Components:

A. Converters:

Two alternator-powered converters are the primary source of DC power for the aircraft electrical system. Each converter transforms alternator output to 28 VDC, 250 ampere current. DC output voltage from the converter is maintained at 28 (±5) volts at the point-of-regulation from zero load to full load.

During normal operation the two converters share the DC electrical system loads with the left converter supplying power to the Left Main and Essential DC buses, and the right converter supplying power to the Right Main DC bus. If one converter should fail and the Transformer-Rectifier Unit (TRU) is failed, the remaining converter assumes all DC loads, although some load shedding may be necessary.

Both converters are located in the tail compartment. For more information about the converters, see Section 2A-24-20, AC Electrical Power System.

B. Transformer-Rectifier Unit:

A 300 ampere-rated DC Transformer-Rectifier Unit (TRU) provides 28 VDC power to the Left and Right Main DC buses during ground operation. If a converter channel fails in flight, the TRU, powered by three-phase power from the Left or Right Main AC bus, can supply 300 amperes of DC power to the Left and Right Main DC buses. The DCBPCU provides overcurrent

sensing and protection for the TRU. Both the TRU and DCBPCU are located on the right avionics rack.

Failure of a TRU is accompanied by an amber TRU FAIL message on the Crew Alerting System (CAS). If TRU temperature exceeds 374° F (190° C), a thermal switch on the TRU will close, causing an amber TRU HOT message to be displayed on CAS.

C. Batteries:

(See Figure 6 Sheet 8, Figure 13, Figure 14, Figure 15 and Figure 17.)

Two 24 VDC, 40 ampere-hour rated, nickel-cadmium (nicad) batteries provide supplementary DC power to the aircraft. Each battery is installed inside its own fireproof stainless steel enclosure in the aft equipment compartment. Access to the batteries is from outside the aircraft through a removable panel; this same panel having two smaller access doors for visual inspection and quick access to the terminal connectors and sense/control cables.

The 20 internal cells inside each battery are designed to prevent spillage in any position and will vent any gases to the battery casing. Each casing in turn vents to the atmosphere through its own venting system.

A cooling and circulation fan is installed in each battery compartment. Both fans receive power from an inverter unit through a thermal switch in the battery compartment. The inverters receive power from the batteries when either or both battery switches are on or from the battery chargers when they are receiving power from the Left and Right Main AC buses. When battery compartment temperature reaches 90°F (32° C), the thermal switch closes and the cooling and circulation fan operates. As temperature drops to 75° F (24° C), the thermal switch opens and the fan stops.

The batteries are controlled from the cockpit by BATT 1 and BATT 2 switchlights, located on the EPMP, on the cockpit overhead panel. From outside the aircraft, the batteries are controlled by the OUTSIDE BATTERY switch, located below the battery access panels. They may also be controlled from an EXTERNAL BATTERY switch, located in the forward external switch panel, if installed. Selection of any battery switch connects the batteries to the Battery Tie bus and their respective battery bus.

Selection of the OUTSIDE BATTERY switch (or EXTERNAL BATTERY switch, if installed) to ON provides battery power to operate the auxiliary hydraulic pump, in order to close the main entrance door, or operate the landing gear wheel well doors. It also provides power to operate the engine oil replenisher pump or start the APU. Whenever either of the external switches is selected ON, a blue EXT BATT SW annunciator on the cockpit overhead panel will illuminate. If CAS is operating, a blue EXT BATT SWITCH ON message will be displayed. In addition, the lower anti-collision beacon is illuminated as an exterior visual reminder that the batteries are on.

From inside the aircraft, the batteries are controlled by the BATT 1 and BATT 2 switchlights. Both switches normally remain selected ON during all phases of flight, capable of providing power to the Essential DC bus if no other power source is available (DC external power, TRU or converter). If CAS is operating, selection of either BATT switchlight to OFF causes an amber EPMP BATT SW OFF message to be displayed.

Selection of the BATT 1 and/or BATT 2 switchlight to ON powers the respective battery bus and connects it to the Battery Tie bus. In addition, the associated battery charger is activated. If AC power is not available, the associated amber CHGR FAIL light illuminates on the EPMP (Figure 6). If CAS is operating, an amber BATT 1-2 CHGR FAIL message is displayed. Aircraft SN 1465 and subsequent and aircraft with ASC 54 incorporated have individual control over batteries and chargers.

As mentioned previously, power from the Battery Tie bus can be supplied to the Essential DC bus, if necessary. Automatic connection will take place provided the following conditions are present:

- Left and Right Main AC buses unpowered
- Left and Right Main DC buses unpowered
- No other power source available
- All EPMP AUTO/SEL switchlights in AUTO

In this configuration, the Essential DC bus will be powered by the Battery Tie bus and Phase A of the Essential AC bus will be powered by the E-inverter.

In situations where the EPMP is manually configured, hard selection of the ESS DC BUS BATT switch (Figure 6) to ON will also connect the Battery Tie bus to the Essential DC bus.

In situations where the main batteries are the only source of power to the Essential DC bus, an amber BATT ON BUS caution message will be displayed on CAS. On SPZ-8000 equipped airplanes with ASC 415 incorporated, an amber DC ESS ON BATT caution message will be displayed on CAS.

D. Battery Chargers:

(See Figure 1, Figure 6, Figure 13 and Figure 14.)

Each battery has its own dedicated charger in the aft equipment compartment. The No. 1 battery charger is powered by the Left Main AC bus; the No. 2 battery charger is powered by the Right Main AC bus. The BATT 1 and/or BATT 2 switchlights must be selected to ON in order for the chargers to operate. On aircraft SN 1465 and subsequent and aircraft with ASC 54 incorporated, the No. 1 battery charger is controlled through the BATT 1 switchlight; the No. 2 battery charger is controlled through the BATT 2 switchlight. The battery chargers have two modes of operation: charge mode and Transformer-Rectifer (TR) mode.

In the charge mode (Main AC buses powered and BATT switchlights ON), each charger provides a constant 38 ampere charge current at a approximately 28 to 36 volts, depending on battery state-of-charge and load resistance, stabilizing at 27.75 volts. At this point, the charge is complete and the battery ammeter should read 0 (±003). If battery voltage drops to 23 volts, the charge cycle resumes. A normal battery should completely recharge within approximately 75 minutes from a fully discharged state.

Battery voltage must be a minimum of 4 volts in order for the charger to initiate a charge cycle. Automatic shutdown of the chargers will take place should any of the following occur:

- The battery terminal connector is disconnected with power applied

- The battery sense and control cable is not connected
- Charger input voltage over 134 VAC or under 94 VAC
- Charger input voltage absent
- Charger output current exceeds 38 amperes (charger mode) or 64 amperes (TR mode)
- Battery temperature reaches 145° F (63° C)

Shutdown or failure of the battery chargers is annunciated to the flight crew by an associated amber CHGR FAIL light illuminating on the EPMP. If CAS is operating, an amber BATT 1-2 CHGR FAIL message is also displayed.

The TR mode of battery charger operation is initiated if the Main AC buses are powered, the BATT switchlights are ON and any one the following events occur:

- The Auxiliary Hydraulic Pump (AUX PUMP) is selected ON (any method)
- The APU START switch is depressed
- Selection of the ESS DC BUS BATT switch to ON (automatically or manually)

In the TR mode, the battery chargers are each capable of supplying a continuous 50 amperes of DC power to the Essential DC bus in order to meet demands. If demand exceeds the total 100 amperes supplied by the chargers, the batteries themselves will begin to share the load and a discharge will be reflected on the battery ammeters.

E. DC External Power System:

The DC power system includes provisions for connecting an external source of regulated 28 VDC (29 VDC maximum) power. An external DC power receptacle is located on the lower right forward fuselage (Figure 9). Working in conjunction with the DCBPCU, the external DC receptacle has provisions to protect aircraft electrical circuits from overvoltage, undervoltage and reversed polarity. Detected faults or input voltage greater than 29 VDC will cause external DC power to be automatically disconnected.

With the external DC power source connected and operating, a blue DC EXT PWR indicator on the cockpit overhead panel (Figure 17) will illuminate. If CAS is operating, an blue DC EXT POWER message is also displayed. With all EPMP AUTO/SEL switchlights in AUTO, selection of the AUX PWR switch (Figure 6, Sheet 8) provides external DC power to the Left Main, Right Main and Essential DC buses. The Essential DC bus in turn supplies power to the Essential AC bus through the E-inverter.

F. Standby Electrical Power System:

The standby electrical power system consists of a hydraulic motor-driven generator and a Generator Control Unit (GCU). When no AC power is available from either main AC bus or the auxiliary power source, Phase A of the Essential AC bus and the Essential DC bus can receive power from this system.

With the motor-generator operating, selection of the E INVERT switchlight (STANDBY ELECTRICAL POWER control panel, Figure 4) to ON supplies 50 amperes of filtered 28 VDC power to the E-inverter. The E-inverter in

turn supplies AC power for Phase A of the Essential AC bus. Selection of the TRU switchlight to ON supplies 5 kVA of 115 VAC, 400 Hz power from the motor-generator to the TRU. The TRU in turn supplies DC power to the Essential DC bus.

For more information about the standby electrical power system, see Section 2A-24-20, AC Electrical Power System.

G. Emergency Power System (SN 1000 through 1466):

The emergency power system provides electrical power to the emergency buses to power flight instruments and other equipment essential for flight safety in the event of total loss of electrical power. The amount of time the emergency power system can operate is dependant upon the load placed on the emergency buses, however, a minimum of forty (40) minutes can be expected.

The emergency power system consists of two emergency battery units, each composed of a power module, battery and control module. The power module contains a battery charger, heater switching circuits and charger fault circuits. The battery module consists of a battery, battery heater and temperature sensor. The control module has circuitry that controls both charger and heater operation, provides fault monitoring and drives the output signals. The forward emergency battery unit (No. 1) is located in the right avionics rack and the aft emergency battery unit (No. 2) is located in the tail compartment.

When the aircraft electrical system is operating normally, the Essential DC bus supplies power to the emergency battery charging system. If both Main DC buses are lost, charging current is inhibited.

If the Essential DC bus is lost, the emergency batteries will automatically power the Emergency buses, wing leading edge evacuation lighting and external overwing evacuation lighting, provided the EMERG POWER ARM switchlight, located on the cockpit overhead panel (Figure 18), was selected to ARM. ARM selection is verified by the switchlight's 1NOT/2NOT amber legends being extinguished. (The adjacent 1ON/2ON and 1OFF/2OFF amber legends are also extinguished.) An impact switch will automatically activate the emergency power system, regardless of any switchlight's position, when it senses a 2.5G or greater force.

Manual selection of the EMERG POWER ON switchlight will switch on the emergency batteries at any time. ON selection is verified by the switchlight's 1ON/2ON amber legends being illuminated. Manual selection of the EMERG POWER OFF switchlight will switch off the emergency batteries, but the Essential DC bus must be powered during the shutdown. OFF selection is verified by the switchlight's 1OFF/2OFF amber legends being illuminated.

When emergency battery temperature is below 60°F (16°C), trickle charge mode is enabled and electric heaters warm the batteries. When battery temperature reaches 60°F (16°C), the heaters shut off automatically and the main charge mode is enabled.

Emergency batteries No.1 and 2 are connected to Data Acquisition Units (DAUs) No. 1 and 2, respectively. If an emergency battery discharges completely, a blue E BATT 1-2 DISCH message is displayed on CAS. If an emergency battery fails, a blue E BATT 1-2 FAIL message is displayed on

CAS.

Two other emergency batteries are installed on the aircraft: IRS-1 and IRS-2. They are located in the left and right avionics racks, respectively, and are identical to the forward and aft emergency batteries. Each IRS battery supplies power to the associated IRS and memory keep-alive for the associated navigation computer.

A table listing the components powered by the emergency batteries is shown in the **Controls and Indications** topic of this section.

H. Emergency Power System (SN 1467 and Subsequent):

The emergency power system provides electrical power to the emergency buses to power flight instruments, interior emergency lighting and other equipment essential for flight safety in the event of total loss of electrical power.

The emergency power system consists of four emergency battery units, each composed of a power module, battery module and control module. The power module contains a battery charger, heater switching circuits and charger fault circuits. The battery module consists of a battery, battery heater and temperature sensor. The control module has circuitry that controls both charger and heater operation, provides fault monitoring and drives the output signals. The forward emergency battery units (No. 1 and 3) are located in the right avionics rack and the aft emergency battery units (No. 2 and 4) are located in the tail compartment. When operating, each battery is capable of delivering 200 watts for 15 minutes and 6 amp-hours if fully charged.

When the aircraft electrical system is operating normally, the Essential DC bus supplies power to the emergency battery charging system. If both Main DC buses are lost, charging current is inhibited.

If the Essential DC bus is lost, emergency batteries No. 1 and 2 will automatically power the Emergency buses, wing leading edge evacuation lighting and external overwing evacuation lighting. Emergency batteries No. 3 and 4 will automatically power the interior emergency lighting and main entrance door emergency lighting (if the main entrance door is open). This automatic powering of buses and components is predicated on the EMERG POWER ARM switchlight, located on the cockpit overhead panel (Figure 19), being selected to ARM. ARM selection is verified by the switchlight's 1NOT/2NOT/3NOT/4NOT amber legends being extinguished. (The adjacent 1ON/2ON/3ON/4ON and 1OFF/2OFF/3OFF/4OFF amber legends are also extinguished.) An impact switch will automatically activate the emergency power system, regardless of any switchlight's position, when it senses a 2.5G or greater force.

Manual selection of the EMERG POWER ON switchlight will switch on the emergency batteries at any time. ON selection is verified by the switchlight's 1ON/2ON/3ON/4ON amber legends being illuminated. Manual selection of the EMERG POWER OFF switchlight will switch off the emergency batteries, but the Essential DC bus must be powered during the shutdown. OFF selection is verified by the switchlight's 1OFF/2OFF/3OFF/4OFF amber legends being illuminated.

When emergency battery temperature is below 60°F (16°C), trickle charge mode is enabled and electric heaters warm the batteries. When battery

temperature reaches 60°F (16°C), the heaters shut off automatically and the main charge mode is enabled.

Emergency batteries No.1 and 2 are connected to Data Acquisition Units (DAUs) No. 1 and 2, respectively. Emergency batteries No.3 and 4 are connected to Fault Warning Computers (FWCs) No. 1 and 2, respectively. If an emergency battery discharges completely, a blue E BATT 1-2-3-4 DISCH message is displayed on CAS. If an emergency battery fails, a blue E BATT 1-2-3-4 FAIL message is displayed on CAS.

Two other emergency batteries are installed on the aircraft: IRS-1 and IRS-2. They are located in the left and right avionics racks, respectively, and are identical to the forward and aft emergency batteries. Each IRS battery supplies power to the associated IRS and memory keep-alive for the associated navigation computer.

A table listing the components powered by the emergency batteries is shown in the **Controls and Indications** topic of this section.

I. DC Bus System:

DC power control and distribution to the DC buses takes place through the EPMP, PDB and DCBPCU. The bus system then distributes power to the various aircraft systems. The DC bus system consists of the Left Main DC bus, Right Main DC bus and Essential DC bus.

Normal system load consists of the left converter powering the Left Main and Essential DC buses, and the right converter powering the Right Main DC bus. Transfer relays allow either the Left or Right Main DC bus to receive power from either the left or right converters. In addition, either or both main DC buses can receive power from the TRU or, if on the ground, external DC power. The Essential DC bus in turn can receive power from either of the main DC buses, the Battery Tie bus or the TRU.

An automatic switching and priority system selects and connects the DC buses to the appropriate power source in a set order. If more than one power source is available, the system automatically assigns power to the:

(1) Left Main DC bus from:

  (a) Left converter

  (b) TRU or external DC power

  (c) Right converter

(2) Right Main DC bus from:

  (a) Right converter

  (b) TRU or external DC power

  (c) Left converter

(3) Essential DC bus from:

  (a) Left Main DC bus

  (b) Right Main DC bus

  (c) Batteries

Circuit breakers and overload sensors protect the DC bus distribution system from shorts and overloads. If a circuit breaker's rated amperage is exceeded, the circuit breaker will automatically open to protect the component. The circuit breakers may also be manually opened as

necessary by the flight crew. Regardless of how circuit breakers are opened, they must always be manually closed.

Overload sensors are used in large current applications to disconnect a system when an overload condition exists. The sensors are operated thermally and although the sensor itself is self-resetting, the circuit breakers that the sensors may open as a result of overload must be manually closed.

J. Ground Service Bus (SN 1455 and Subsequent):

On airplanes SN 1455 and subsequent, a ground service bus is incorporated to provide DC power to equipment required to perform minimal servicing of the aircraft while on the ground. Use of the ground service bus prevents the unnecessary powering of avionics equipment. In the production standard configuration, only the wheel well lights, utility lights and service lights receive power from the ground service bus. Other items may be connected as desired by the operator during completion and outfitting.

Selection of the ground service bus on or off is accomplished using the GROUND SERVICE switch, located on the forward external switch panel. See Figure 15. Whenever the GROUND SERVICE switch is selected to ON, the ground service bus receives power from BATT 2. Other airplane power sources may be connected as desired by the operator during completion and outfitting.

A GND SVCE BUS annunciator, located in the cockpit overhead annunciator panel and shown in Figure 17, illuminates amber when the ground service bus is powered and airplane electrical power (battery power minimum) is ON.

K. DC Bus Power Control Unit:

The DC Bus Power Control Unit (DCBPCU), located in the right avionics rack, controls the connection of the DC power sources to the DC buses. By providing automatic control capability over the DC power relays and contactors, it controls the DC buses and batteries by switching DC bus power to an alternate source should a malfunction occur. The DCBPCU also provides power supply/voltage regulation, external power monitoring, bus feeder fault protection, and manual or automatic control capabilities.

The DCBPCU also protects the DC buses from external DC overvoltage and undervoltage, TRU undervoltage and overcurrent, Left, Right and Essential DC bus undervoltage and overcurrent, and bus fault protection on the main and essential DC bus feeders.

L. Power Distribution Box:

The Power Distribution Box (PDB), located in the right avionics rack, is the load center of the electrical system. It serves as the terminal for all input power and output to the buses and other loads in the aircraft. Power switching to the DC buses is accomplished through relays within the PDB that are controlled by the DCBPCU and converters. The PDB also provides switching and control for the AC electrical system.

3. DC Electrical Power System Operation:

(See Figure 6.)

The flight crew controls the electrical power system through the EPMP. It consists

of 13 digital displays and 33 switchlights that allow the selection of electrical power sources and display of voltages, amperages, frequencies and percentages of total load.

When normally configured (L MAIN, R MAIN and ESS DC BUS switches selected to AUTO; BATT 1 and/or BATT 2 switch(es) selected ON), the EPMP allows automatic operation and selection of electrical power sources in order to minimize flight crew workload. Manual operation is always available, however, in order to hard-select or isolate power sources.

With an external DC power source available and the ELECTRIC MASTER AUX PWR switch selected ON, external DC power is routed through the PDB to the Left Main, Right Main and Essential DC buses.

With the alternators operating and the converters supplying power, selection of the ELECTRIC MASTER LEFT PWR and RIGHT PWR switches to ON allows left converter output to the Left Main and Essential DC buses and right converter output to the Right Main DC bus.

If the APU is the only source of power and the ELECTRIC MASTER AUX PWR switch is selected ON, the APU alternator powers the Left Main, Right Main and Essential AC buses. The Left Main, Right Main and Essential DC buses in turn receive power from the TRU.

The Essential DC bus provides power to equipment essential for safe flight. Normally, the Left Main DC bus supplies the Essential DC bus but should it or the Right Main DC bus not be available (and the Left Main, Right Main and Essential AC buses are also not available), the main batteries are capable of powering the bus through the Battery Tie bus. In this configuration, the Essential DC bus also powers Phase A of the Essential AC bus through the E-inverter.

Illumination of the DC RESET amber switchlight indicates a transient problem, bus fault or logic problem. If the problem is a Left Main or Right Main DC bus fault or overcurrent condition, depressing the switchlight resets the system. If the switchlight extinguishes, the fault has been cleared. If it remains illuminated, the fault remains. If the problem is an Essential DC bus fault or overcurrent condition, the DCBPCU will automatically attempt to transfer the bus to another power source.

4. Controls and Indications:

A. Circuit Breakers:

Circuit breakers controlling the DC bus distribution system are shown in bold print in the tables that follow. Circuit breakers receiving power from the controlling circuit breaker are shown immediately following that circuit breaker.

Example: The WARN LTS PWR #1 circuit breaker receives power from the PILOT 1 section of the Essential DC bus. Thus the PILOT 1 circuit breaker is the controlling circuit breaker.

Essential DC Bus Pilot 1 CBs
Table 4

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT 1	**PDB**	**DC ESS**
WARN LTS PWR #1	P	F-1

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT 1	**PDB**	**DC ESS**
WARN LTS PWR #2	P	F-2
WARN LTS PWR #3	P	F-3
WARN LTS PWR #4	P	F-4
WARN LTS PWR #5	P	F-5
WARN LTS PWR #6	P	G-1
WARN LTS PWR #7	P	G-2
WHEEL WELL LTS	P	F-6
L FIRE DET LOOP B	P	G-6
L SEC LOCK	P	G-7
L T/REV CONTROL	P	I-7
ENGINE START	P	K-6
L #1 IGN	P	I-8
R #1 IGN	P	I-9
APU FIRE EXT	P	I-10
AFT EMERG BATT	P	I-13
FWD IMPACT	P	J-13
MAIN PUMP CONT	PO	A-1
L FUEL S/O	PO	A-2
L FUELING S/O	PO	A-3
FUEL X FLOW V	PO	D-2
FUEL PUMP IND	PO	A-4
EPMP SW PWR #1	PO	A-5
BATT CHGR CONT	PO	A-6
BATT AMMETER	PO	A-7
AC EXT PWR	PO	A-8
L BLEED AIR	PO	A-10
R BLEED AIR IND	PO	C-12
L AIR COND	PO	C-13
CKPT/CABIN TEMP IND	PO	A-11
CABIN PRESS IND	PO	A-12
BOT A/C LT GND OPER (No ASC 10)	P	E-6
AIR INLET DOOR	PO	C-11
SGL PACK	PO	C-14
ESS DC VM	P	D-13
CKPT TEMP CONT	PO	B-13
AFT EMER BAT	P	I-13
ANTISKID OUTBD (1183 & 1214 & Subs)	PO	A-2
ELWS 1	PO	A-9

Essential DC Bus Pilot 2 CBs
Table 5

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT 2	**PDB**	**DC ESS**
#2 STANDBY WARN PWR	P	B-1
R CKPT ANN LTS PWR	P	B-2
PED ANN LTS PWR	P	D-2
ANN LTS CONT STBY	P	B-3
WARN LTS PWR #8	P	G-3
WARN LTS PWR #10	P	G-5
WARN LTS PWR #11	P	H-3
WARN LTS PWR #12	P	H-4
WARN LTS PWR #13	P	H-5
WARN LTS PWR #14	P	I-4
WARN LTS PWR #15	P	I-5
WARN LTS TEST	P	J-5
SPD BRAKE/FLAP ALARM	P	D-4
FLAP/STAB WARN	P	D-3
FIRE BELL	P	B-4
DOME LT	P	B-6
UTILITY LTS	P	D-9
R FIRE DET LOOP B	P	H-6
R SEC LOCK	P	H-7
R ENG O'HT	P	H-9
FIRE EXT SHOT #2	P	H-8
APU START	P	H-10
STBY PWR CONT	P	H-12
FWD EMER BATT	P	H-13
AFT IMPACT	P	K-13
R FUEL S/O	PO	B-2
R FUELING S/O	PO	B-3
FUEL INTERTANK V	PO	B-4
FUEL LOW LEVEL	PO	B-1
EPMP SW PWR #2	PO	B-5
BATT CHGR FAIL ANN	PO	B-6
BACKUP CONT PWR	PO	B-7
R BLEED AIR	PO	B-10
L BLEED AIR IND	PO	B-12
CABIN PRESS 28V	P	B-11
OVHD ANN LTS PWR #4	P	C-4
OVHD ANN LTS PWR #5	P	C-5
R AIR COND	PO	D-13
OVHD PNL PRI (1040 & Subs)	P	B-9
CABIN TEMP CONT	PO	D-14

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT 2	**PDB**	**DC ESS**
FWD EMER BATT	P	H-13
ANTISKID INBD	PO	B-2
ELWS 2	PO	B-9

Essential DC Bus Pilot 3 CBs
Table 6

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT 3	**PDB**	**DC ESS**
#1 STANDBY WARN PWR	P	A-1
L CKPT ANN LTS PWR	P	A-2
ANN LTS CONT MAIN	P	A-3
OVHD ANN LTS PWR #1	P	C-1
OVHD ANN LTS PWR #2	P	C-2
OVHD ANN LTS PWR #3	P	C-3
LDG WARN HORN	P	E-4
TONE WARN #1	P	A-5
APU FIRE WARN	P	A-4
FLOOD LITES OVERRIDE	P	A-6
L FIRE DET LOOP A	P	I-6
R FIRE DET LOOP A	P	J-6
L T/R EMER STOW	P	L-7
R T/R EMER STOW	P	K-7
ENGINE OILER	P	L-9
L ENG O'HT	P	G-9
L #2 IGN	P	J-8
R #2 IGN	P	J-9
FIRE EXT SHOT #1	P	G-8
APU PWR #3	P	L-10
R T/REV CONTROL	P	J-7
L FLT PNL/GSHLD (1040 & Subs)	P	A-7

Essential DC Bus Copilot 1 CBs
Table 7

Circuit Breaker Name	CB Panel	Location (PDB Section)
COPILOT 1	**PDB**	**DC ESS**
BUS CONT #1	CP	K-2
BUS CONT #3	CP	M-2
SYM GEN #1	CP	K-3
DISPLAY UNIT #1	CP	E-6
IRU #1 DC SEC	CP	J-5
E DDRMI #1	CP	H-5
L PITOT HT CONT	CP	L-12
STBY PITOT HT CONT	CP	L-13
AOA PRB HTR #1	CP	L-14
FWC #1	CP	A-13
DAU #1A	CP	A-14
MANUAL FLAP CONT	CPO	A-1
COMB HYD QTY	CPO	A-4
COMB HYD PRESS	CPO	A-3
UTILITY HYD PUMP OFF	CPO	C-3
FLT HYD PRESS	CPO	C-4
L HYD S/O	CPO	A-5
STAB AUG SERVO #1	CPO	A-6
FGC #1	CPO	A-7
STALL BARRIER #1	CPO	A-9
STALL BARR VALVE #1	CPO	A-12
STALL BARR DUMP VALVE	CPO	A-8
SHAKER #1	CPO	A-10
STALL WARN CMPTR #1	CPO	A-11
L NUTCRACKER	CPO	A-13
CKPT VOICE RECORDER	CP	D-13

Essential DC Bus Copilot 2 CBs
Table 8

Circuit Breaker Name	CB Panel	Location (PDB Section)
COPILOT 2	**PDB**	**DC ESS**
DISPLAY MASTER #2	CP	J-2
BUS CONT #2	CP	L-2
DADC #1	CP	F-3
DISPLAY UNIT #3	CP	G-6
DBDI #2	CP	G-4
APPLIED BRAKE PRESS	CPO	D-6
NAV/DME CONT #1	CP	D-9
R COWL A/I PRESS	CP	K-12
R COWL ANTI-ICE	CP	K-13
R WING ANTI-ICE	CP	K-14
CVR SHUTDN	CP	E-13
DAU #2B	CP	D-14
FLAP CONT	CPO	B-1
BCS CHL #2	CPO	B-2
UTILITY HYD PRESS	CPO	B-3
FLT HYD QTY	CPO	B-4
AUX HYD PRESS	CPO	D-4
WHEEL BRK ACCUM PRESS	CPO	D-3
R HYD S/O	CPO	B-5
DOOR CONT/WARN	CPO	B-8
A/P SERVO #1	CPO	C-8
NUTCRACKER	CPO	B-13
STEER BY WIRE #1	CPO	C-12
FLT GDNC PNL #1	CPO	C-7
R NUTCRACKER	CPO	C-13
DISPLAY CONT #1	CP	F-5
DISPLAYS FAN #1 (1096 & Subs)	CP	D-5
AUX HYD PUMP	CPO	D-5
RH RR FAN CONT	CP	H-2
FDR CONT #1	CP	C-8

Essential DC Bus Copilot 3 CBs
Table 9

Circuit Breaker Name	CB Panel	Location (PDB Section)
COPILOT 3	**PDB**	**DC ESS**
CDU #1	CP	D-7
NAV CMPTR #1	CP	D-8
DME #1	CP	F-10
ADF CONT #1	CP	F-12
L WSHLD WIPER	CP	J-11
L COWL A/I PRESS	CP	J-12
L COWL ANTI-ICE	CP	J-13
L WING ANTI-ICE	CP	J-14
CKPT AUDIO #2	CP	I-8
DAU #2A	CP	B-14
DAU #1B	CP	C-14
LH RR COOL FAN	CP	K-1
NAV RCVR #1 (1144 & Subs)	CP	F-9
ADF #1 (1144 & Subs)	CP	F-11
SMOKE DET	CP	E-14
G METER	CP	C-13
FWC 2	CP	B-13

Other Essential DC Bus CBs
Table 10

Circuit Breaker Name	CB Panel	Location (PDB Section)
ACCESS	**PDB**	**DC ESS**
L CONV	**PDB**	**DC ESS**
R CONV	**PDB**	**DC ESS**
TRU	**PDB**	**DC ESS**
E INV	**PDB**	**DC ESS**
EXT PWR	**PDB**	**DC ESS**
L MAIN BOOST PUMP	**PDB**	**DC ESS**
R MAIN BOOST PUMP	**PDB**	**DC ESS**

Left Main DC Bus CBs
Table 11

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT 1	**PDB**	**LEFT DC**
L MAIN DC SENSE	P	H-14
L ENG IDLE	P	K-8
SIGN LTS	P	D-5

Circuit Breaker Name	CB Panel	Location (PDB Section)
PEDESTAL	P	A-8
TAXI LTS CONT	P	C-9
FUEL PUMP LOGIC	PO	C-4
DC BPCU PWR #1	PO	C-6
AC BPCU PWR #1	PO	C-5
ADC XFER	PO	D-15
L MAIN DC VM	P	E-13
L CONSOLE PWR SUPPLY	P	D-1
AFT PED PWR SUPPLY #2	P	E-3
AFT PED PWR SUPPLY #1	P	E-2
R CONSOLE PWR SUPPLY	P	E-1
PILOT 2	**PDB**	**LEFT DC**
L LDG LIGHT CONT	P	D-7
L CONSOLE FLOOD	P	A-10
OVHD/PED FLOOD LTS	P	A-9
NAV/INSP LTS CONT	P	D-8
L ENG ACU SOL (1048 & Subs)	P	J-11
COPILOT 1	**PDB**	**LEFT DC**
RADIO ALTM #1	CP	I-4
IRU #1 DC PRI	CP	K-5
DATA LOADER	CP	I-7
W RDR R/T	CP	F-13
W RDR CONT #1	CP	F-14
FLT HYD CONT	CPO	C-5
GND SPOILER	CPO	C-6
A/T SERVO #1	CPO	C-9
DISPLAY UNIT #5	CP	I-6
COPILOT 2	**PDB**	**LEFT DC**
WHEEL SPEED	CPO	C-10
PERF #1A	CPO	C-11
COPILOT 3	**PDB**	**LEFT DC**
AHRS DC	CP	M-5
SYM GEN #3	CP	M-3
IRS BATT CHGR #1	CP	H-13
HF MIC ADAPTER #1 (1000 - 1095)	CP	F-8
HF R/T CPLR #1	CP	D-12
HF CONT #1	CP	D-11
ACCESS	**PDB**	**LEFT DC**

Right Main DC Bus CBs
Table 12

Circuit Breaker Name	CB Panel	Location (PDB Section)
PILOT 1	**PDB**	**RIGHT DC**

Circuit Breaker Name	CB Panel	Location (PDB Section)
R ENG IDLE	P	L-8
WARNING LTS PWR #9	P	G-4
R FLT PNL/GSHLD	P	B-7
BOT A/C LT (No ASC 10)	P	D-6
BCN LTS CONT (ASC 10)	P	D-6
ENGINE SYNC	P	K-9
ALT PUMP CONT	PO	D-4
DC-BPCU PWR #2	PO	D-6
AC-BPCU PWR #2	PO	D-5
F E-BAT ALT PWR	P	J-14
PILOT 2	**PDB**	**RIGHT DC**
R MAIN DC SENSE	P	I-14
R LDG LT CONT	P	F-7
R CONSOLE FLOOD	P	B-10
STROBE LTS CONT	P	F-8
R MAIN DC VM	P	F-13
R ENG ACU SOL	P	K-11
TONE WARN #2	P	B-5
CABIN TEMP CONT (1120 & Subs)	PO	D-14
A E-BAT ALT PWR	P	K-14
COPILOT 1	**PDB**	**RIGHT DC**
RADIO ALTM #2	CP	J-4
DADC #2	CP	G-3
SYM GEN #2	CP	L-3
DISPLAY UNIT #2	CP	F-6
DISPLAY UNIT #6	CP	J-6
IRU #2 DC	CP	L-5
CDU #2	CP	E-7
NAV CMPTR #2	CP	E-8
NAV/DME CONT #2	CP	E-9
TOTAL TEMP VALVE	CP	M-10
R PITOT HT CONT	CP	M-12
W RDR CONT #2	CP	G-14
STAB AUG SERVO #2	CPO	B-6
FGC #2	CPO	B-7
A/P SERVO #2	CPO	D-8
STALL BARR VALVE #2	CPO	B-12
STALL WARN CMPTR #2	CPO	B-11
FLT GNDC PNL #2	CPO	D-7
SPZ-8000 SHUT OFF	CP	L-4
COPILOT 2	**PDB**	**RIGHT DC**
R WSHLD WIPER	CP	K-11
A/T SERVO #2	CPO	D-9
STALL BARRIER #2	CPO	B-9

Circuit Breaker Name	CB Panel	Location (PDB Section)
SHAKER #2	CPO	B-10
PERF #2A	CPO	D-11
STEER BY WIRE #2	CPO	D-12
NOSE COMPT COOL VLV	CP	L-1
BTMS	CP	D-10
COPILOT 3	**PDB**	**RIGHT DC**
DISPLAY UNIT #4	CP	H-6
DME #2	CP	G-10
VHF COMM #2	CP	I-9
VHF COMM CONT #2	CP	I-10
ADF CONT #2	CP	G-12
TDR/CAD #2	CP	I-11
TDR/CAD CONT #2	CP	I-12
AOA PRB HTR #2	CP	M-14
IRS BATT CHGR #2	CP	I-13
DISPLAY CONT #2	CP	G-5
PED COOL FAN	CP	J-1
#2 HF MIC ADAPTER (1000 - 1095)	CP	G-8
R STBY ENG INSTR	CP	B-11
ADF #2 (1144 & Subs	CP	G-11
NAV RCVR #2 (1144 & Subs	CP	G-9
DISPLAYS FAN #2	CP	D-6
NOSE COMPT COOL FAN (1096 & Subs)	CP	M-1
HF RT/CPLR #2 (1168 & Subs)	CP	E-12
HF CONT #2 (1168 & Subs)	CP	E-11
ACCESS	**PDB**	**RIGHT DC**
L ALT BOOST PUMP	**PDB**	**RIGHT DC**
R ALT BOOST PUMP	**PDB**	**RIGHT DC**

Components Powered By The Emergency Batteries
Table 13

Battery Function	No. 1 Emergency Battery	No. 2 Emergency Battery
Emergency Buses	Inboard Anti-Skid	Outboard Anti-Skid
	No. 1 Clock	No. 2 Clock
	No. 1 Standby DDRMI/DBDI	No. 2 Standby DDRMI/DBDI
	No. 1 Comm	Standby Horizon
	No. 1 Comm Control	Left Fuel Quantity Indicator
	No. 1 Audio	Right Fuel Quantity Indicator
	No. 1 Transponder (TDR) and Control Adapter (CAD)	Left Standby Engine Instruments
	Standby Mach/Airspeed Indicator	Right Standby Engine Instruments
	Standby Altimeter	Cockpit Lights Control Box
	Standby Electrical Power Panel	Landing Gear Position Lights
	BCS Channel 1	Flap/Horizontal Stabilizer Position Indicator
Memory Keep Alive	No. 1 COMM	No. 2 COMM
	No. 1 TDR/CAD	No. 2 TDR/CAD
	No. 1 NAV	No. 2 NAV
	No. 1 DME	No. 2 DME
	No. 1 ADF	No. 2 ADF
Note for SN 1467 and subs: When activated, the No. 3 and 4 emergency batteries will automatically power the interior emergency lighting and main entrance door emergency lighting (if main entrance door is open).		

B. Crew Alerting System (CAS) Messages:

(1) Caution (Amber) CAS messages associated with the DC electrical power system are:

CAS Message	Possible Cause(s)
BATT 1-2 CHGR FAIL	Battery or Battery Charger has failed, or input power CB is open.
BATT ON BUS	Essential DC Bus powered by batteries only.
L-R CONV FAN FAIL	A converter cooling fan has failed.
L-R CONV HOT	Converter temperature is above 220° F (104° C).
DC ESS ON BATT [1]	Essential DC Bus powered by batteries only.
L-R DC POWER FAIL	DC output from converter has dropped off line.
EPMP BATT SW OFF	BATT 1 (or BATT 2) switch is selected to OFF.
TRU FAIL	Transformer-rectifier unit has no output.
TRU HOT	Transformer-rectifier unit temperature above 374° F (190° C).

[1] For SPZ-8000 equipped airplanes having ASC 415 incorporated.

Annunciation	Cause or Meaning
DC RESET light (amber) illuminated on EPMP.	DC Bus fault indicated.

Annunciation	Cause or Meaning
BATT 1 or BATT 2 CHGR FAIL light (amber) illuminated on EPMP.	Battery or battery charger has failed, Left Main AC bus or Right Main AC bus has failed, or input power circuit breaker is open.

(2) Advisory (Blue) CAS messages associated with the AC electrical power system are:

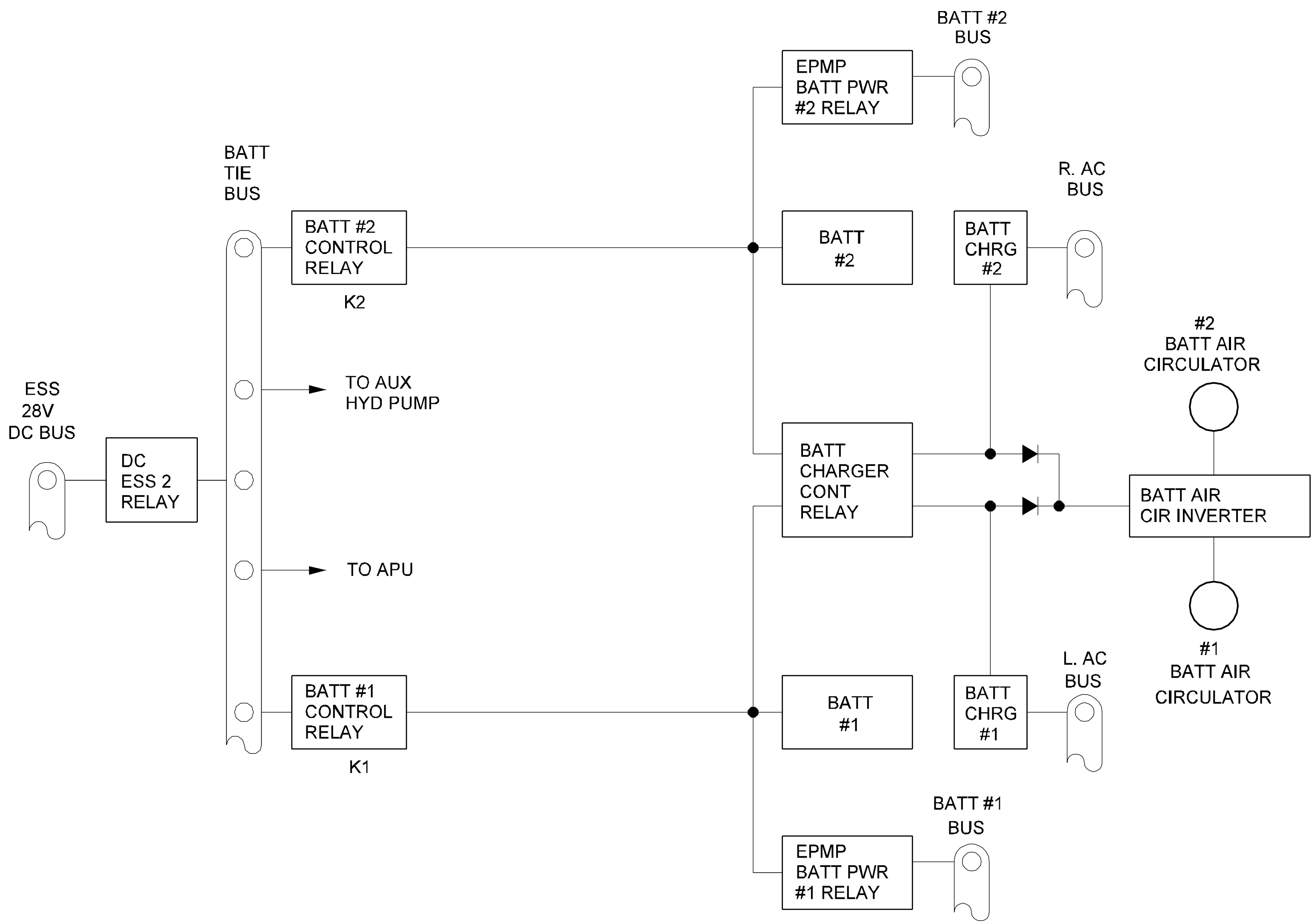
CAS Message	Possible Cause(s)
BATT ON BUS [(1)]	Battery chargers are in T/R mode (APU start or AUX pump ON), or Essential DC bus is on battery power.
DC EXT POWER	DC external power is connected to airplane.
E BATT 1-2 DISCH [(2)]	Indicated emergency battery is discharging.
E BATT 1-2 FAIL [(3)]	Indicated emergency battery has failed.
EXT BATT SWITCH ON	External battery switch is ON.

[(1)] For SPZ-8400 equipped airplanes.
[(2)] For airplanes 1467 & subs: E BATT 1-2-3-4 DISCH
[(3)] For airplanes 1467 & subs: E BATT 1-2-3-4 FAIL

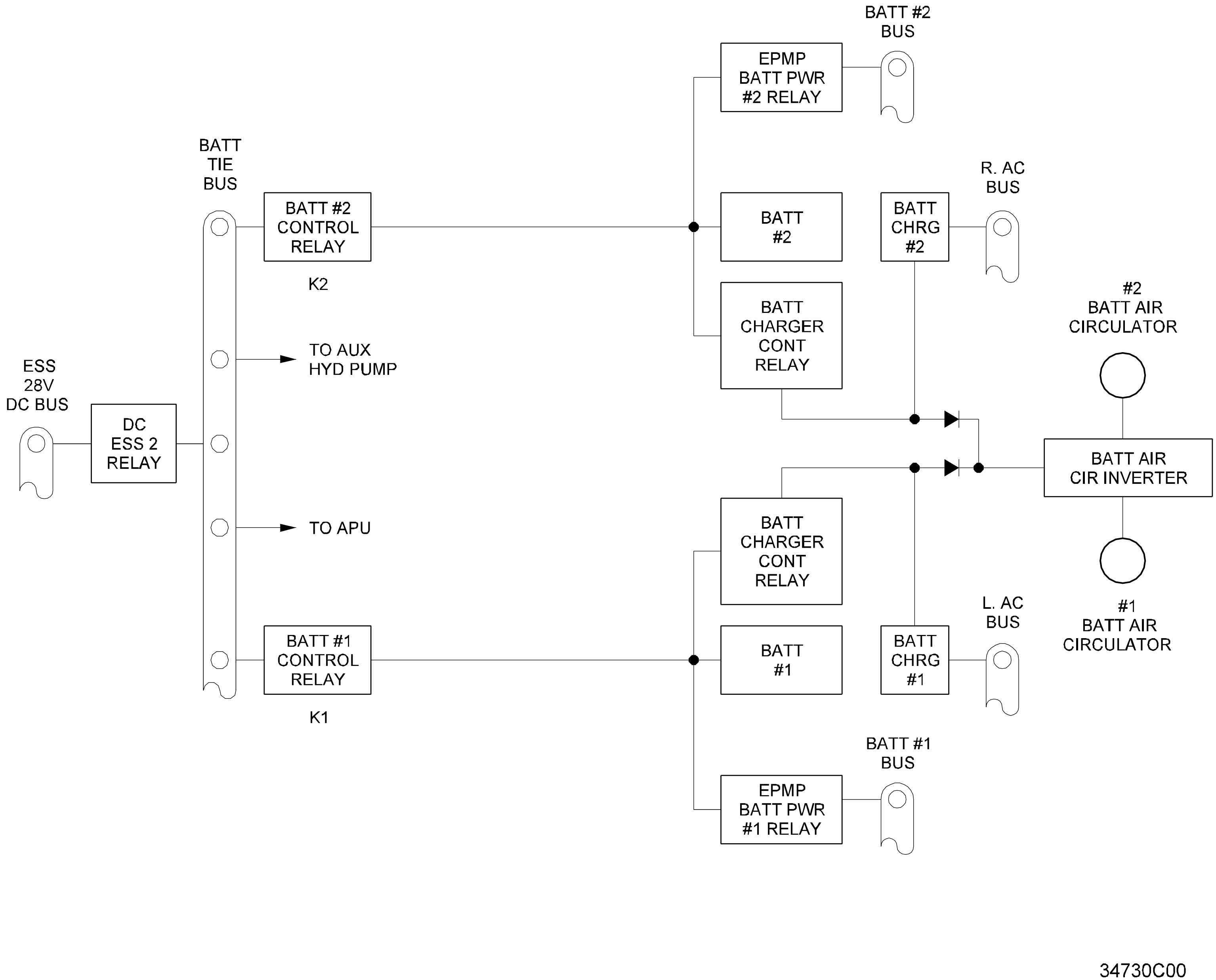
Annunciation	Cause or Meaning
DC EXT PWR light (blue) illuminated on overhead panel.	DC external power applied.
EXT BATT SW light (blue) illuminated on overhead panel.	OUTSIDE BATTERY switch (or EXTERNAL BATTERY switch, if installed) has been selected to ON.
GND SVCE BUS annunciator(amber) illuminated on overhead panel.	Ground service bus energized. Requires airplane electrical power (battery power minimum) for annunciator to illuminate.

5. Limitations:

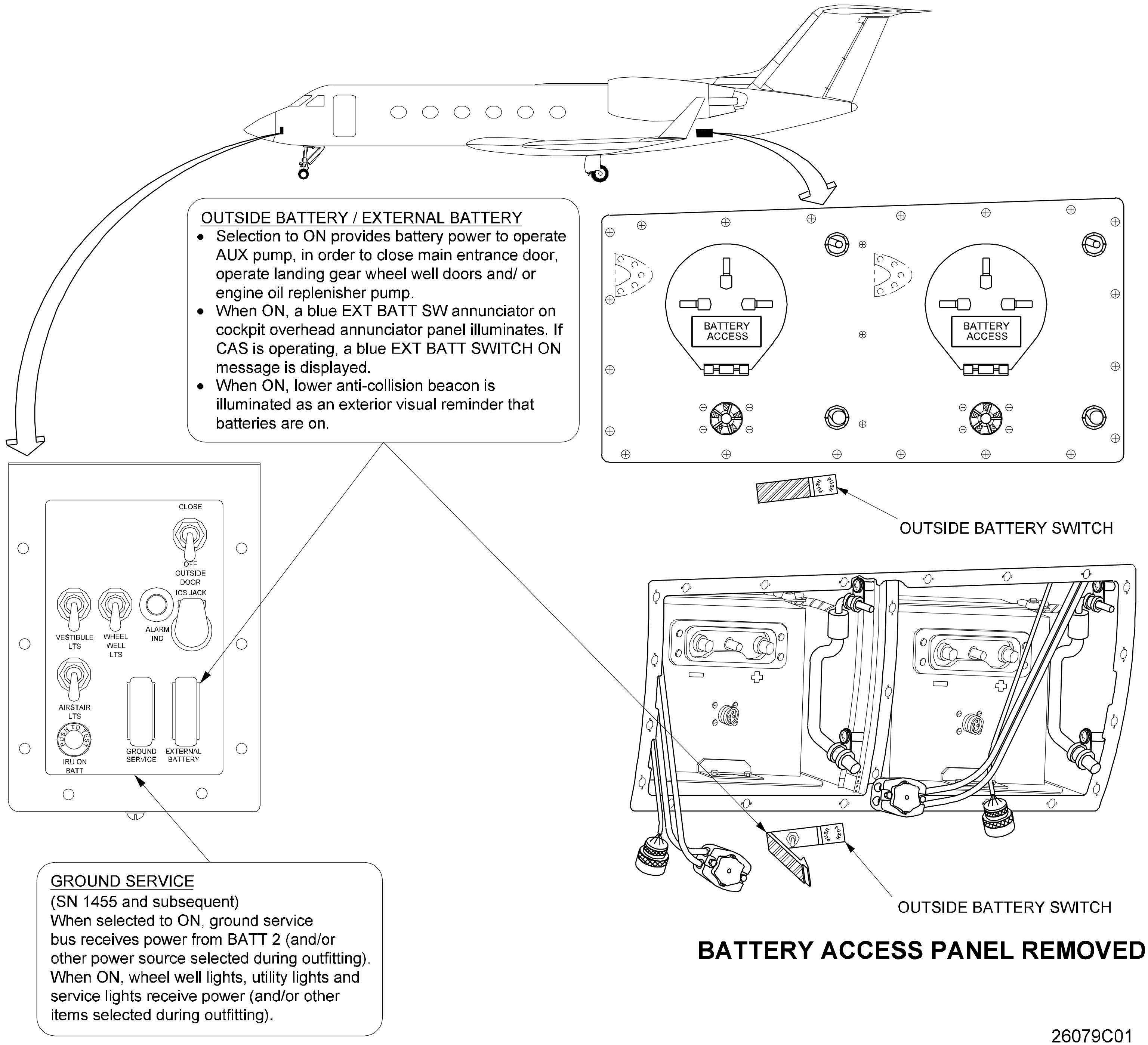
There are no limitations for the DC electrical power system at the time of this revision.



Battery Power System
Simplified Block Diagram
(SN 1000 - 1464 Not
Having ASC 54)
Figure 13



Battery Power System
Simplified Block Diagram
(SN 1465 & Subs, SN
1000 - 1464 Having ASC
54)
Figure 14



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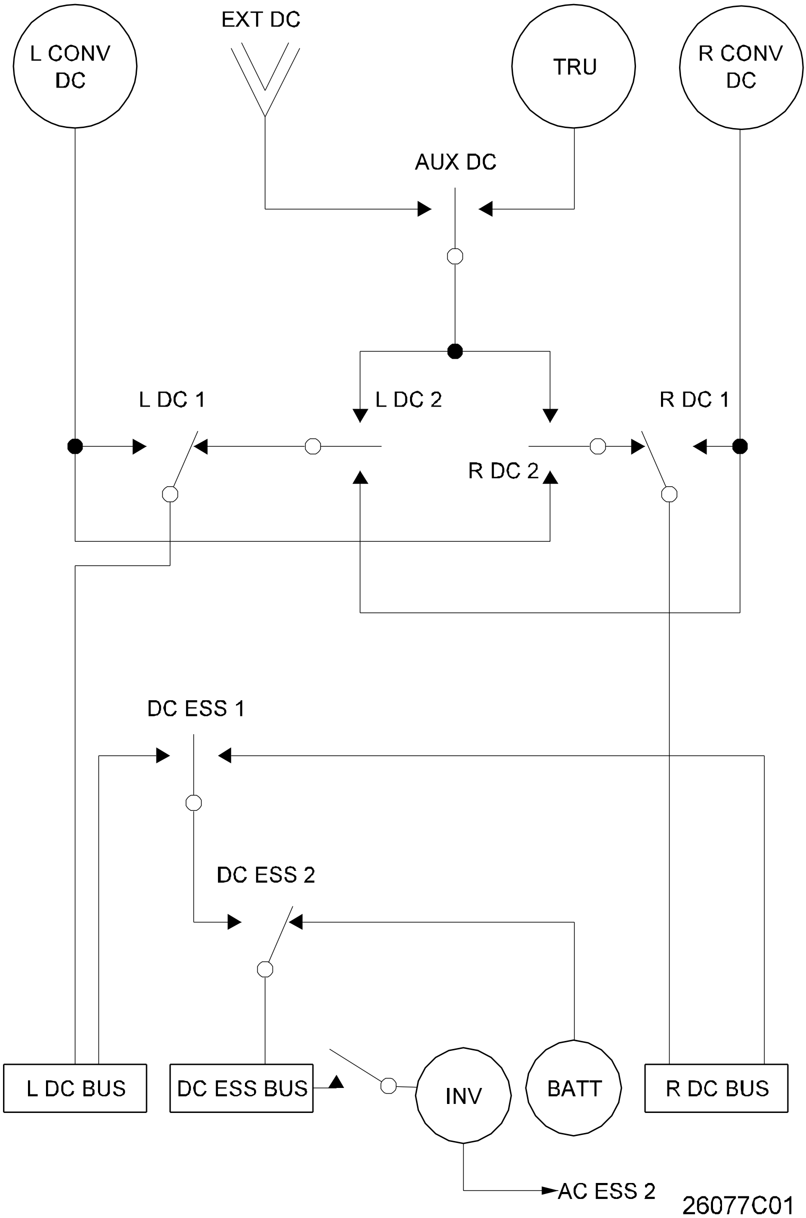
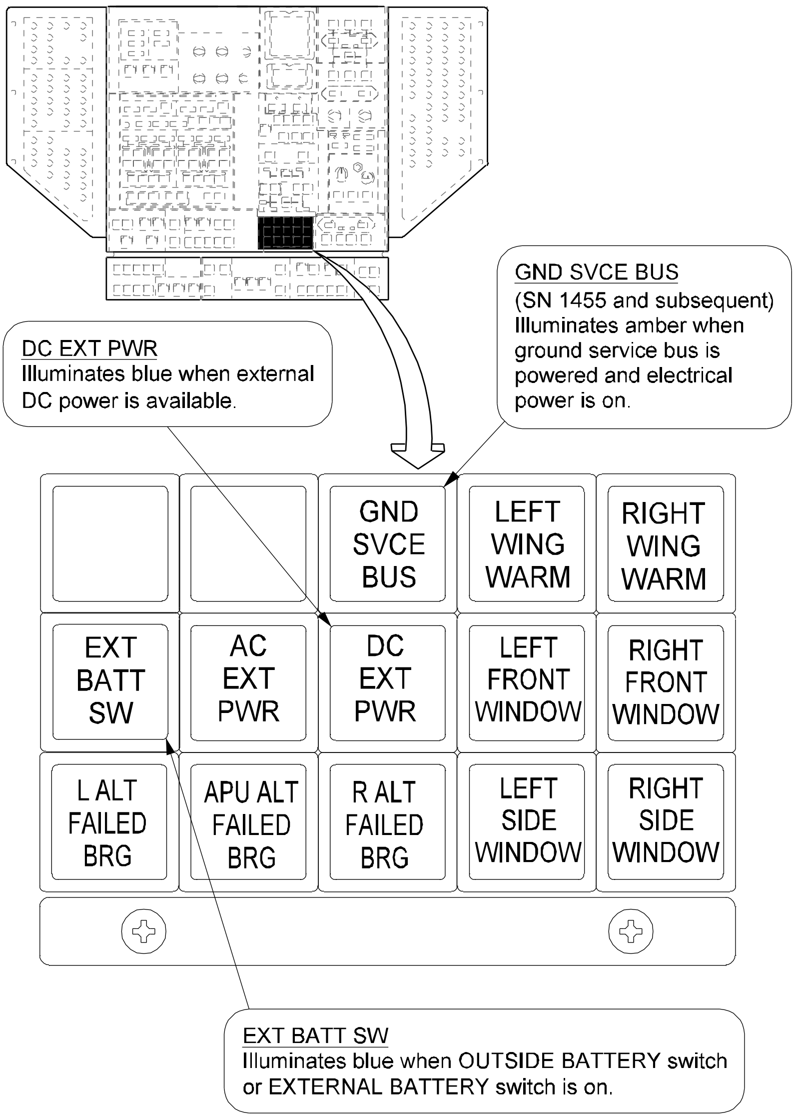
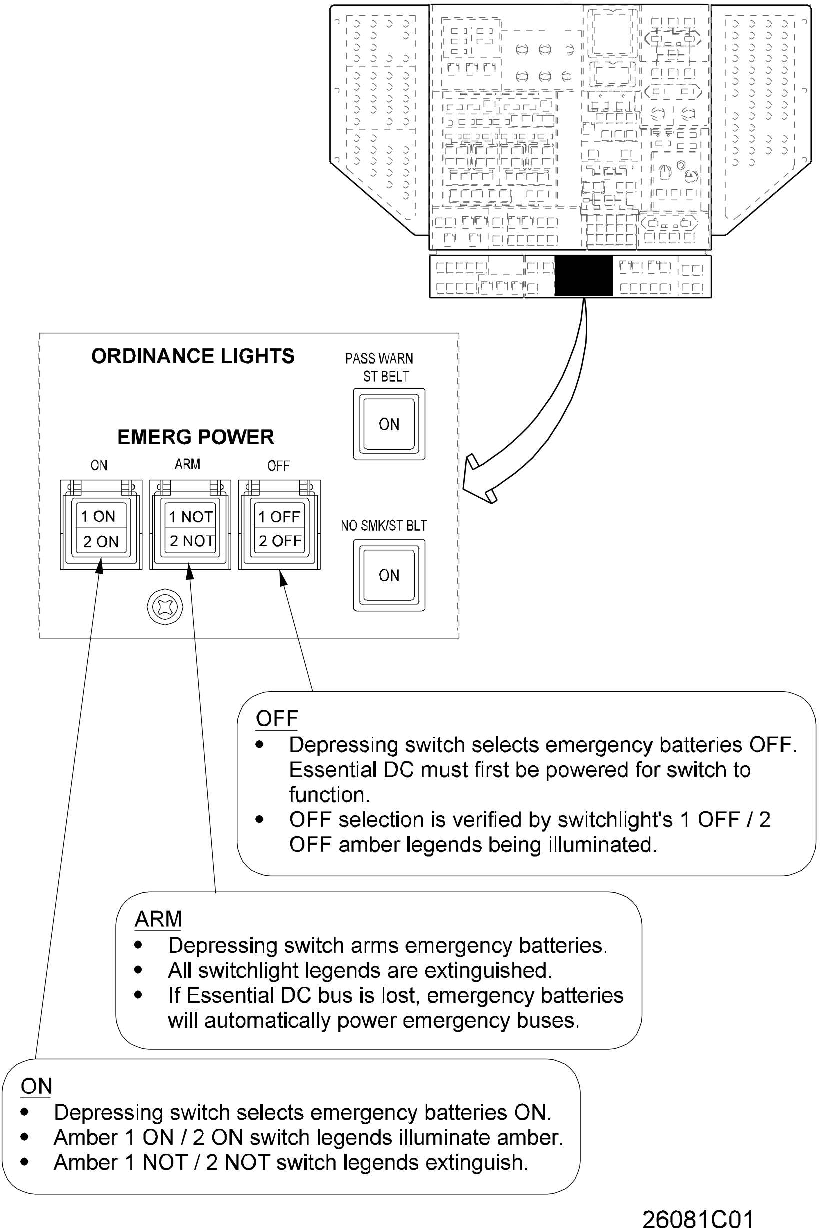
External Battery Switches / Ground Service Bus Switch
Figure 15



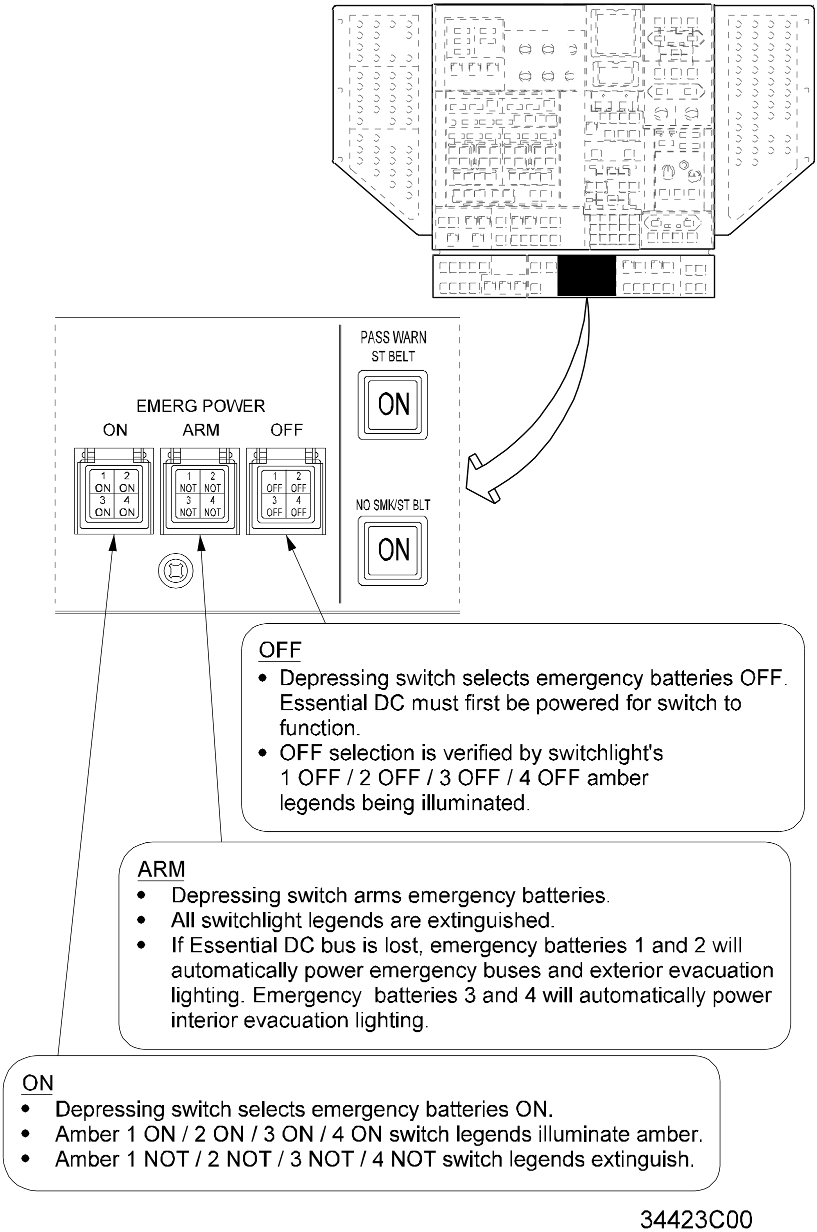
DC Electrical Power System Simplified Block Diagram
Figure 16



Overhead Annunciator Panel
Figure 17



EMERG POWER Control Panel (SN 1000 - 1466)
Figure 18



EMERG POWER Control Panel (SN 1467 & Subs)
Figure 19

2A-26-10: General

The Gulfstream IV fire protection system provides a means to detect and indicate fire or overheat conditions, and to store and distribute fire extinguishing agent to all protected areas of the aircraft.

The fire detection system alerts the flight crew whenever a fire or overheat condition develops in the engine nacelles or APU enclosure. This is accomplished presenting both aural and visual warnings when temperatures reach an overheat condition. Area detectors are used in large zones; spot detectors are used in more focused areas. The engine portion of the system also provides an alert when the sensing elements (called firewires) develop a fault.

The fire extinguishing system suppresses left engine nacelle, right engine nacelle and APU compartment fires. Three extinguishing agent bottles are installed in the tail compartment; two for engine fire extinguishing and one for APU fire extinguishing. Provisions for testing the extinguishing system are also provided. In addition, two portable fire extinguishers are provided, both accessible in flight.

The Fire Protection System is divided into the following subsystems:

- 2A-26-20: Smoke Detection and Evacuation System
- 2A-26-30: Fire Detection and Warning System
- 2A-26-40: Fire Extinguishing System

2A-26-20: Smoke Detection and Evacuation System

1. **General Description:**

   A. **Smoke Detection System:**

   Aircraft SN 1034, SN 1156 and Subsequent

   The smoke detection system provides a photo-cell smoke detector in the baggage compartment to detect and warn of smoke. It is operable and capable of being tested any time 28V Essential DC bus power is available.

   Aircraft with ASC 268 incorporated have an additional smoke detector installed in the forward lavatory / radio rack area.

   B. **Smoke Evacuation System:**

   Smoke evacuation is accomplished by depressurizing the baggage compartment door seal, allowing smoke to escape overboard. Depending on aircraft effectivity, depressurizing the seal is accomplished using either the baggage door handle or the smoke evacuation valve.

2. **Description of Subsystems, Units and Components:**

   A. **Smoke Detection System:**

   Aircraft SN 1034, SN 1156 and Subsequent

   (1) Smoke Detector:

   Located in the baggage compartment, the smoke detector receives power from the Essential DC bus. It functions by passing a steady beam of light across a white surface. Smoke particles entering the detector interrupt the light beam, resulting in the following annunciations, depending on effectivity:

(a) SPZ-8000 equipped aircraft:

- Red SMOKE DETECT warning message on CAS
- Red SMOKE DETECT warning message on Standby Warning Lights Panel (SWLP)
- Red BAG COMPT SMOKE annunciator illuminating above copilot's NAV display (Figure 1)

(b) SPZ-8400 equipped aircraft:

- Red AFT BAG SMOKE warning message on CAS
- Red SMOKE DETECT warning message on SWLP (if installed)

(2) Other Smoke Detector Annunciations:

(a) On SPZ-8400 equipped aircraft, a red FWD LAV SMOKE DETECT warning message is displayed on CAS if the forward smoke detector detects smoke in the forward lavatory / radio rack area.

(b) On SPZ-8000 and SPZ-8400 equipped aircraft with ASC 415 incorporated, and airplanes SN 1390 and subsequent, a red RADIO RACK SMOKE warning message is displayed on CAS if the forward smoke detector detects smoke in the radio rack area. This CAS message is accompanied by a red SMOKE DETECT warning message on the SWLP (if installed).

(3) Smoke Detector Test Switch:

A test switch, labeled SMOKE DET TEST and shown in Figure 2, is provided to test the smoke detection system. The location and type of switch (pushbutton on center pedestal or toggle switch on copilot's console) depends on production serial number of the aircraft and operator preference during outfitting. Provided Essential DC bus power is available, selection of the TEST function results in the annunciations listed in (1).

B. Smoke Evacuation System:

Aircraft having ASC 18 (CAA Aircraft); SN 1000 through 1155 (excluding SN 1034) having ASC 157; SN 1034; SN 1156 and subsequent: Rotation of the emergency smoke evacuation valve handle (Figure 3) to the EVAC SMOKE position allows deflation of the baggage compartment door seal. With the door seal deflated, pressurized cabin air escapes around the door frame, drawing with it any smoke. Cabin altitude will climb accordingly. When smoke ceases, rotation of the valve handle to NORMAL OPS allows the door seal to reinflate.

Aircraft not having ASC 18; SN 1000 through 1155 (excluding SN 1034) not having ASC 157: Rotation of the baggage compartment door handle 45°, without opening the door, depressurizes the baggage compartment door seal.

3. **Controls and Indications:**

A. **Circuit Breakers (CBs):**

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
SMOKE DET	CP	E-14	ESS DC Bus

B. **Messages and Annunciations:**

CAS Message:	SWLP Indication	Cause or Meaning:
AFT BAG SMOKE (1)	SMOKE DETECT	Smoke detected in aft baggage compartment.
FWD LAV SMOKE DETECT (1)	None	Smoke detected in forward lavatory area or radio rack area.
RADIO RACK SMOKE (2)	SMOKE DETECT	Smoke detected in radio rack area.
SMOKE DETECT	SMOKE DETECT	Smoke detected in area illuminated by annunciation.

NOTE(S):

(1) For SPZ-8400 equipped airplanes.

(2) For SPZ-8000 and SPZ-8400 equipped airplanes with ASC 415, and airplanes SN 1390 and subsequent.

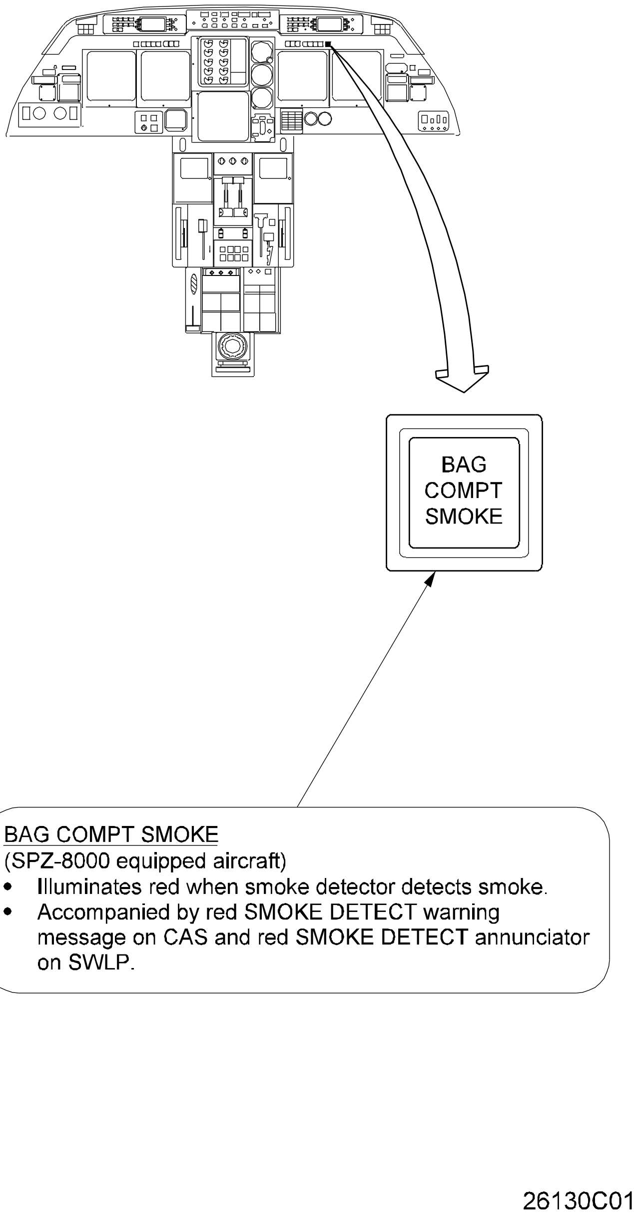
Annunciation:	Cause or Meaning:
Red BAG COMPT SMOKE light above copilot’s NAV display. (1)	Smoke detected in aft baggage compartment.

NOTE(S):

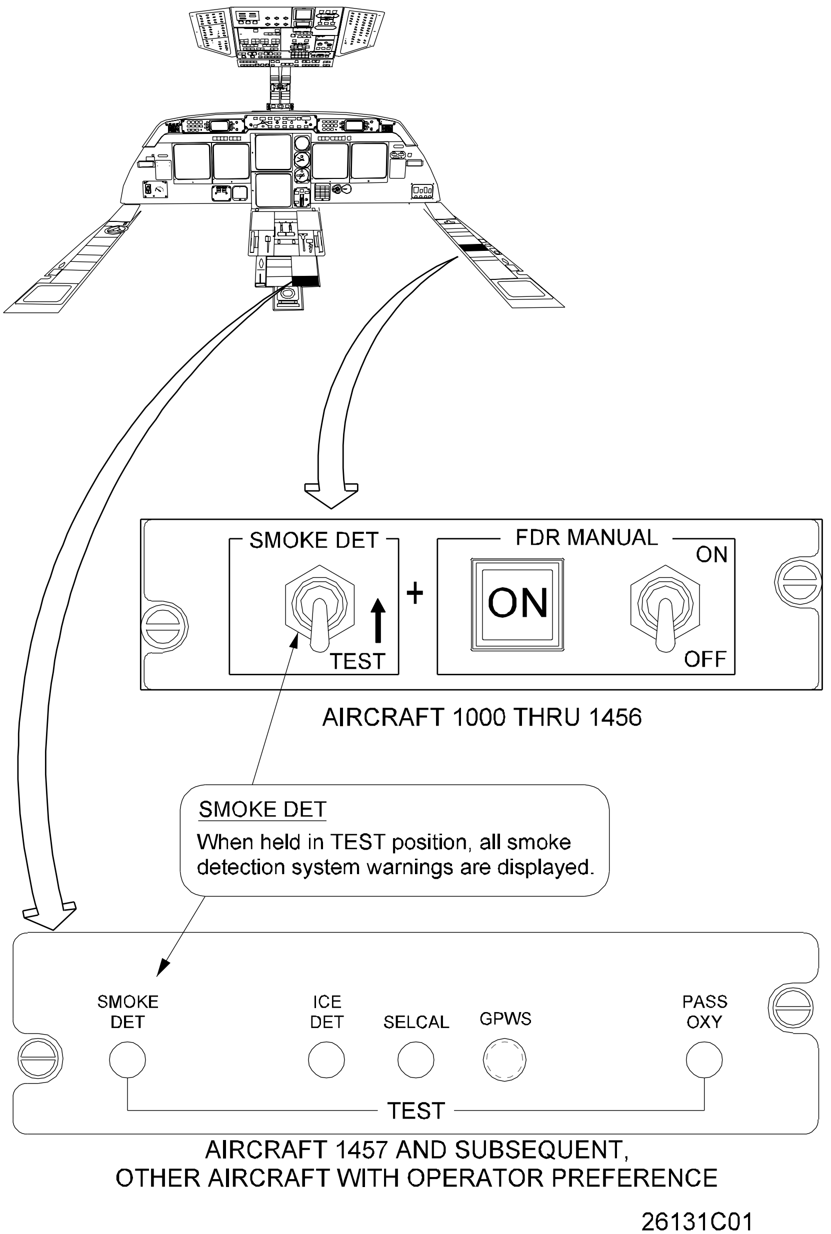
(1) For SPZ-8000 equipped aircraft.

4. **Limitations:**

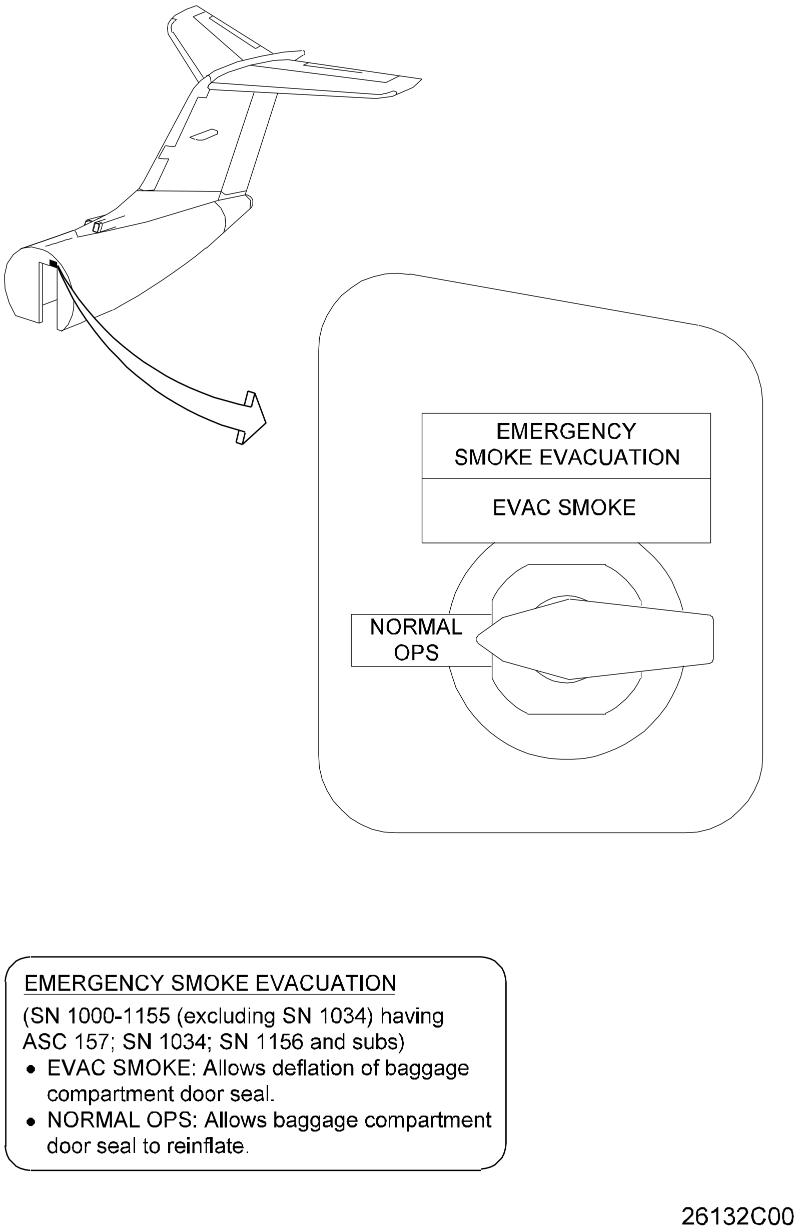
There are no limitations established for this system as of this revision.



Baggage Compartment Smoke Annunciator
Figure 1



Smoke Detector System TEST Switch
Figure 2



Emergency Smoke Evacuation Valve
Figure 3

2A-26-30: Fire Detection and Warning System

1. General Description:

The fire detection and warning system provides a means to detect and indicate the presence of an overheat or fire condition in the engine areas and APU enclosure. A dual channel system, it will remain functional should a single channel fail or one of its power sources be lost.

The system receives power upon selection of the battery switches to ON. It will remain operable at all times provided the Essential DC bus is powered and all associated circuit breakers are closed. Prior to APU start, the APU fire detection system is tested by the flight crew. Likewise, prior to engine start, the engine fire detection system is also tested. Any failures or faults during engine portion of the fire detection system testing are annunciated.

The fire detection and warning system remains active in sensing its respective areas during all phases of flight. The engine portion of the fire detection system continuously monitors self-health, annunciating any failures or faults.

During shutdown, the fire detection and warning system remains operable until selection of the battery switches to OFF.

2. Description of Subsystems, Units and Components:

A. Engine Fire Detection System:

A dedicated continuous-wire fire detection system monitors engine fire zone temperature with two parallel dual loops. The parallel dual loops, LOOP A and LOOP B, which are approximately one inch apart, provide independent, yet related, fire detection. Each loop consists of three series-wired sensing elements that are attached to the lower side of the engine combustion area, forward fixed cowl and aft fixed cowl. Each engine's two loops then connects to a digital engine fire detection control unit located in the tail compartment.

In response to an engine fire, the fire detection control unit triggers the appropriate visual and aural warnings. If the detection system malfunctions, the control unit triggers the appropriate CAS message and illuminates the overhead panel fault lights.

(1) Sensing Loop and Elements:

The engine's dual loops consist of two element assemblies parallel-mounted to rigid stainless steel rails. Each rail conforms to the shape of the area for which fire detection is provided. Each loop consists of three series-wired sensing elements, commonly called firewire segments.

A firewire consists of a thin stainless steel sheath containing a coaxial wire. The coaxial wire is insulated from the sheath by a semi-conductor material. Resistance of the semi-conductor varies proportionally with temperature, falling as sensor temperature increases. Simultaneously, capacitance of the sensor increases, providing a basis for logic discrimination between fire or fault. Both ends of each coaxial wire is connected to its respective portion of the fire detection control unit, while the sheath connects to the aircraft's electrical ground.

If either loop should fail or become shorted or fail for any reason, a fault alert illuminates. The flight crew may then select the faulty loop

off, leaving the remaining loop to function as a single loop system. Should a loop ever be severed, it will continue to operate normally, but will indicate as faulty during testing.

(2) Fire Detection Control Unit (FDCU):

The FDCU is a digital device that contains separate but identical circuit boards for the LOOP A and LOOP B fire detection loops. Each circuit board contains a voltage regulator, fire and fault warning logic circuits, fire and fault warning output circuits, a firewire driver and a fire warning comparator and reference selector.

Supplied with 28 VDC power from the Essential DC bus through the L/R FIRE DET circuit breakers, the FDCU generates a 400 Hz squarewave that is supplied to the fire detection loops. The FDCU uses this squarewave output and the resulting feedback signal to provide fire and fault detection.

(3) Test Control Panels:

Two system test control panels are provided for the flight crew to verify proper system operation. They are located on the cockpit overhead panel and are labeled FIRE TEST (Figure 4) and FIRE DETECTION (Figure 5). With the Essential DC bus powered, testing of the system is possible.

(4) Fire Annunciations:

When exposed to high temperatures, (e.g., engine fire), semi-conductor resistance drops and capacitance rises. If this occurs in both loops (except when a failed loop is isolated) and the changes are within the predetermined bounds as caused by heating of the firewire, the FDCU responds by triggering the fire warnings and providing a signal to the Data Acquisition Unit (DAU). Fire warnings include:

- Both upper and lower segments (LOOP A and LOOP B) of the affected engine FIRE TEST switchlight illuminate red
- The affected engine FIRE handle illuminates red. The handle's safety solenoid is activated, allowing the handle to be pulled. (Manual release of the handle is still available in the event of solenoid failure.)
- The affected engine HP fuel cock illuminates red
- Red ENG FIRE LOOP ALRT and warning message is displayed on CAS
- Red L FIRE DET LOOP capsule illuminates on the Standby Warning Lights Panel (SWLP), if the SWLP is in manual mode, or if in automatic mode coupled with CAS failure
- Both red MASTER WARN lights illuminate, with corresponding aural warning tone
- The ENGINE FIRE checklist is displayed on the lower center portion of the copilot's navigation display (DU 5) in MAP or COMP or PLAN mode (airplanes SN 1144 and subsequent and SN 1000 through 1143 with ASC 178 incorporated)

As the temperature drops, (e.g., after the fire is extinguished), semi-conductor resistance rises, capacitance drops and the FDCU

deactivates the fire warnings.

(5) Fault Annunciations:

If resistance and capacitance changes fall outside the predetermined bounds for a fire alarm, a fault alarm is generated. Additional protection against false alarms is achieved by requiring fire signals from both LOOP A and LOOP B before all cockpit fire alarm warnings are activated. A fire signal from either LOOP A or LOOP B only results in the following faulty loop annunciations, unless one of the loops has been selected off:

- Upper (FAULT) segment of the affected LOOP switchlight illuminates amber
- An amber ENG FLT LOOP ALRT caution message is displayed on CAS
- Both amber MASTER WARN lights illuminate, with corresponding aural caution tone

The flight crew would then isolate the faulty loop by selection of the illuminated switch to OFF. This restores the system to operation on the remaining good loop.

(6) Engine Fire Detection Test:

The flight crew normally performs the following engine fire detection test in the course of their normal procedures:

(a) Depress and hold the L ENG FIRE TEST switch. Verify the following:

- LOOP A and LOOP B segments of the L ENG FIRE TEST switchlight illuminate red
- Left (L) FIRE handle illuminates red
- Left engine HP fuel cock illuminates red
- Red ENG FIRE LOOP ALRT and L ENGINE HOT warning messages are displayed on CAS
- Red L ENGINE HOT and FIRE DET LOOP capsules illuminate on the Standby Warning Lights Panel (SWLP), if the SWLP is in manual mode
- Both red MASTER WARN lights illuminate, with corresponding aural warning tone
- The ENGINE FIRE checklist is displayed on the lower center portion of the copilot's navigation display (DU 5) in MAP or COMP or PLAN mode (airplanes SN 1144 and subsequent and SN 1000 through 1143 with ASC 178 incorporated)

(b) Release the L ENG FIRE TEST switch.

(c) Depress and hold the R ENG FIRE TEST switch. Verify the same annunciations are present, corresponding to the right engine.

(d) Release the R ENG FIRE TEST switch.

(7) Engine Fire Detection Fault Test:

The flight crew normally performs the following engine fire detection

fault test in the course of their normal procedures:

(a) Depress and hold the FIRE DETECTION FAULT TEST switch. Verify the following:

- Upper (FAULT) segment of the all four (4) LOOP switchlights illuminate amber
- An amber ENG FLT LOOP ALRT caution message is displayed on CAS
- Both amber MASTER WARN lights illuminate, with corresponding aural caution tone

(b) Release the FIRE DETECTION FAULT TEST switch.

B. APU Fire Detection System:

The APU is installed in a titanium and stainless steel enclosure in the tail compartment. The enclosure serves to isolate the APU from the aircraft structure and provide fire containment. Fire detection is provided by three thermal switches strategically located on the APU enclosure, with the sensing portion of the switches inside the enclosure. A fourth thermal sensing switch is placed in the air inlet duct. Placement and trip points of the sensors are:

- Top left side (APU accessory section) (450° F [232° C])
- Top center forward (load control valve) (600° F [316° C])
- Bottom aft right corner (450° F [232° C])
- APU air inlet duct (450° F [232° C])

The four thermal switches are parallel connected; therefore, any one of the four switches reaching its trip point will trigger the visual and aural APU fire annunciations. Accordingly, failure of one switch does not render the system inoperative. When a thermal switch closes, 28 VDC from the Essential DC bus (through the APU FIRE DET circuit breaker) closes a logic relay, causing the following visual and aural APU fire annunciations:

- Red APU FIRE warning capsule on APU control panel illuminates (Figure 6)
- Red APU FIRE warning message is displayed on CAS
- Red APU FIRE capsule illuminates on the SWLP, if the SWLP is in manual mode, or if in automatic mode coupled with CAS failure
- Both red MASTER WARN lights illuminate, with corresponding aural warning tone
- Nosewheel well warning bell (or tone speaker) sounds (if aircraft is on the ground)
- The APU FIRE checklist is displayed on the lower center portion of the copilot's navigation display (DU 5) in MAP or COMP or PLAN mode (airplanes SN 1144 and subsequent and SN 1000 through 1143 with ASC 178 incorporated)
- If the logic relay closes due to an actual fire warning (not a test), the APU will also be made to flame out

When the temperature drops below the thermal switch trigger point, the switch opens to de-energize the logic relay, deactivating the fire warnings.

Testing the APU fire detection system is accomplished using the APU FIRE

TEST switch, located on the cockpit overhead panel and shown in Figure 4. With Essential DC bus power available and the switch depressed and held, all of the previously-listed annunciations will occur, except the APU, if running, will not be made to flame out. Releasing the switch removes all annunciations.

C. Engine Overheat Detection:

Low pressure engine bleed air cools various parts of the engine. After cooling the engine, the air exhausts overboard through a duct on the bottom of the engine cowling. A thermal switch in this duct monitors cooling air temperature as it is exhausted overboard.

If engine cooling air temperature exceeds 860 ±25° F (460 ±15° C), the thermal switch closes to trigger the following annunciations:

- Red L-R ENGINE HOT warning message is displayed on CAS
- Red L (or R) ENGINE HOT capsule illuminates on the SWLP, if the SWLP is in manual mode, or if in automatic mode coupled with CAS failure
- Both red MASTER WARN lights illuminate, with corresponding aural warning tone

Like the fire detection systems, when the temperature drops below the thermal switch trigger point, the switch opens, deactivating the overheat warnings. Also like the fire detection systems, the overheat detection system receives power from the Essential DC bus.

D. Other Overheat Detectors:

Although not included in the engine or APU fire / overheat detection and warning system, the following overheat sensing components and their resultant annunciations bear discussion here. For more information on these overheat sensing components, see their respective system description.

(1) Aft Equipment Compartment:

A thermal switch is installed on each side of the aft equipment compartment (tail compartment), adjacent to the bleed air manifolds. The two switches, wired in parallel, close at 200 ±5° F (93 ±3° C) to trigger a red AFT EQUIP HOT warning message on CAS. A red AFT EQUIP HOT capsule will illuminate on the SWLP, if installed.

(2) Engine Pylons:

Three thermal switches installed in each engine pylon adjacent to the high stage bleed air pressure regulator and precooler. If any of the thermal switches reaches 300 ±5° F (149 ±5° C) the switch closes to trigger a red L-R PYLON HOT warning message on CAS. A red L (or R) PYLON HOT capsule will illuminate on the SWLP, if installed.

(3) Bleed Air Ducts:

A thermal switch is installed downstream of each bleed air precooler heat exchanger. The switch closes when air temperature downstream of the precooler reaches 200 ±5° F (93 ±3° C), triggering an amber L-R BLEED AIR HOT caution message on CAS.

(4) Nose Cowl Anti-Icing:

A thermal switch is installed downstream of each engine's anti-ice valve. The switch closes when air temperature downstream of the valve reaches 662 ±15° F (350 ±8° C), triggering an amber L-R COWL A/I OVHT caution message on CAS.

(5) Wing Anti-Icing:

Each wing leading edge plenum contains three thermal switches. The switches monitor wing leading edge temperature at the inboard, mid-wing and outboard areas to provide overheat warnings. Normally, wing leading edge temperature with anti-icing operating is approximately 130° F (54° C).

Each wing's three switches are wired in parallel and will close at 180 ±5° F (82 ±3° C), triggering an amber L-R WING HOT caution message on CAS.

(6) Radio Racks and Nose Avionics Bay:

Ten thermal switches, eight installed in the left and right avionics bays and two in the nose avionics bay, monitor temperatures in these areas. The switches, wired in parallel, close at 200 ±5° F (93 ±3° C), triggering an amber FWD RADIO RACK HOT caution message on CAS.

(7) Under Floor (Aircraft 1280 and Subsequent):

Eight thermal switches are mounted under the floor in the aft cabin. The switches, wired in parallel, close between 120-135° F (49-58° C) to trigger an amber UNDER FLOOR O'HEAT caution message on CAS.

(8) Pressure Bulkhead Inspection Window:

An inspection window is installed in the center of the aft pressure dome bulkhead to permit visual inspection of the tail compartment from within the baggage compartment.

3. Controls and Indications:

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
APU FIRE WARN	P	A-4	ESS DC Bus
FIRE BELL	P	B-4	ESS DC Bus
L ENG O'HT	P	G-9	ESS DC Bus
R ENG O'HT	P	H-9	ESS DC Bus
L FIRE DET LOOP A	P	I-6	ESS DC Bus
L FIRE DET LOOP B	P	G-6	ESS DC Bus
R FIRE DET LOOP A	P	J-6	ESS DC Bus
R FIRE DET LOOP B	P	H-6	ESS DC Bus

B. Warning (Red) Messages and Annunciations:

CAS Message:	SWLP Indication	Cause or Meaning:
AFT EQUIP HOT	AFT EQUIP HOT	Aft equipment area temperature above 200° F (93° C). Possibility exists that high pressure duct has blown or that fire is in progress.

CAS Message:	SWLP Indication	Cause or Meaning:
APU FIRE	APU FIRE	APU fire detected.
ENG FIRE LOOP ALRT	FIRE DET LOOP	Engine fire loop senses fire.
L-R ENGINE HOT	L ENGINE HOT R ENGINE HOT	Engine cooling air temperature above 860° F (460° C).
L-R PYLON HOT	L PYLON HOT R PYLON HOT	Pylon temperature is above 325° F (163° C).

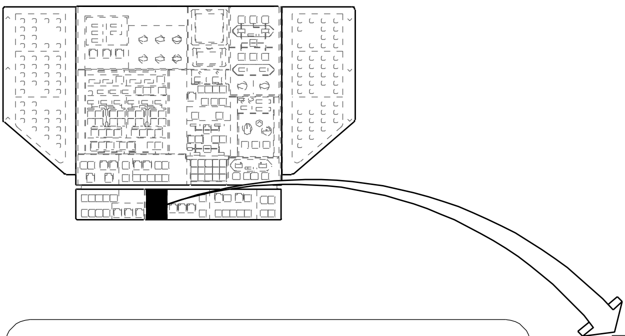
Annunciation:	Cause or Meaning:
Red LOOP A or LOOP B segments of L ENG or R ENG FIRE TEST switches illuminated. Red light in L / R Fire Handle. Red light in L / R HP FUEL COCK handle.	An engine fire loop senses fire.
Red APU FIRE light on the APU control panel.	APU fire sensors detect fire.
Nosewheel well APU fire bell or speaker tone sounds.	APU fire sensors detect fire.

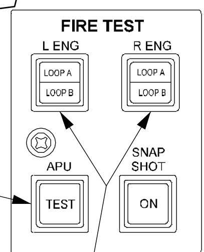
C. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
L-R BLEED AIR HOT	Bleed air temperature is above 550° F (288° C).
L-R COWL A/I OVHT	Engine cowl temperature is above 662° F (350° C).
ENG FLT LOOP ALRT	Engine fire detection loop fault detector active.
FWD RADIO RACK HOT	Inside radome, left or right equipment bay temperature has exceeded 200° F (93° C).
L-R WING HOT	Wing anti-ice exhaust duct temperature is greater than 180° F (82° C).
UNDER FLOOR O'HEAT	Bleed air leak detected under cabin floor.

4. Limitations:

There are no limitations established for this system as of this revision.





APU FIRE TEST

With switchlight depressed and held:

- Red APU FIRE warning capsule APU control panel illuminates.
- Red APU FIRE warning message is displayed on CAS.
- Red APU FIRE capsule illuminates on SWLP, if SWLP is installed and in manual mode.
- Both red MASTER WARN lights illuminate, with corresponding aural warning tone.
- Nosewheel well warning bell (or tone speaker) sounds (if aircraft is on ground).
- APU FIRE checklist is displayed on copilot's nav display (SN 1144 and subs; SN 1000-1143 with ASC 178).

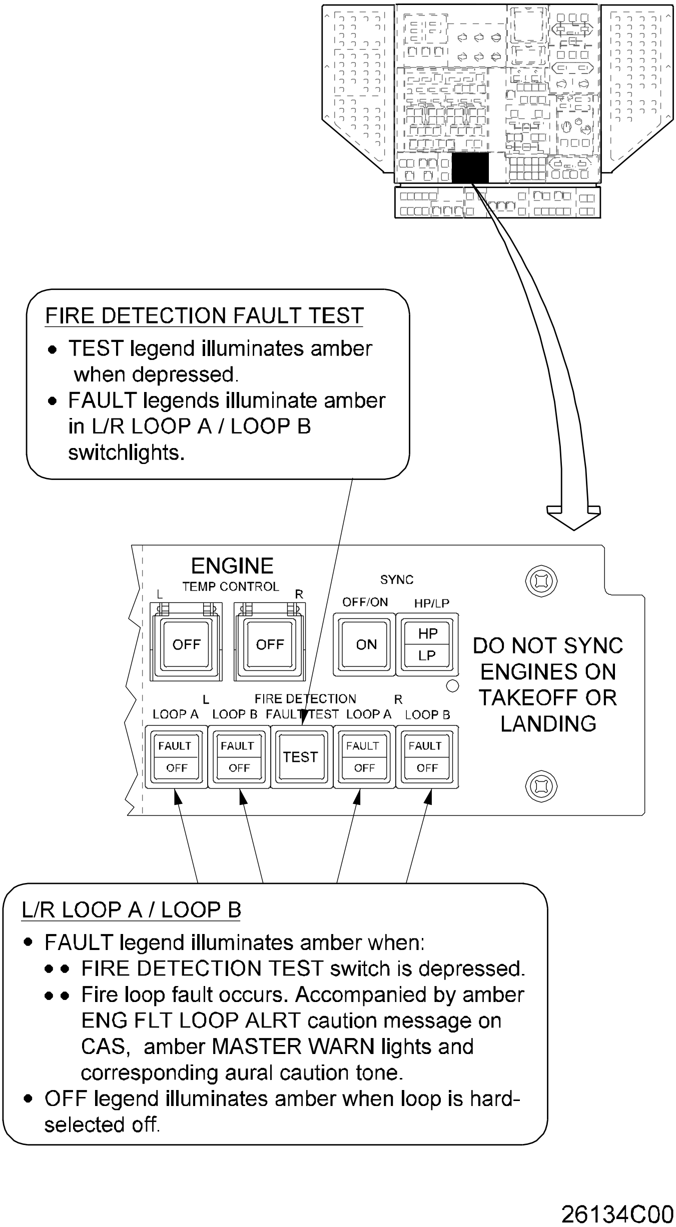
L ENG / R ENG FIRE TEST

With switchlight depressed and held:

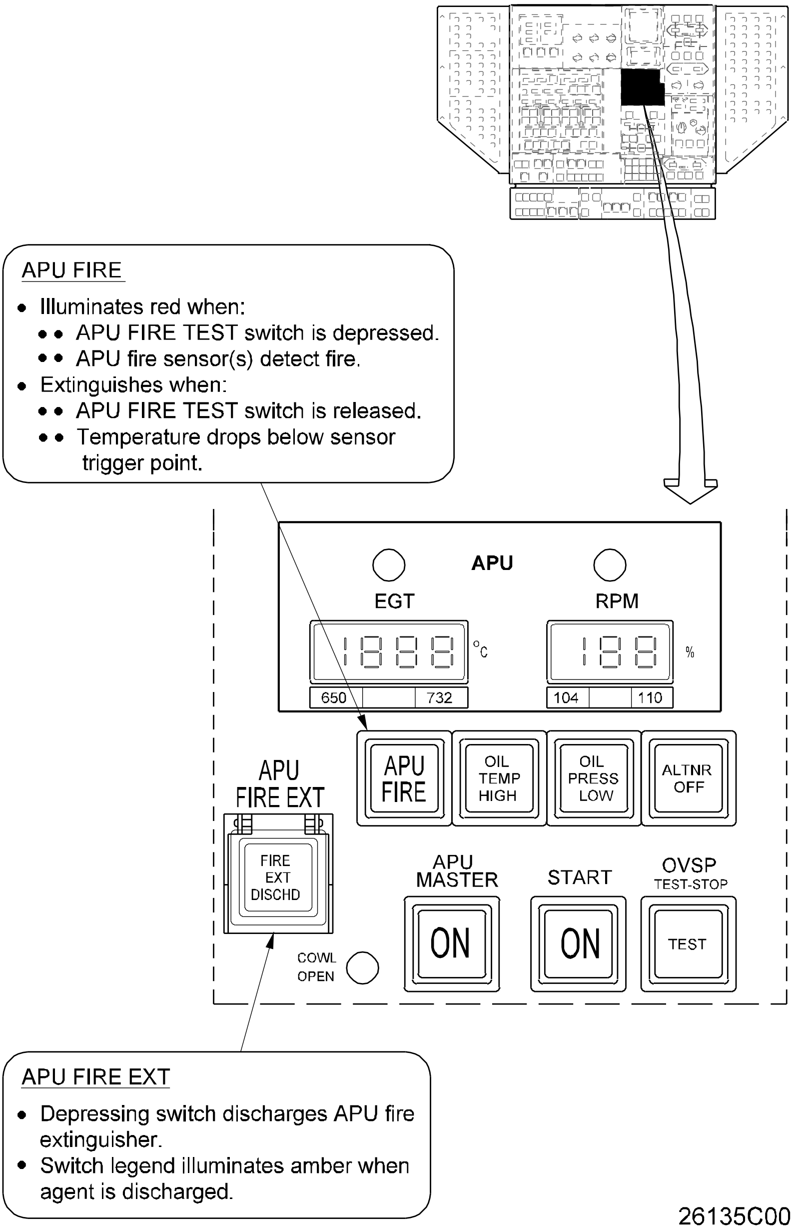
- Corresponding LOOP A and LOOP B segments of switchlight illuminates red.
- Corresponding FIRE handle illuminates red.
- Corresponding engine HP fuel cock illuminates red.
- Corresponding red ENGINE HOT and ENG FIRE LOOP ALRT and warning messages are displayed on CAS.
- Corresponding red ENGINE HOT and FIRE DET LOOP capsules illuminate on SWLP, if SWLP is installed and in manual mode.
- Both red MASTER WARN lights illuminate, with corresponding aural warning tone.
- ENGINE FIRE checklist is displayed on copilot's nav display (SN 1144 and SUBS; SN 1000-1143 with ASC 178).

26133C01

Engine / APU FIRE TEST Switches
Figure 4



Engine FIRE DETECTION TEST Switches
Figure 5



APU Fire Annunciator / APU FIRE EXT Switch
Figure 6

2A-26-40: Fire Extinguishing System

1. **General Description:**

The fire extinguishing system provides the flight crew with fixed and portable methods to suppress fire in each engine nacelle, the APU compartment or pressure vessel area. The following subsystems, units and components together compose this system:

- Engine Fire Extinguishing System
- APU Fire Extinguishing System
- Portable Fire Extinguishing System

The engine and APU fire extinguishing systems use Halon™1301 (CF_3Br) pressurized with gaseous nitrogen. A relatively nontoxic and noncorrosive agent, Halon™ 1301 works by chemically interfering with the combustion process. Both the engine and APU fire extinguishing systems remain fully operative down to and including battery-only power configuration.

Two portable fire extinguishers are installed in production standard aircraft. The cockpit fire extinguisher contains either Halon™ 1211 or Carbon Dioxide (CO_2). The cabin fire extinguisher is filled with water.

2. **Description of Subsystems, Units and Components:**

A. **Engine Fire Extinguishing System:**

The engine fire extinguishing system, shown simplified in Figure 7, is classified as a “two shot” system. It consists of a left and right fire bottle and associated agent discharge handle (rotary switch) for the aircraft engines. Power for the system is supplied from the 28V Essential DC bus.

(1) Fire Extinguishing Bottles:

Two identical, single-shot fire extinguishing bottles are mounted on the left and right sides in the tail compartment. Each single-shot bottle is loaded with 5.6 pounds by weight of Halon™ 1301 and is charged with nitrogen to 600 +25 / -0 psi at 70° F. Each bottle has two firing heads, each containing an electrically-fired dual-squib cartridge. Either head being fired discharges the extinguishing agent into the distribution system. The bottles are fired by switches activated by rotation of the FIRE handle to the left or right.

Each bottle contains a thermal discharge device. If excessive temperature builds up within a fire bottle, a frangible disc ruptures and the entire contents of the bottle is discharged into the tail compartment. If excessive pressure builds within up the bottle, a blowout disc in the thermal discharge port ruptures at 1600 ±200 psi, releasing the contents into the tail compartment.

Each bottle has an incorporated pressure gauge which can be viewed from outside the aircraft through a small window built into the skin adjacent to the bottles. The windows, located on each side of the fuselage under the pylon, are used to check bottle pressure during preflight inspections.

(2) Extinguishing Agent Plumbing:

The extinguishing agent plumbing routes extinguishing agent from the fire bottle to the engine cowling interior areas. Crossover lines and double-check tees provide the capability to discharge either

bottle to either engine or both bottles to one engine.

On each side, the line from the common side of the double-check tee is routed through the pylon into the engine fire zone. The agent is then distributed by means of a distribution line which runs fore and aft with dual nozzles at each end.

(3) Fire Handles:

Located on the forward portion of the cockpit center pedestal and shown in Figure 8, the FIRE handles are labeled L (for left) and R (for right). They are normally locked in the IN position. When a valid fire signal is provided by the engine fire detection system, an internal lock-release solenoid is automatically energized, allowing the associated FIRE handle to be pulled to the OUT position. A manual override button is also provided underneath each handle to override the lock-release solenoid, allowing the FIRE handle to be pulled to the OUT position at all times.

Pulling the FIRE handle OUT approximately ½ to ¾ inches causes the following to occur:

- Engine fuel shutoff valves will close
- Hydraulic shutoff valves will close
- Alternator will be shut off
- Thrust reverser will be disabled

Once in the OUT position, each FIRE handle is capable of rotation to two positions, labeled DISCH 1 and DISCH 2. Rotation of the handle to the appropriate position supplies 28 VDC power to the associated cartridge on the bottle. Application of power detonates the squib, freeing the extinguishing agent to flow to the affected engine. Discharge logic is shown in the following table:

FIRE HANDLE PULLED	ROTATED TO	DISCHARGES	INTO
L	DISCH 1	LEFT Fire Bottle	Left Engine Nacelle
	DISCH 2	RIGHT Fire Bottle	
R	DISCH 1	RIGHT Fire Bottle	Right Engine Nacelle
	DISCH 2	LEFT Fire Bottle	

NOTE:

Once a fire bottle has been discharged, it must be removed for refilling.

B. APU Fire Extinguishing System:

The APU fire extinguishing system, shown simplified in Figure 7, incorporates a single-shot bottle located in the tail compartment. The APU fire bottle is loaded with 2.5 pounds by weight of Halon™ 1301 and is charged with nitrogen to 600 +25 / -0 psi at 70° F. The firing head being is identical to those used on the engine fire bottles, except only one is installed on the APU fire bottle.

The APU fire bottle is located on the aft side of the APU compartment, under the utility hydraulic pump. A pressure gauge is installed on the bottle,

allowing pressure to be checked on preflight inspections. The tail compartment must be entered, however to gain access to the gauge. Like the engine fire bottles, pressure and thermal relief are provided. Also like the engine fire bottles, once discharged, the APU fire bottle cannot be refilled on the aircraft.

Firing the bottle is accomplished by depressing the APU FIRE EXT switch, located on the APU control panel on the cockpit overhead panel. Depressing the switch allows 28 VDC power to detonate the squib, freeing the extinguishing agent to flow through a single dedicated line to the APU compartment. The amber FIRE EXT DISCHD legend in the switch will then illuminate.

C. Portable Fire Extinguishing System:

The aircraft is equipped with two portable fire extinguishers (Figure 9) to aid the flight crew in combating different types of fires which may occur. The cockpit fire extinguisher contains either Halon™ 1211 or Carbon Dioxide (CO_2). The cabin fire extinguisher contains water.

The cockpit fire extinguisher is mounted behind the copilot's seat and is used to combat Class A, B and C fires. It is typically loaded with Halon™ 1211 and pressurized with nitrogen. Halon™ 1211 is a relatively nontoxic agent that leaves no residue.

Cockpit fire extinguishers loaded with CO_2 contain a safety disc to permit release of the agent through the horn in the event of excessive pressure.

The cabin fire extinguisher is typically located on the aft end of the radio rack. It is loaded with water and uses an incorporated CO_2 cartridge to expel the water. It should be used to combat only Class A fires. Twisting the handle punctures the cartridge and pressurizes the container, making the extinguisher ready for use. If inadvertently pressurized, the extinguisher should not be left in the aircraft. Rather, it should be removed from the aircraft, discharged and serviced in accordance with the GIV Aircraft Maintenance Manual.

3. Controls and Indications:

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
APU FIRE EXT	P	I-10	ESS DC Bus
FIRE EXT SHOT #1	P	G-8	ESS DC Bus
FIRE EXT SHOT #2	P	H-8	ESS DC Bus

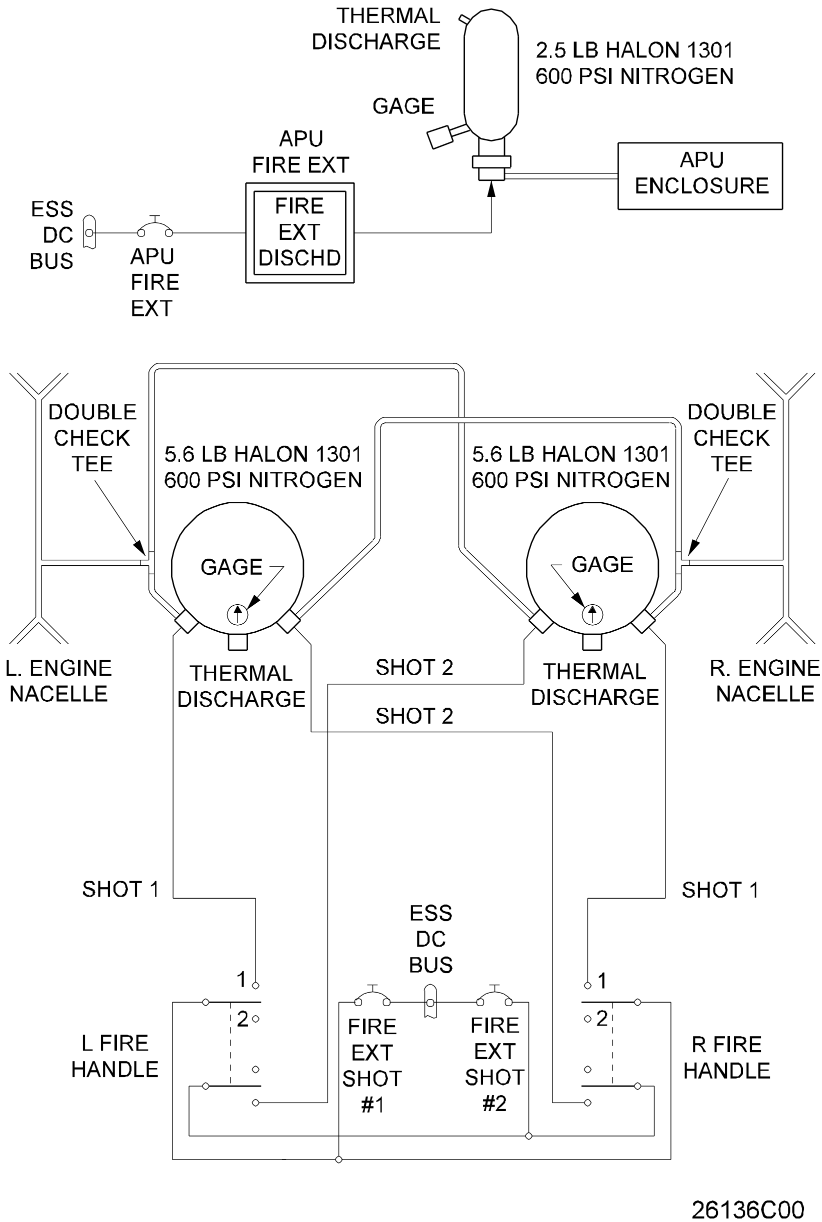
B. Messages and Annunciations:

Annunciation:	Cause or Meaning:
Amber FIRE EXT DISCHD legend illuminated on APU FIRE EXT switch on APU control panel.	APU fire extinguisher agent discharged.

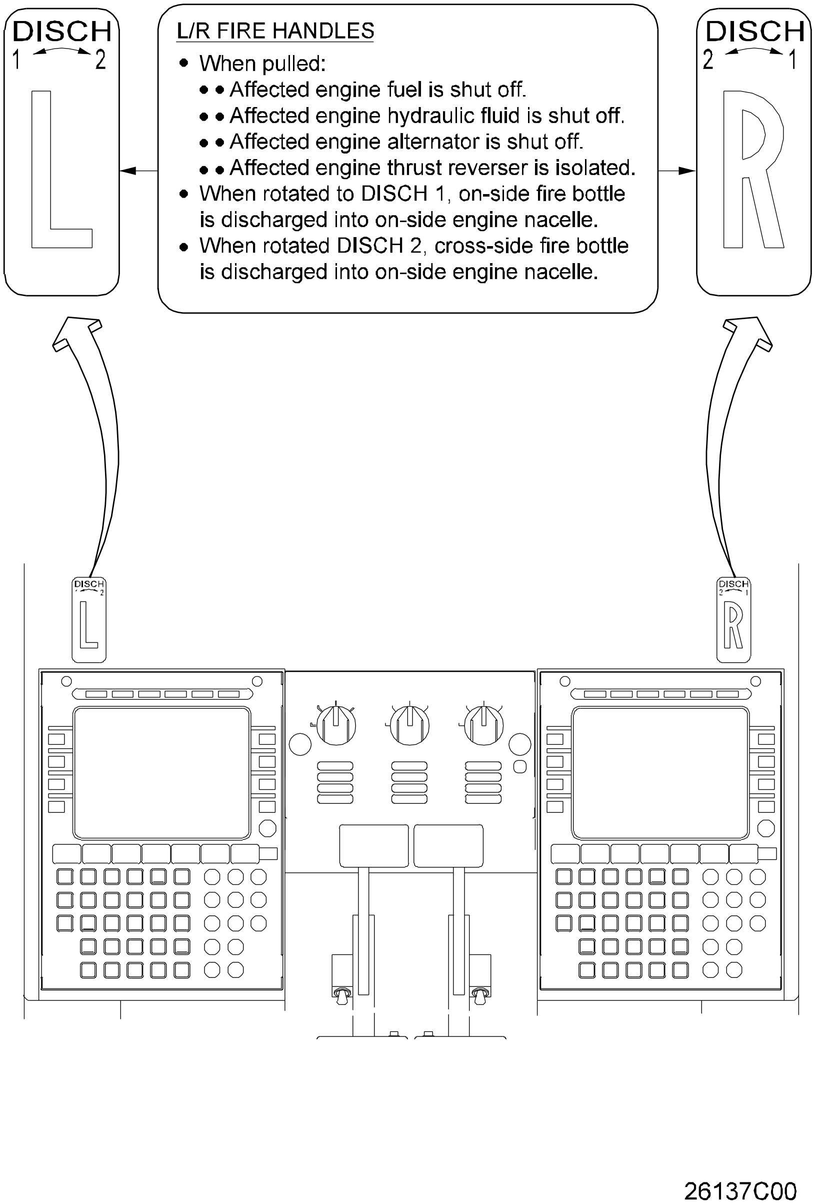
4. Limitations:

A. Flight Manual Limitations:

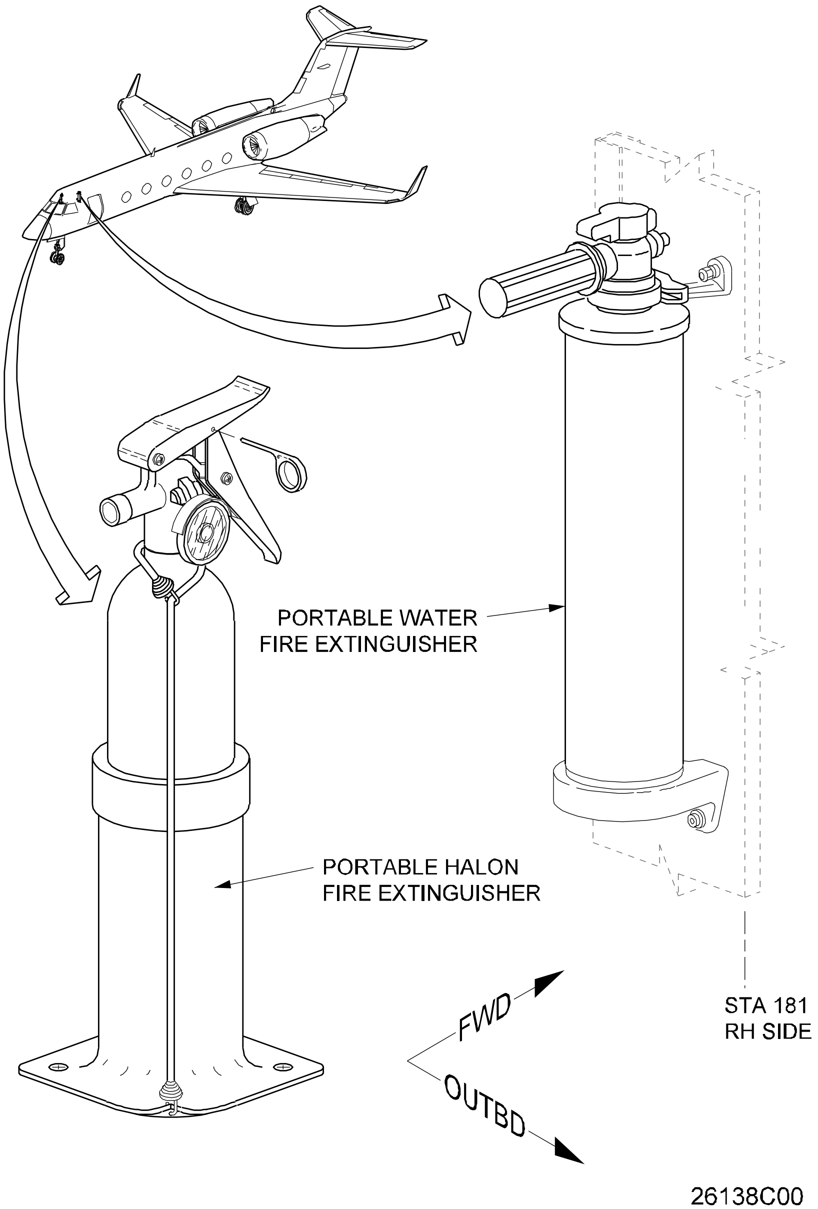
There are no limitations established at the time of this revision.



Engine / APU Fire Extinguishing System Simplified Block Diagram
Figure 7



Engine FIRE Handles
Figure 8



Portable Fire Extinguishers
Figure 9

2A-27-10: General

The Gulfstream IV primary flight controls system, shown in Figure 1, is a mechanically actuated, hydraulically operated system that provides boosted surface control to overcome the aerodynamic forces associated with high speed flight. This allows the aircraft to be comfortably and reliably steered through the pitch, roll and yaw axes.

The primary flight control surfaces (elevators, ailerons and rudder) are positioned by tandem type hydraulic actuators. The actuators receive hydraulic operating pressure from both the Combined and Flight hydraulic systems, as shown in Figure 2. Both hydraulic systems maintain a system pressure of 3000 psi. Loss of a single hydraulic system has no effect on operation of the primary flight controls, as the remaining system is capable of maintaining actuator load capacity. In the event of total loss of hydraulic pressure in both hydraulic systems, the primary flight controls revert to manual operation.

Mechanical pitch, roll and yaw trim systems allow the flight crew to trim the aircraft. The pitch trim system can also be controlled electrically by pitch trim switches on the control wheels.

A gust lock secures the elevators, ailerons and rudder to prevent wind gust damage to the surfaces.

Secondary flight controls, shown in Figure 1, include flaps, ground spoilers and speedbrakes. These flight controls are hydraulically powered and electrically or mechanically controlled. The mechanically operated horizontal stabilizer moves in conjunction with the flaps to maintain longitudinal trim.

An Angle-of-Attack (AOA) system provides outputs to the control column shakers, control column pusher, approach indexers, normalized AOA display and stall barrier system. The control column shakers provides early warning of a stall scenario by vibrating the control column before the stall while the control column pusher automatically initiates lowering the nose if the stall is imminent.

The Gulfstream IV uses an aircraft configuration warning system to monitor landing gear, flap, speed brake and power lever position. If an unsafe configuration is detected, the system provides a visual and / or aural warning.

On CAA certified aircraft, a flight control automatic failure detection system compares control inputs to actuator outputs. If a malfunction is detected, the system shuts off power to the affected actuator.

The flight controls system is divided into the following subsystems:

- 2A-27-20: Pitch Flight Control System
- 2A-27-30: Yaw Flight Control System
- 2A-27-40: Roll Flight Control System
- 2A-27-50: Horizontal Stabilizer System
- 2A-27-60: Flaps System
- 2A-27-70: Spoiler System
- 2A-27-80: Gust Lock System

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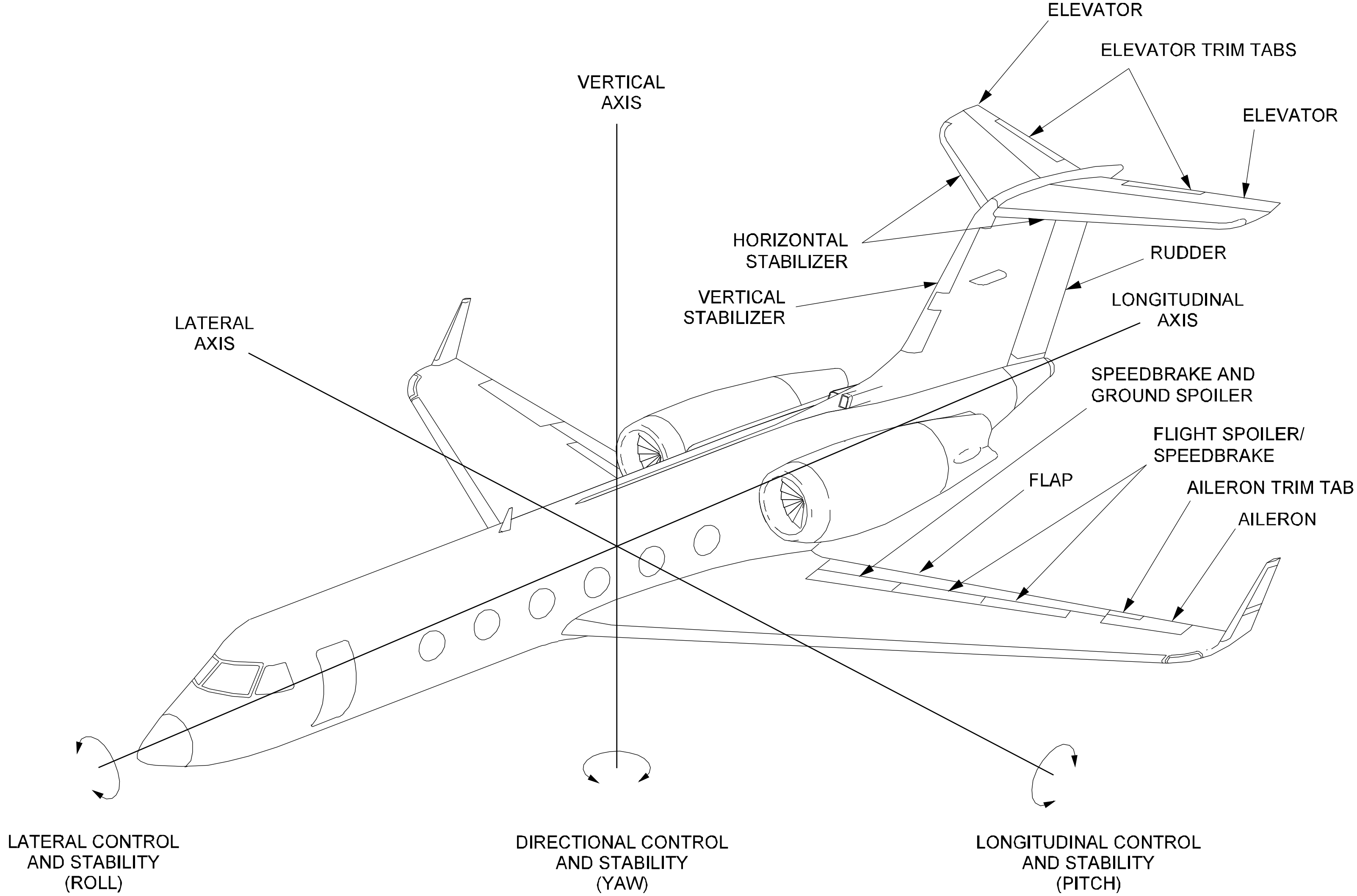
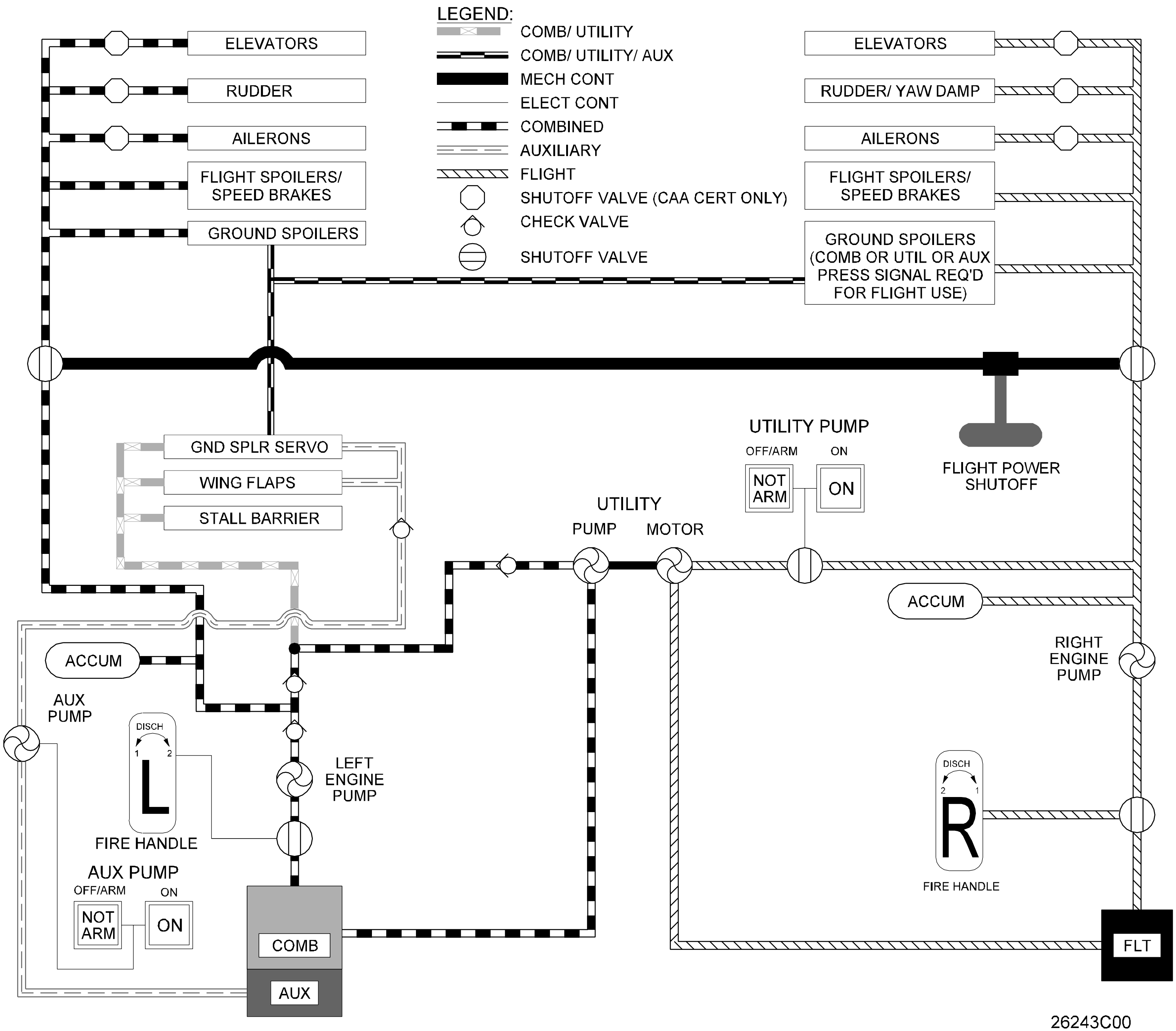


GIV Flight Controls / Aerodynamic Axes
Figure 1



GIV Flight Controls Fluid Power Diagram
Figure 2

2A-27-20: Pitch Flight Control System

1. General Description:

Aircraft movement about the lateral axis (pitch) is controlled by the position of the elevators. The elevators are manually controlled, mechanically actuated and hydraulically boosted airfoils mounted on the trailing edge of the horizontal stabilizer. Total elevator travel ranges from 24° trailing edge up to 13° trailing edge down.

The elevators are positioned by a tandem type hydraulic actuator. The actuator receives hydraulic operating pressure simultaneously from both the Combined and Flight hydraulic systems during normal operations. Loss of a single hydraulic system has no effect on operation of the elevators, as the remaining system is capable of maintaining actuator load capacity. In the event of total loss of hydraulic pressure in both systems, the elevators revert to manual operation. Manual reversion is also possible through use of a flight power shutoff valve and its pedestal-mounted control handle.

A pitch trim system is used to position a trim tab attached to the trailing edge of each elevator. The tabs are positioned either manually by a pedestal-mounted control wheel or electrically by pitch trim switches on the control wheels.

An Angle-of-Attack (AOA) system provides outputs to the control column shakers, control column pusher, approach indexers, normalized AOA display and stall barrier system. The control column shakers provides early warning of a stall scenario by vibrating the control column before the stall while the control column pusher automatically initiates lowering the nose if the stall is imminent.

On CAA certified aircraft, a flight control automatic failure detection system compares control column inputs to elevator actuator outputs. If a malfunction is detected, the system shuts off either or both hydraulic power sources to the affected actuator.

The pitch flight control system consists of the following subsystems, units and components:

- Control column system
- Mechanical actuation system
- Hydraulic boost system
- Manual reversion system
- Pitch trim system
- Angle-of-attack / stall barrier system
- Failure detection system (CAA aircraft only)

2. Description of Subsystems, Units and Components:

A. Control Column System:

(See Figure 7.)

The pilot's and copilot's control columns mount to a common transverse torque tube supported by a bearing on each end. Moving the control column fore and aft rotates the torque tube that, in turn, transmits the inputs rearward through conventional mechanical linkage. Adjustable stops on the torque tube limit control column movement to eight inches aft and five inches forward of the neutral position.

An eddy current damper is connected to the bottom of the control column

output crank. The damper is designed to sense control column movement and provide a counteracting damping force or artificial feel. Damping force is generated in proportion to control column velocity. Should the eddy current damper jam or fail, its internal clutch releases to allow control column movement.

B. Mechanical Actuation System:

(See Figure 3.)

Control column inputs are transmitted rearward through a system of push-pull rods, bellcranks, cables and a sector assembly to an input cable sector below the base of the vertical stabilizer in the tail compartment. Connected to the input cable sector are the elevator actuator, autopilot servo and stability springs. Final output of the sector is to the elevator actuator input lever.

Movement of the actuator input lever displaces the elevator actuator's servo control valve, directing Combined and Flight hydraulic system pressure to the actuator cylinders. When under pressure, the actuator body moves while the its piston remains motionless. Movement of the actuator body in turn moves its output crank. Output crank movement is transmitted upward and aft to the left and right elevators through a system of push-pull rods and cranks, resulting in the desired elevator deflection.

Stability springs, commonly referred to as down springs, provide approximately 13 pounds of pull on the control columns. This pulling force keeps pilot feel forces out of the friction band at low airspeeds.

C. Hydraulic Boost System:

The elevator actuator is a dual tandem actuator consisting of two pistons secured to a common shaft. The pistons move inside a common cylinder divided to create two separate cylinders. One cylinder receives Flight hydraulic system pressure while the other cylinder receives Combined hydraulic system pressure.

Mechanical movement of the actuator input lever moves the servo control valve from its neutral position. The servo control valve then directs hydraulic pressure to one of the actuator's two cylinders and connects each cylinder's opposite side to return. When the elevator reaches the desired deflection, the servo control valve shifts to its neutral position to lock hydraulic pressure within the actuator, in effect preventing further surface movement.

The elevator actuator also has an internal hydraulic damper that provides damping force proportional to the square of its input velocity. This damping action ensures operational stability for the elevators when hydraulically boosted whereby a portion of the actuator's output is fed back to the input system.

D. Manual Reversion System:

During normal flight operations, the Combined and Flight hydraulic systems each supply and maintain 3000 psi to the elevator actuator. Loss of system pressure due to a single system failure has no effect on operation of the pitch flight control system.

Loss of system pressure from both hydraulic systems will automatically revert the pitch flight control system to manual control. As pressure at the

actuator drops below 60 psi, bypass valves within the actuator open to allow the actuator piston to idle. With the actuator piston idling, the system is said to be in manual reversion, technically explained in the following paragraph.

The actuator input crank and output crank rotate on a common pivot point. They are linked by a pin and elongated slot arrangement. The input crank is on the pin and the output crank is on the slot. With the actuator piston idling, the pin moves within the slot until it reaches either end. At this point, mechanical contact is made and the input crank now drives the output crank. The output crank in turn drives the left and right elevators through its push-pull rods and cranks.

Manual reversion of the pitch flight control system is also possible by closing a normally open flight power shutoff valve. The flight power shutoff valve is a mechanically operated shutoff valve located between the Combined and Flight hydraulic system pressure sources and the elevator actuator (as well as the aileron, rudder and flight / ground spoiler actuator) pressure lines. The valve consists of two mechanically connected but hydraulically isolated sections. A controlex cable connects the valve to a FLIGHT POWER SHUT OFF handle located on the left aft side of the cockpit center pedestal. See Figure 8.

Moving the FLIGHT POWER SHUT OFF handle up from its stowed (horizontal) position to the vertical position mechanically closes the flight power shutoff valve. With the valve closed, operating pressure is removed from the actuator, allowing the piston to idle.

The resultant advantage of the flight power shutoff provision is the ability to bypass a malfunctioning actuator (such as would be the need in the unlikely event of an actuator jam) and manually fly the aircraft. Although control column effort and response time to inputs are increased while in manual reversion, the aircraft remains capable of positive and harmonious control.

E. Pitch Trim System:

(See Figure 5 and Figure 9.)

(1) Elevator Trim Tabs:

A trim tab is installed on the trailing edge of each elevator. The tabs are mechanically positioned through cable-driven drum actuators located in each elevator. The trim actuators can be operated manually or electrically as described in the following paragraphs. Elevator trim tab travel ranges from 22 ±1° tab trailing edge down (aircraft nose up) to 8 ±1° tab trailing edge up (aircraft nose down).

(2) Manual Trim Control:

Manual control of pitch trim is accomplished by an interconnected manual trim control wheel set. A trim control wheel and elevator trim scale are provided on each side of the cockpit center pedestal. With electric pitch trim disengaged, moving either manual trim control wheel adjusts pitch trim to the desired setting; the opposite wheel moves in unison. With electric pitch trim engaged, both manual trim control wheels move in unison corresponding to the amount of electric pitch trim movement. Each elevator trim scale range is incremented to a maximum of 22 units aircraft nose-up (22 ±1° tab

trailing edge down) and 8 units aircraft nose-down (8 ±1° tab trailing edge up).

Mechanical stops limit elevator trim wheel movement to 6.6 turns from each stop. A shear rivet installed in the cockpit portion of the system prevents application of excessive force by shearing at approximately 31 pounds of force.

(3) Electric Pitch Trim:

Located on the pilot's flight panel, the PITCH TRIM ENG / DISENG switch engages or disengages the electric pitch trim. Pitch trim is also engaged whenever the autopilot is engaged. With electric pitch trim engaged (amber DISEN switch legend extinguished), pitch trim can be adjusted through use of a split-half pitch trim switch (sometimes referred to as a "beep" switch) installed on the outboard grip of each control wheel. Switch positions are labeled NOSE DOWN and NOSE UP. Inadvertent actuation of pitch trim, including runaway, is minimized through the split-half switch design. In order for the pitch trim to be actuated, both halves of the switch must be simultaneously moved in the same direction.

Movement of the electric pitch trim switch to NOSE DOWN or NOSE UP actuates the autopilot elevator trim servo. The trim servo is connected to the manual trim control wheel set by a chain. The chain-driven movement of the manual trim control wheel set in turn positions the elevator trim tabs.

The electric pitch trim is normally checked by the flight crew on the first flight of the day, during the Before Starting Engines checklist. A check usually consists of running the elevator trim fully up, then fully down, using normal methods, i.e., using both halves of the switch simultaneously. This is followed by attempting to run the pitch trim using each half of the switch alone. Any movement resulting from using either half of the switch alone indicates a malfunction that should be corrected before flight. The check is concluded by setting pitch trim for the takeoff Center of Gravity (CG) condition as determined using the Airplane Flight Manual.

(4) Elevator Trim Tab Actuator Heat System:

Electrically-heated elevator trim tab actuators are incorporated on airplanes SN 1380 and subsequent and SN 1000 through 1379 having ASC 342. These actuators are designed to alleviate frozen or stiff trim tab actuators possible in extreme cold temperatures. The system receives power from Phase C of the Left Main AC bus. Operation of the system is automatic and transparent to the flight crew.

F. Angle-of-Attack / Stall Barrier System:

(See Figure 4, Figure 10 and Figure 11.)

While in flight, the Angle-of-Attack (AOA) system monitors aircraft AOA to provide warnings of an approaching stall. If AOA continues to increase toward aerodynamic stall, the system applies a nose down control input through the stall barrier system.

The AOA / stall barrier system consists of the following units and components:

- AOA probes
- AOA display and approach indexer
- Stall warning computers
- Pilot and copilot control column shaker motors
- Stall barrier system

AOA becomes fully functional as the aircraft becomes airborne, i.e., when the nutcracker shifts to the AIR mode. The control column shaker, however, is disabled for the first five seconds following rotation to eliminate nuisance activity. Pulling the SHAKER #1 and / or SHAKER #2 circuit breakers, as appropriate, is the only way to disable a control column shaker in flight; however, such action completely disables the associated stall warning computer(s). System design is such that either stall warning computer is capable of operating the control column shaker and control column pusher should the other computer become disabled.

(1) AOA Probes:

An AOA probe / transducer assembly is installed on the left and right forward fuselage. The cone-shaped probes freely rotate in the airstream to provide AOA reference data for the stall warning / stall barrier systems and AOA display data for the flight crew. The left AOA probe provides data the No. 1 stall warning computer while the right AOA probe provides data the No. 2 stall warning computer. Heating elements prevent ice accumulation on the probe and condensation within the transducer case.

(2) AOA Display and Approach Indexer:

AOA display data supplied by the probes includes the normalized AOA display and the approach indexer. Normalized AOA display is shown on the lower left portion of the Primary Flight Display (PFD) and consists of a vertical scale marked from 0.2 to 1.1 in 0.1 increments. At the bottom of the scale is a three-digit display surrounded by a pointer that provides AOA indication within a 0.01 resolution. As AOA changes, the display / pointer moves up and down to correspond with the indication on the scale.

An AOA approach indexer on either side of the windshield center post indicates the optimum AOA for approach and landing. The No. 1 AOA system drives the pilot's indexer while the No. 2 AOA system drives the copilot's indexer. During approach and landing, the AOA system illuminates each indexer's red chevron if AOA is too high, a green circle if AOA is correct or an amber chevron if AOA is too low.

(3) Stall Warning Computers:

The No. 1 and No. 2 stall warning computers receive the following inputs and then provide outputs to the control column shaker motors and stall barrier system:

- AOA reference data from the associated probes
- Altitude data from the DADCs
- Nutcracker mode from the nutcracker relays
- Flaps position from the 39° flap relay

(4) Control Column Shaker Motors:

A control column shaker motor is attached to the pilot and copilot control columns. When activated by a stall warning computer, the motor drives an off-center weight that vibrates the control column. Activation of one motor affects both control columns due to their mechanical interconnection.

(5) Stall Barrier System:

A stall barrier system (control column pusher) is incorporated in the pitch flight control system to prevent a stall by forcing the control columns forward when the flight crew fails to respond to either visual indications or to the control column vibrations that warn of impending stall. The system consists of two normally closed stall barrier valves and an actuating cylinder that is mechanically linked to the elevator actuator input sector. One valve receives signals from the No. 1 stall warning computer while the other valve receives signals from the No. 2 stall warning computer. If one system fails, the remaining system is capable of operating the system.

When a high AOA is reached, the control column shaker motors are activated. When a more severe AOA is reached, the control column pusher trip detector activates its respective stall barrier valve. The activation signal will originate from whichever system is operating - No. 1, No. 2 or both. When a stall barrier valve is activated, Combined (or Utility) hydraulic system pressure is ported to the extend side of the stall barrier actuating cylinder. As the cylinder extends, it applies an input to the elevator actuator input sector. This input causes the elevator actuator to drive the elevator trailing edges down; the control column drives forward accordingly, to approximately one inch forward of neutral. When AOA has decreased more than one degree, the stall barrier system disengages.

The force generated by the stall barrier system is sufficient to overcome any autopilot force, however, the system can be manually overcome by the flight crew.

The stall barrier system can be deactivated by pressing the BARR DISC button on either control wheel. The BARR DISC button also serves as the autopilot disconnect button, thus is also labeled A/P DISC accordingly. Deactivation of the stall barrier system is also possible through selection of the STALL BARRIER switchlight to OFF. The switchlight is located on the cockpit center pedestal just below the left HP fuel cock. An amber OFF legend in the switchlight will illuminate when the system is deactivated and will extinguish when activated.

(6) Stall Warning / Stall Barrier System Test:

The stall warning / stall barrier system is normally tested by the flight crew on the first flight of the day or every eight hours of flight time. The test is performed only on the ground and cannot be tested in flight. It consists of the following steps:

(a) Select the STALL BARRIER switch to on. Verify amber OFF legend is extinguished.

(b) On both the pilot's and copilot's display controllers, depress the TEST function key.

(c) On both the pilot's and copilot's display controllers, simultaneously depress and hold the Sea Level (S/L) line select key.

(d) Continue holding both S/L line select keys until the normalized AOA indicator pointer slews to full scale and observe the following:

- Stall warning (control column shaker) occurs between 0.70 and 0.80
- Stall barrier (control column pusher) occurs between 0.95 and 1.07
- Check that the BARR DISC button will override the pusher

(e) On both the pilot's and copilot's display controllers, simultaneously depress and hold the ALT line select key.

(f) Continue holding both ALT line select keys until the normalized AOA indicator pointer slews to full scale and observe the following:

- Stall warning (control column shaker) occurs between 0.54 and 0.65
- Stall barrier (control column pusher) occurs between 0.79 and 0.90
- Check that the BARR DISC button will override the pusher

NOTE:

Both pilot's and copilot's sides have to be tested simultaneously in order to activate the control column pusher.

NOTE:

Another momentary push of the TEST function key may be required to ensure the AOA indicator is in the normal area prior to takeoff.

G. Failure Detection System:

CAA Certified Aircraft Only: A flight control automatic failure detection system monitors flight control inputs from the control columns and compares them to the elevator actuator outputs. If the system detects a failure, it automatically shuts off hydraulic pressure to the actuator and triggers the appropriate warning on the Crew Alerting System (CAS). Once activated by a malfunction, hydraulic pressure is inhibited until power to the respective monitoring system is interrupted, for instance, by pulling and resetting the appropriate circuit breaker.

The monitoring system is a dual-channel system. One channel controls the Combined hydraulic system pressure source while the other controls the Flight hydraulic system pressure source. Power for the system is received from the 28 VDC Essential DC bus.

A pair of limit switches monitor applied control column input while a pair of

reed switches monitor actuator output in response to the input.

If a disagreement occurs between control column input and actuator output, the associated limit switch and reed switch close to complete a circuit to the respective hydraulic shutoff delay relay. If the relay remains energized for more than ½ second, it energizes the respective hydraulic shutoff control relay. The control relay in turn powers its hydraulic shutoff valve to the closed position. Activation of the shutoff valve also causes an amber EL CMB HYD OFF (or EL FLT HYD OFF) message to be displayed on CAS.

3. Controls and Indications:

(See Figure 6 and Figure 7 through Figure 9.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
SHAKER #1	CPO	A-10	Essential DC Bus
SHAKER #2	CPO	B-10	R Main DC Bus
STALL BARR DUMP VALVE	CPO	A-8	Essential DC Bus
STALL BARR VALVE #1	CPO	A-12	Essential DC Bus
STALL BARR VALVE #2	CPO	B-12	R Main DC Bus
STALL BARRIER #1	CPO	A-9	Essential DC Bus
STALL BARRIER #2	CPO	B-9	R Main DC Bus
STALL WARN CMPTR #1	CPO	A-11	Essential DC Bus
STALL WARN CMPTR #2	CPO	B-11	R Main DC Bus
ELEV COMB HYD S/O (1)	CPO	B-15	Essential DC Bus
ELEV FLT HYD S/O (1)	CPO	A-15	Essential DC Bus
ELEV TRIM TAB ACTR HTR (2)	CP	L-9	L Main AC Bus, ϕC

NOTE(S):

(1) CAA certified aircraft only.

(2) SN 1380 & subs; SN 1000 - 1379 having ASC 342.

B. Warning (Red) Messages and Annunciations:

Annunciation:	Cause or Meaning:
Red chevron illuminated on pilot's / copilot's AOA indexer.	AOA for approach and landing is too high.

C. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
AOA HEAT 1-2 FAIL	Angle of attack probe heater failed.
EL CMB HYD OFF (1)	The flight control automatic failure detection system has shut off Combined hydraulic system pressure to the elevator actuator.
EL FLT HYD OFF (1)	The flight control automatic failure detection system has shut off Flight hydraulic system pressure to the elevator actuator.
EL MISTRIM NOSE UP/DN	Autopilot elevator trim out of trim in direction indicated.

CAS Message:	Cause or Meaning:
MACH TRIM LIMIT	Elevator trim has reached electrical trim limit while operating airplane in Mach Trim speed region (greater than 0.80 Mach).
MACH TRIM OFF	PITCH TRIM switch selected OFF or electric pitch trim has failed. (This message is inhibited at less than 0.82 Mach.)
STALL BARRIER 1-2	Stall barrier system giving stall angle indication.
STALL BARR 1-2 FAIL	Stall barrier failed. It is normal for STALL BARR 1 FAIL message to be displayed any time EMERGENCY FLAPS are used and flaps position is greater than 22°.
STALL BARRIER OFF	STALL BARRIER switch is OFF or system not powered.
TRIM LIMIT	Autopilot elevator trim has reached electrical trim limits.

NOTE(S):

(1) CAA certified aircraft only.

Annunciation:	Cause or Meaning:
Amber chevron illuminated on pilot's / copilot's AOA indexer.	AOA for approach and landing is too low.

4. Limitations:

A. Angle-of-Attack (AOA) System:

(1) Use As A Reference:

Angle-of-Attack (AOA) may be used as reference, but does not replace airspeed as the primary reference.

(2) Indication Parameters:

AOA indication must be within white band once forward airspeed is attained during takeoff roll.

(3) Use As A Speed Reference:

AOA shall not be used as a speed reference for takeoff rotation.

B. Stall Barrier / Stall Warning:

(1) Takeoff Requirements:

There are two stall barrier / stall warning systems installed on the airplane. Dispatch with one stall barrier / stall warning system inoperative is allowed with reference to the MEL.

(2) Use of System:

Stall barrier systems must be ON during all flight operations except as noted in Section 05-15-40, Stall Barrier Malfunction. Refer to this system description for a description of the stall warning / stall barrier system checkout procedure.

C. Mach Trim Compensation / Electric Elevator Trim:

(1) Use of mach trim compensation:

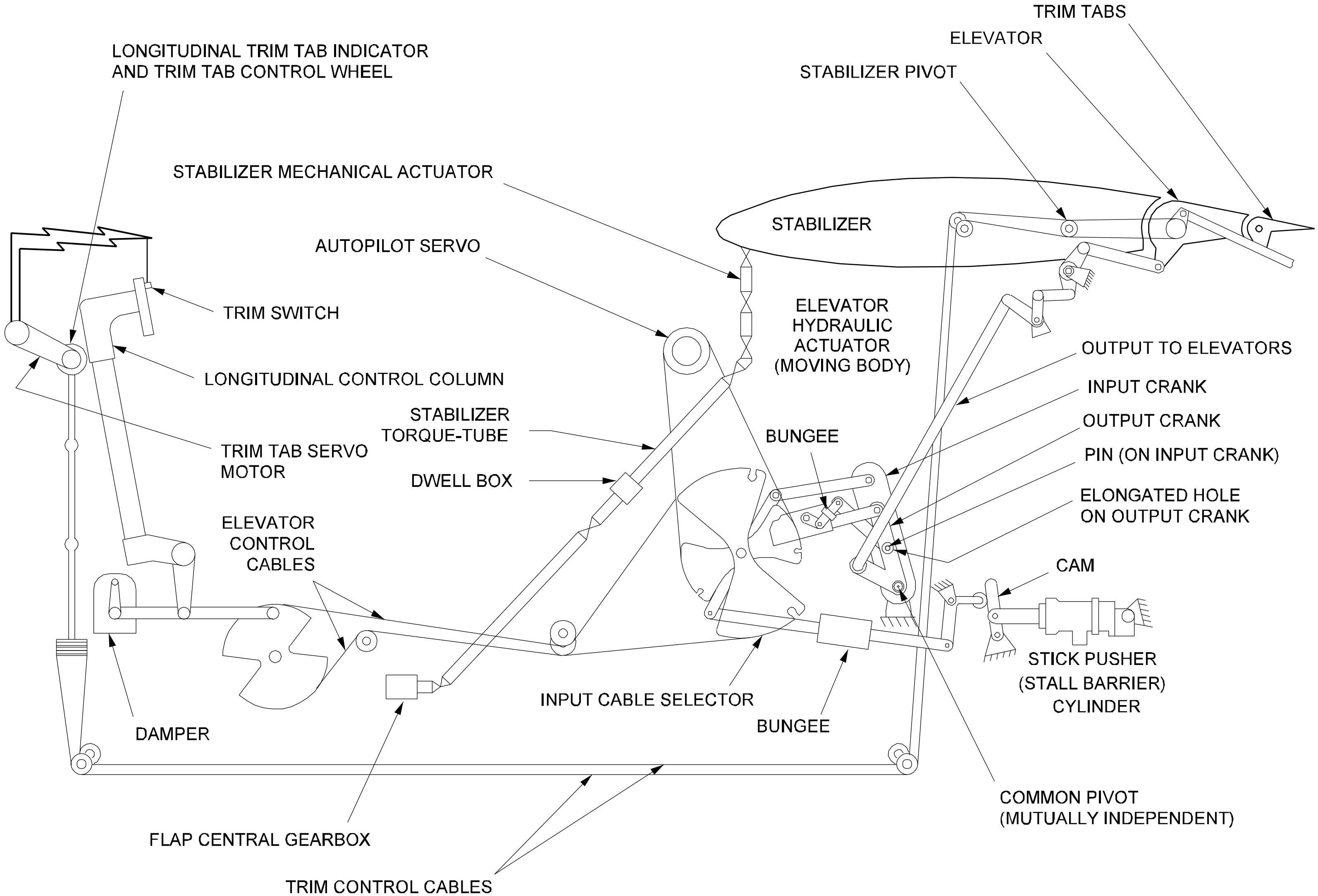
Mach trim compensation must be ON during all flight operations except as provided for in Section 05-03-40, Mach Trim Compensation Failure.

(2) If mach trim compensation failure is coupled with yaw damper failure:

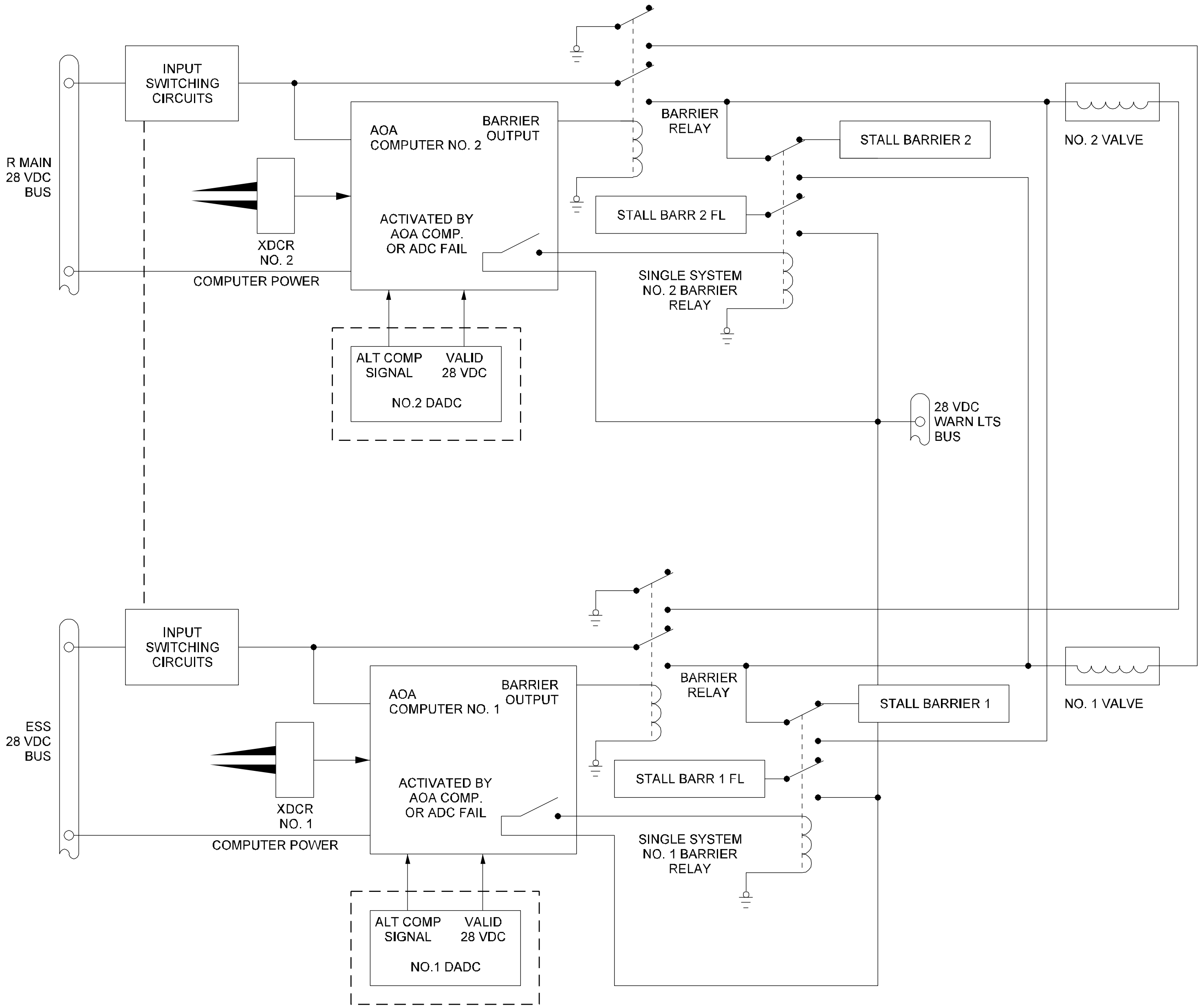
When mach trim compensation failure is coupled with yaw damper failure, observe speed limitations for both failures and limit altitude to 41,000 ft.

D. Mach Trim / Electric Elevator Trim Inoperative Speed:

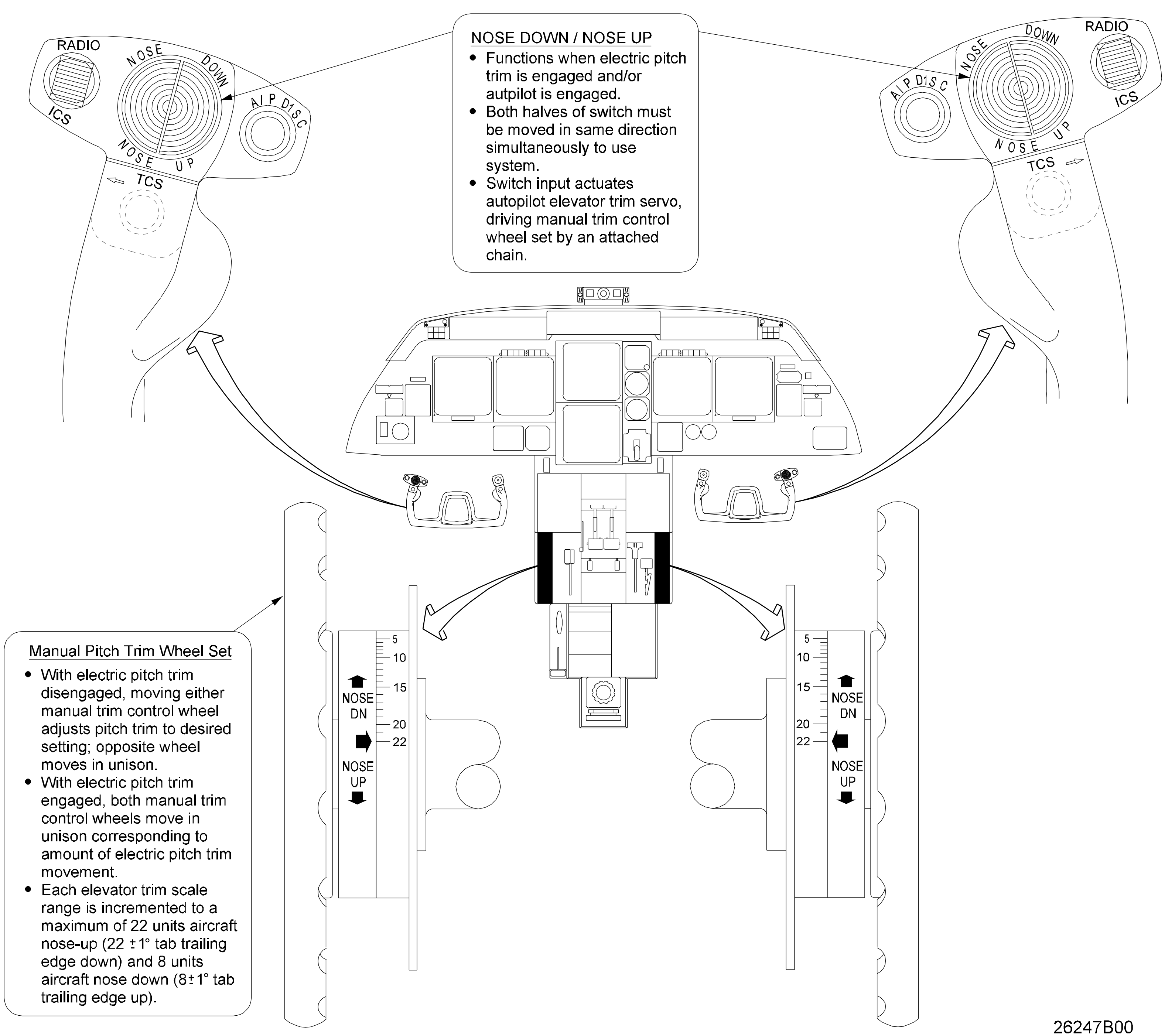
With both mach trim compensators inoperative or electric elevator trim inoperative, the maximum operating limit speed is 0.75 MT.



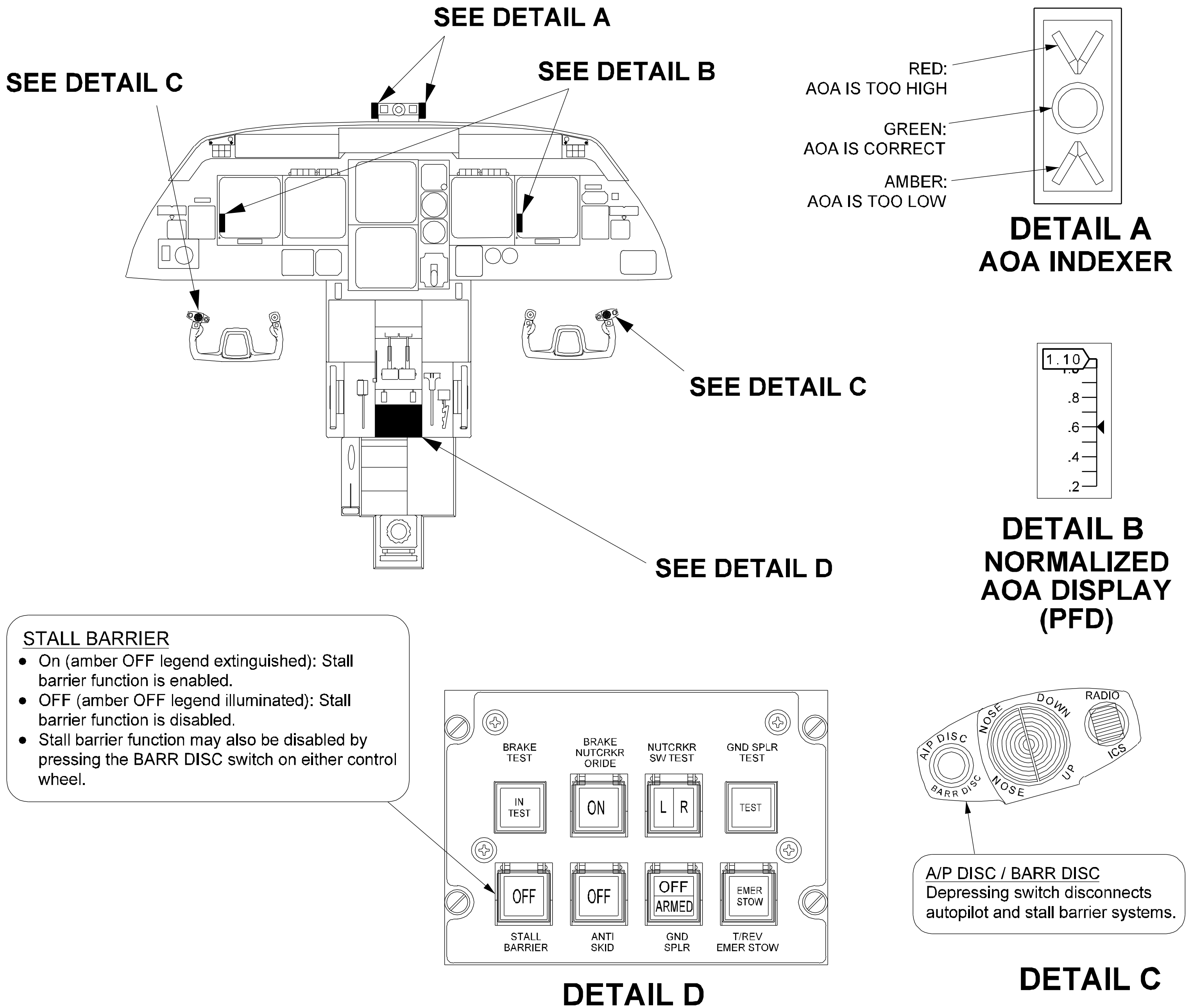
Pitch Flight Control
System Simplified Block
Diagram
Figure 3



Stall Barrier / Angle of Attack Wiring Schematic
Figure 4



Pitch Trim Controls
Figure 5



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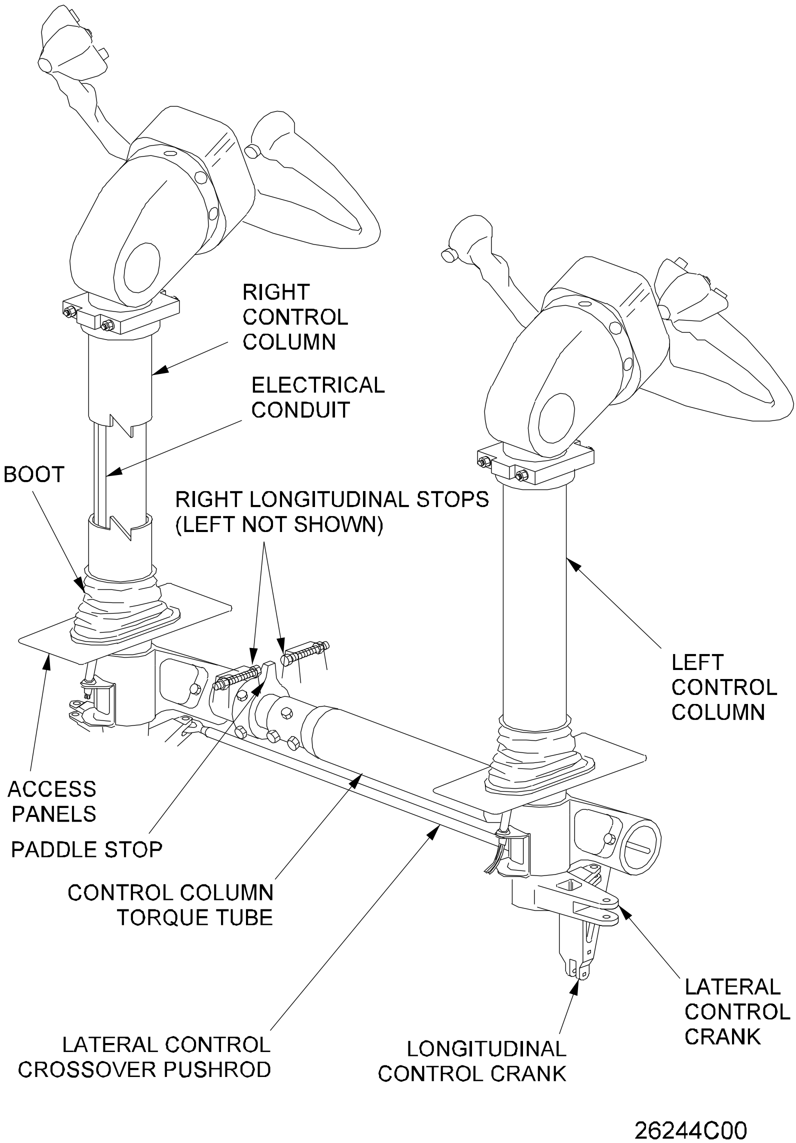
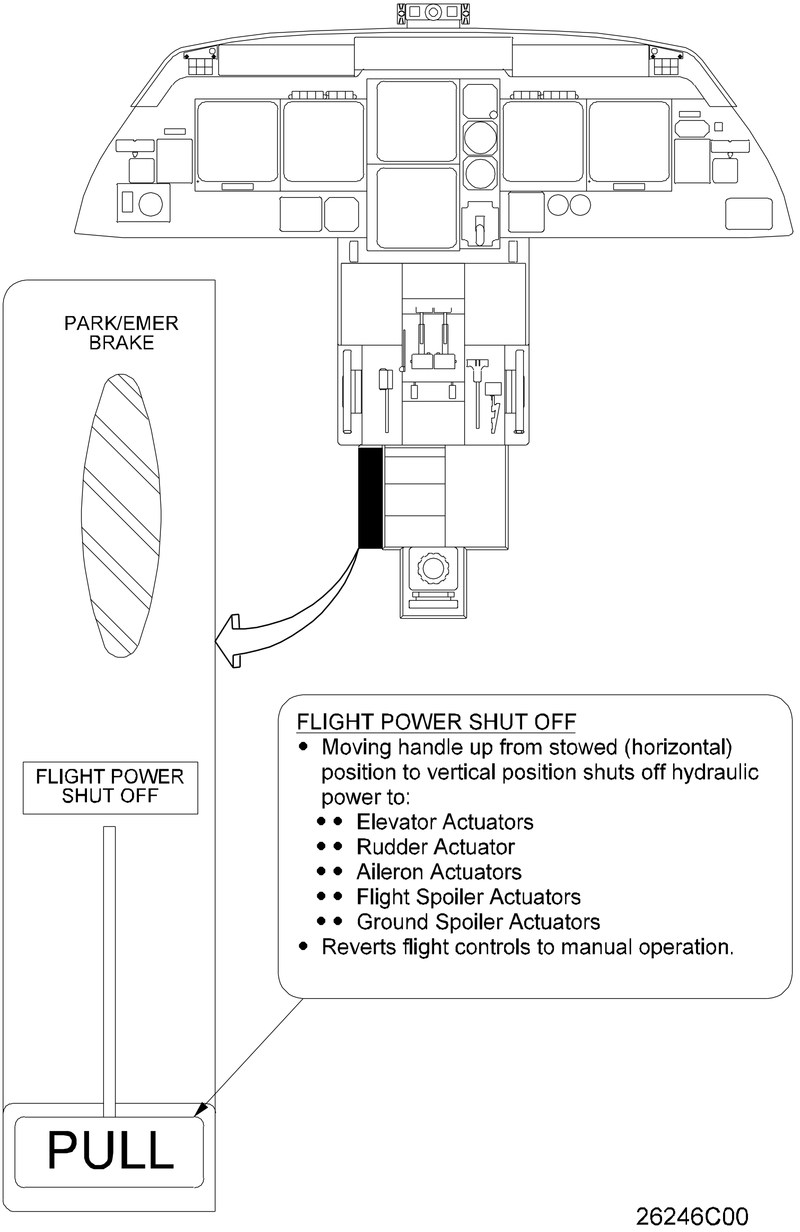
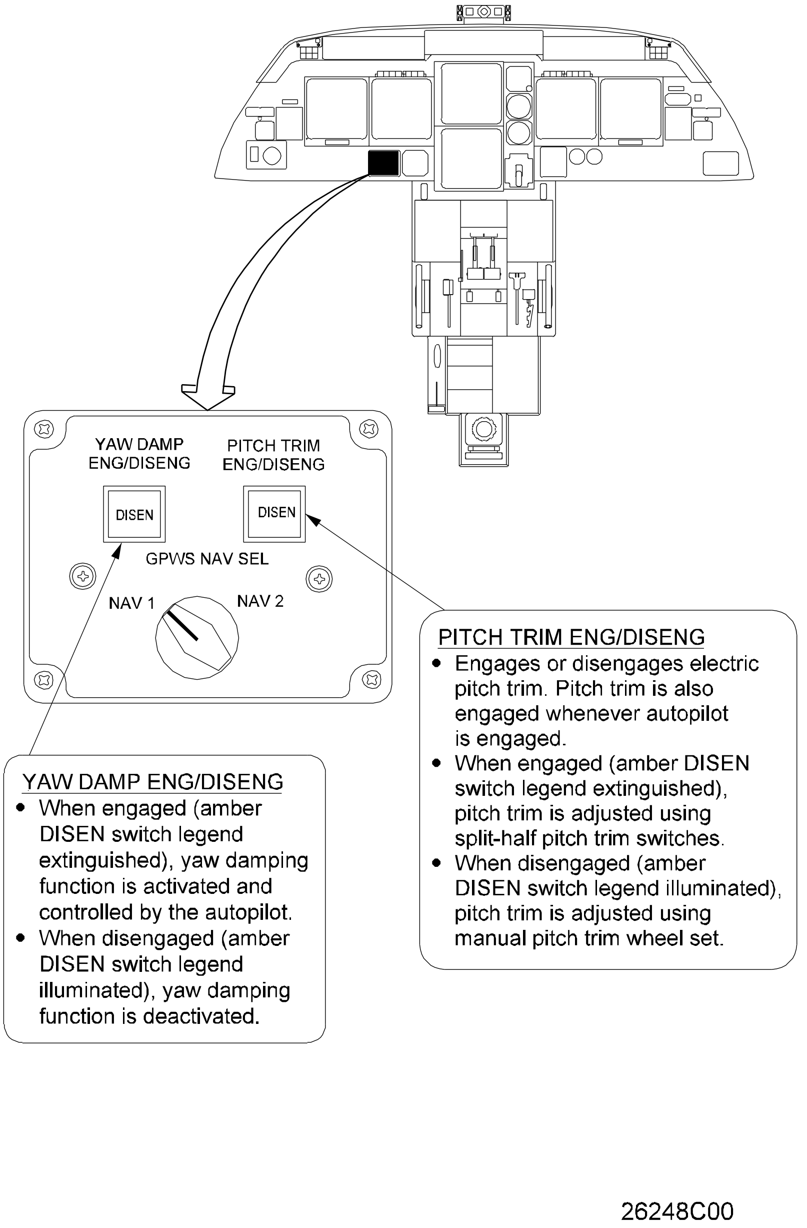
Stall Barrier / Angle of Attack Controls and Indications
Figure 6



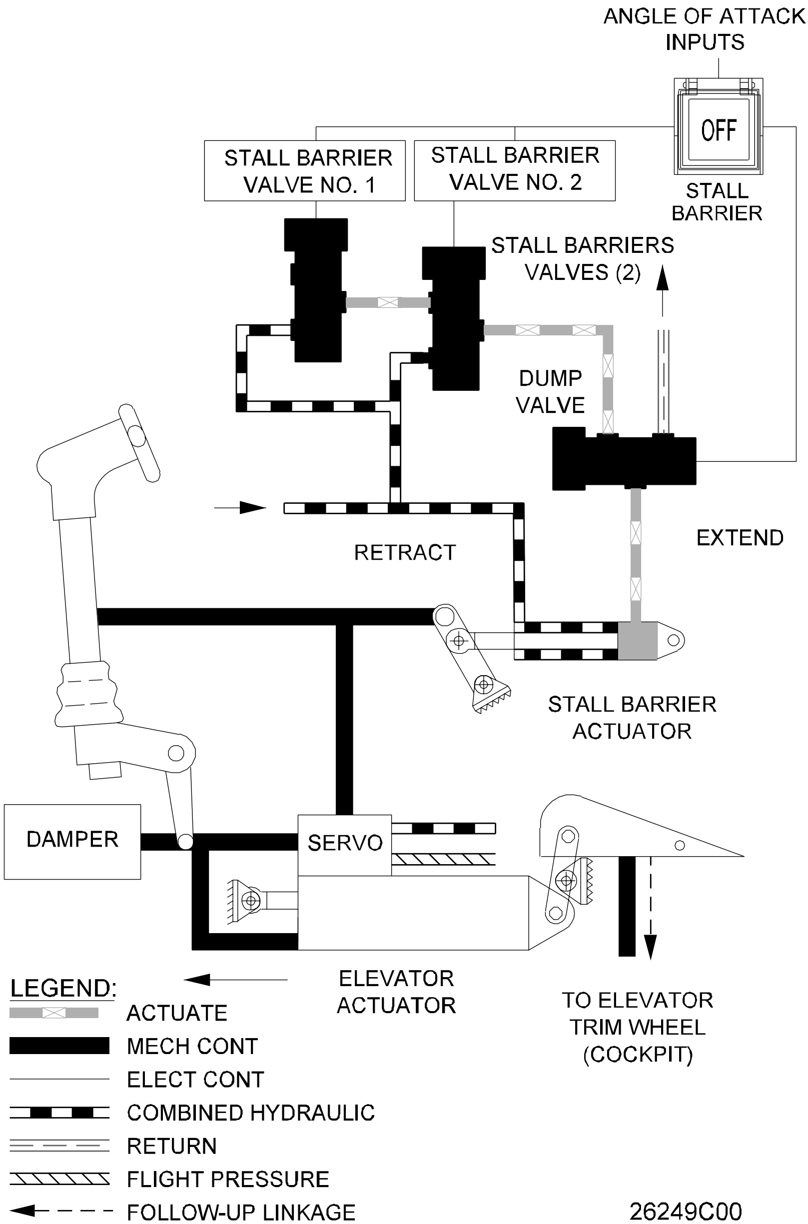
Control Columns
Figure 7



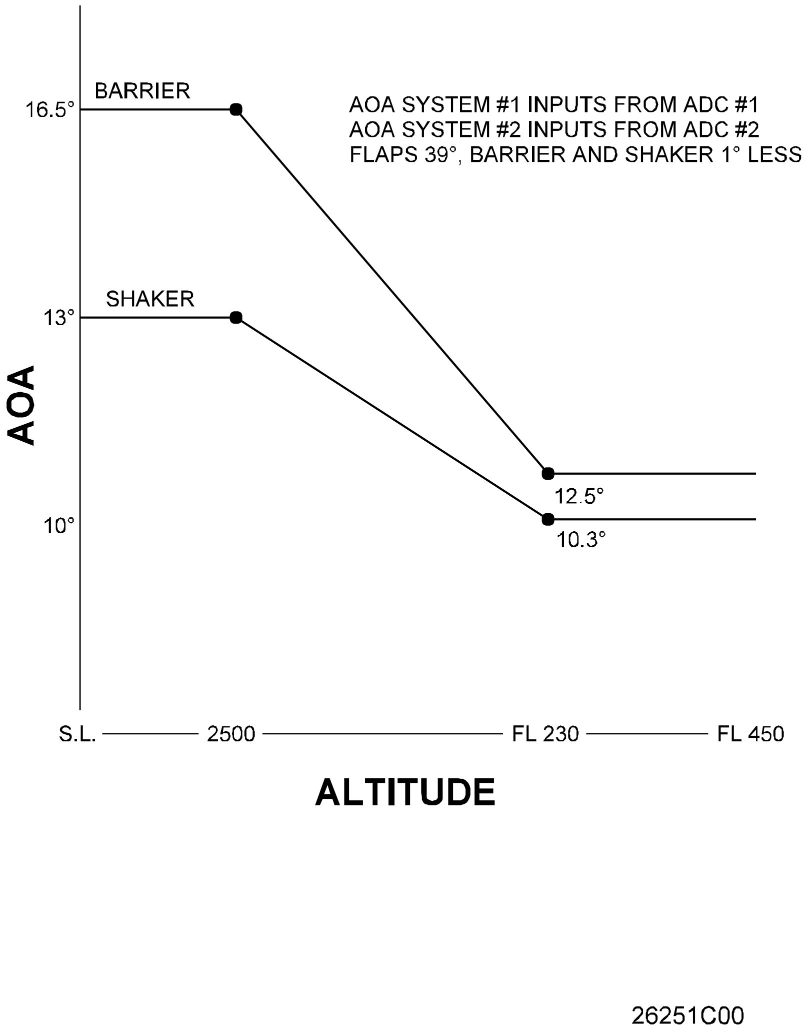
FLIGHT POWER SHUT OFF Handle
Figure 8



Yaw Damper / Pitch Trim Control Panel
Figure 9



Stall Barrier Simplified Block Diagram
Figure 10



Stall Barrier / Angle of Attack Warning Parameters
Figure 11

2A-27-30: Yaw Flight Control System

1. **General Description:**

Aircraft movement about the vertical axis (yaw) is controlled by the position of a single rudder. The rudder is both manually (rudder pedals) and electrically (yaw damper) controlled, mechanically actuated and hydraulically boosted. It is mounted on the trailing edge of the vertical stabilizer. Total rudder travel ranges from 22° left or right of neutral.

The rudder is positioned by an electrohydraulic servoactuator located in the tail compartment. The actuator receives hydraulic operating pressure simultaneously from the Combined and Flight hydraulic systems during normal operations. In the event of total loss of hydraulic pressure in both systems, the rudder reverts to manual operation. Manual reversion is also possible through use of a flight power shutoff valve and its pedestal-mounted control handle.

A rudder trim system is used to position the entire rudder surface (no trim tab is incorporated). Manual trim is accomplished by a pedestal-mounted control knob.

A yaw damping system is incorporated to provide stability augmentation about the vertical axis by counteracting any dutch roll tendency. Electrical inputs made by the yaw damper to the rudder actuator are transparent to the flight crew and can be made simultaneously with other inputs from the flight crew.

On CAA certified aircraft, a flight control automatic failure detection system compares rudder pedal inputs to rudder actuator outputs. If a malfunction is detected, the system shuts off either or both hydraulic power sources to the actuator.

The yaw flight control system consists of the following subsystems, units and components:

- Rudder pedal system
- Mechanical actuation system
- Hydraulic boost system
- Manual reversion system
- Rudder load limiting system
- Yaw damper system
- Yaw trim system
- Failure detection system (CAA aircraft only)

2. **Description of Subsystems, Units and Components:**

   A. **Rudder Pedal System:**

(See Figure 13.)

Each pair of rudder pedals functions both as a conventional rudder control and toe brake control. With nosewheel steering operating, limited steering control (7° left and right of neutral) is also available using the rudder pedals.

Each pilot can adjust the resting position of their rudder pedals to enhance comfort and usability. Lifting an adjusting lever located between each pair of rudder pedals releases a lock to permit fore and aft adjustment. The lever handle is then pulled aft or pushed forward, moving the rudder pedal set in the same direction. Once in the desired position, releasing the lever locks the pedals in place.

Each individual rudder pedal attaches to a pedal hanger arm that pivots fore and aft on a lateral support tube. A rod assembly connected between each pedal hanger arm and a vertical torque tube translates the fore and aft movement of the pedal into rotational movement of the vertical torque tube. An interconnect pushrod between the bases of the left and right torque tubes interconnects the pilot's and copilot's rudder pedal sets. An output crank attached to base of the right (copilot's) torque tube ties the rudder pedals to the rudder mechanical actuation system. A rotating lug-type stop attached to base of the left (pilot's) torque tube limits rudder pedal travel when it contacts either one of two adjustable stop bolts.

Artificial feel forces are supplied by a spring bungee attached to the rudder actuator input cable sector. The bungee provides a resistant force proportional to rudder pedal displacement.

B. Mechanical Actuation System:

(See Figure 12.)

Rudder pedal inputs are transmitted rearward through a system of push-pull rods, bellcranks, cables and a sector assembly to an input cable sector near the base of the rudder in the tail compartment. As the right torque tube rotates, its output crank moves an attached pushrod forward and aft. The pushrod transmits this movement through a motion reversing crank and another pushrod to a cable sector. Cables from the cable sector transmit control inputs rearward to an input cable sector. Movement of the input cable sector then drives the rudder actuator servo control valve through an input crank arm.

C. Hydraulic Boost System:

(See Figure 12.)

The rudder actuator is a dual tandem actuator consisting of two pistons secured to a common shaft. The pistons move inside a common cylinder divided to create two separate cylinders. One cylinder receives Flight hydraulic system pressure while the other cylinder receives Combined hydraulic system pressure.

Mechanical movement of the actuator input crank arm moves the servo control valve from its neutral position. The servo control valve then directs hydraulic pressure to one of the actuator's two cylinders and connects each cylinder's opposite side to return. Movement of the actuator then drives the rudder through an output crank, pushrod, and horn and tube combination. When the rudder reaches the desired deflection, the servo control valve shifts to its neutral position to lock hydraulic pressure within the actuator, in effect preventing further surface movement.

The input cable sector and output crank are connected through a center rear crank through a pin and slot arrangement. This pin and slot arrangement creates 4° of differential motion between the input cable sector and the output crank, allowing maximum servo control valve displacement while providing a total of 3° of yaw damper authority.

D. Manual Reversion System:

During normal flight operations, the Combined and Flight hydraulic systems each supply and maintain pressure to the rudder actuator. Loss of system pressure from both hydraulic systems will automatically revert the yaw flight control system to manual control. As pressure at the actuator

drops below 75 psi, bypass valves within the actuator open to allow the actuator piston to idle. With the actuator piston idling, the system is said to be in manual reversion, technically explained in the following paragraph.

The input cable sector and output crank rotate on a mutually independent common pivot. They are linked by a pin and elongated slot arrangement. The input crank is on the pin and the output crank is on the slot. With the actuator piston idling, the pin moves within the slot until it reaches either end. At this point, mechanical contact is made and the input cable sector now drives the output crank. The output crank, in turn, drives the rudder through its push-pull rods and cranks.

Manual reversion of the yaw flight control system is also possible by closing a normally open flight power shutoff valve. The flight power shutoff valve is a mechanically operated shutoff valve located between the Combined and Flight hydraulic system pressure sources and the rudder actuator (as well as the aileron, elevator and flight / ground spoiler actuator) pressure lines. The valve consists of two mechanically connected but hydraulically isolated sections. A controlex cable connects the valve to a FLIGHT POWER SHUT OFF handle located on the left aft side of the cockpit center pedestal. See Figure 8.

Moving the FLIGHT POWER SHUT OFF handle up from its stowed (horizontal) position to the vertical position mechanically closes the flight power shutoff valve. With the valve closed, operating pressure is removed from the actuator, allowing the piston to idle.

The resultant advantage of the flight power shutoff provision is the ability to bypass a malfunctioning actuator (such as would be the need in the unlikely event of an actuator jam) and manually fly the aircraft. Although rudder pedal effort and response time to inputs are increased while in manual reversion, the aircraft remains capable of positive and harmonious control.

E. Rudder Load Limiting System:

(See Figure 12.)

Load-limiting (force-modulating) valves inside the rudder actuator restrict Combined and Flight hydraulic system pressure to prevent vertical stabilizer structural overload. Surface movement is limited by these valves when higher airspeeds increase the airloads imposed against the rudder. When the hinge moment limit is reached, the force-modulating shift, reducing pressure to a maximum of approximately 2,250 psi, regardless of hydraulic system pressure. Operation of either valve to restrict hydraulic pressure closes an internal microswitch that triggers a blue RUDDER LIMIT advisory CAS message. With the RUDDER LIMIT message displayed, further displacement of the rudder surface is inhibited.

When on the ground with the Combined hydraulic system operating, rudder actuator load limiting can be checked by applying gradual input force to the rudder pedals until the rudder surface is bottomed against its stops (left or right). The RUDDER LIMIT advisory CAS message will be displayed at the instant of pressure limiting within the actuator. Flight crews are cautioned that both the gust lock and nose wheel steering must be OFF before performing this test. Also, it should be noted that if performing this test with only the Flight hydraulic system operating, the RUDDER LIMIT message may or may not be displayed.

In the event of a malfunction or failure of internal rudder actuator load limiting, redundant limiting is provided by an additional external load limiting valve installed upstream of the rudder actuator. With this configuration, continuous rudder load limiting is afforded as either load limiting valve (internal or external) will limit output pressure. The external load limiting valve limits total hydraulic pressure to a maximum of 2,650 (Combined and Flight systems) to the rudder actuator.

The external load limiting valve is checked by observing CAS for display of an amber SNGL RUDDER LIMIT caution message. Proper operation of the valve is indicated by display of the message during the following sequence:

- Before starting engines (power on) — message displayed
- Right engine only operating — message displayed
- Both engines operating — message not displayed

Proper operation of the valve can also be verified during the following sequence:

- 0 psi Combined pressure / 3,000 psi Flight pressure — message displayed
- 3,000 psi Combined pressure / 3,000 psi Flight pressure — message not displayed
- 3,000 psi Combined pressure / 0 psi Flight pressure — message not displayed

F. Yaw Damper System:

A series mode yaw damper is incorporated into the rudder actuator. Controlled by the autopilot, the yaw damper provides stability augmentation by automatically counteracting any dutch roll tendency within the aircraft. The system is referred to as a series mode system in that no feedback is provided through the rudder pedals.

The yaw damper system consists of a solenoid-operated shutoff valve, transfer valve, servo ram and summing lever / transducer. Electrical power is supplied to the system through the Right Main and Essential 28 VDC buses. Hydraulic power is supplied to the system by the Flight hydraulic system only.

Located on the pilot's flight panel, the YAW DAMP ENG / DISENG switch engages or disengages the yaw damper. With the yaw damper engaged (amber DISEN switch legend extinguished), 28 VDC power energizes the solenoid-operated shutoff valve to the open position. Flight hydraulic system pressure then flows to the transfer valve. Hydraulic pressure flow and volume through the transfer valve is controlled by the torque motor using a jet pipe / receiver pipe arrangement. The yaw damper / pitch trim control panel is shown in Figure 9.

Depending on commands provided by the autopilot, the torque motor positions a jet pipe to direct hydraulic pressure to two receiver pipes inside the transfer valve. If equal commands are sent by the autopilot, the jet pipe remains in a centered, or null, position. When unequal commands are sent, the torque motor deflects the jet pipe in the necessary direction to supply more pressure to one receiver pipe than the other. This shifts the transfer valve spool and hydraulic pressure flows from the transfer valve to the servo ram.

Under hydraulic pressure, the servo ram then moves in the desired direction to reposition the rudder actuator servo control valve which, in turn, results in rudder movement.

As the system moves the rudder, a summing lever / transducer assembly provides position information to the yaw servo amplifier. The servo amplifier, in turn, nulls the signal to the transfer valve.

G. Yaw Trim System:

Trim control about the vertical axis is controlled by displacing the entire rudder surface (there is no rudder trim tab). This is accomplished through the use of a rudder trim control wheel mounted on the upper aft end of the center pedestal, shown in Figure 14.

Rotation of the rudder trim control wheel left or right rotates a cable drum below the cockpit floor through the use of a torque tube. As the drum rotates, cables transmit the movement to a screw-type rudder trim actuator. The actuator, in turn, extends and retracts, repositioning the rudder actuator input cable sector through the trim actuator attachment to the artificial feel bungee. Movement of the input cable sector provides input to the rudder actuator servo control valve, thus driving the rudder to effect trim position.

Since the are no stops incorporated in the rudder trim actuator, total travel is determined by integral stops in the rudder trim control wheel. Total trim wheel travel is approximately 6½ turns from stop to stop. This yields a maximum of 10 units (7.5°) NOSE L (left) and 10 units (7.5°) NOSE R (right) of rudder deflection from the neutral position.

H. Failure Detection System:

CAA Certified Aircraft Only: A flight control automatic failure detection system monitors flight control inputs from the rudder pedals and compares them to the rudder actuator outputs. If the system detects a failure, it automatically shuts off hydraulic pressure to the actuator and triggers the appropriate warning on the Crew Alerting System (CAS). Once activated by a malfunction, hydraulic pressure is inhibited until power to the respective monitoring system is interrupted, for instance, by pulling and resetting the appropriate circuit breaker.

The monitoring system is a dual-channel system. One channel controls the Combined hydraulic system pressure source while the other controls the Flight hydraulic system pressure source. Power for the system is received from the 28 VDC Essential DC bus.

A pair of limit switches monitor applied rudder pedal input while a pair of reed switches monitor actuator output in response to the input.

If a disagreement occurs between rudder pedal input and actuator output, the associated limit switch and reed switch close to complete a circuit to the respective hydraulic shutoff delay relay. If the relay remains energized for more than ½ second, it energizes the respective hydraulic shutoff control relay. The control relay, in turn, powers its hydraulic shutoff valve to the closed position. Activation of the shutoff valve also causes an amber RD CMB HYD OFF (or RD FLT HYD OFF) message to be displayed on CAS.

3. Controls and Indications:

(See Figure 9 and Figure 14.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
STAB AUG SERVO #1	CPO	A-6	Essential DC Bus
STAB AUG SERVO #2	CPO	B-6	R Main DC Bus
RUD COMB HYD S/O (1)	CPO	C-14	Essential DC Bus
RUD FLT HYD S/O (1)	CPO	A-16	Essential DC Bus

NOTE(S):

(1) CAA certified aircraft only.

B. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
RD CMB HYD OFF (1)	The flight control automatic failure detection system has shut off Combined hydraulic system pressure to the rudder actuator.
RD FLT HYD OFF (1)	The flight control automatic failure detection system has shut off Flight hydraulic system pressure to the rudder actuator.
SNGL RUDDER LIMIT	Failure of input pressure load limiter (with Flight and Combined Hydraulic System pressure normal.)
YAW DAMPER OFF	Yaw damper (YAW DAMP ENG / DISENG) switch is OFF. **OR:** Yaw damper failure.

NOTE(S):

(1) CAA certified aircraft only.

C. Advisory (Blue) Messages and Annunciations:

CAS Message:	Cause or Meaning:
RUDDER LIMIT	Rudder actuator torque limiter is in operation.

4. Limitations:

A. Yaw Damper Limitations:

(1) If yaw damper fails prior to takeoff:

Maximum fuel quantity permitted for takeoff is 9000 lb (4082 kg).

(2) If yaw damper fails in flight:

(a) Above 18,000 feet:

Maintain airspeed at or above 220 KCAS.

(b) Below 18,000 feet:

Maintain airspeed as a function of fuel quantity at or above that shown on the following table until ready to configure for approach and landing.

Fuel Qty - 1000 Lb	8	10	12	14	16	18	20	22	24	26	28	30
Minimum Airspeed - KCAS	96	107	117	126	135	143	151	158	165	172	178	184

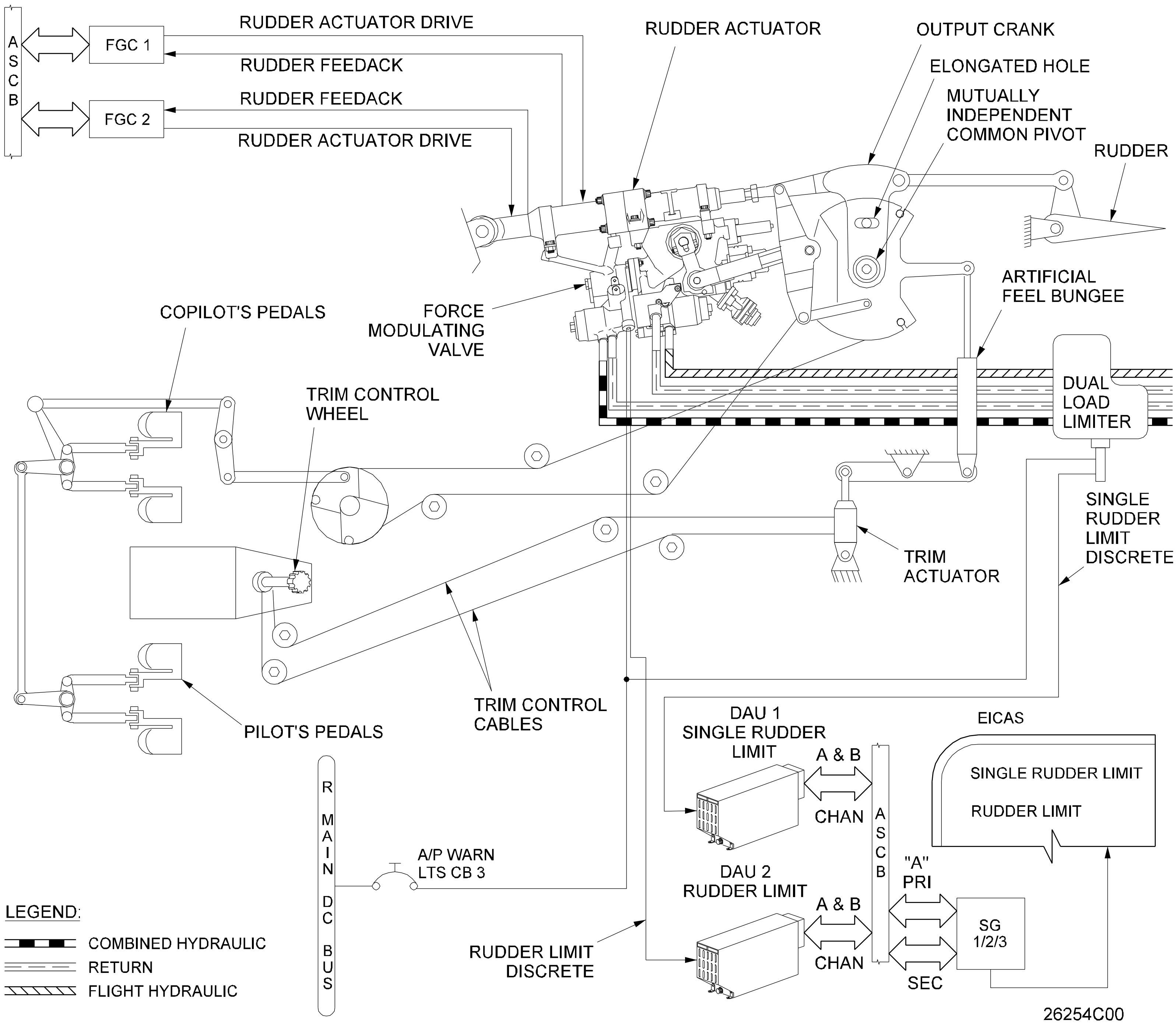
B. Mach Trim Compensation / Electric Elevator Trim:

(1) Use of mach trim compensation:

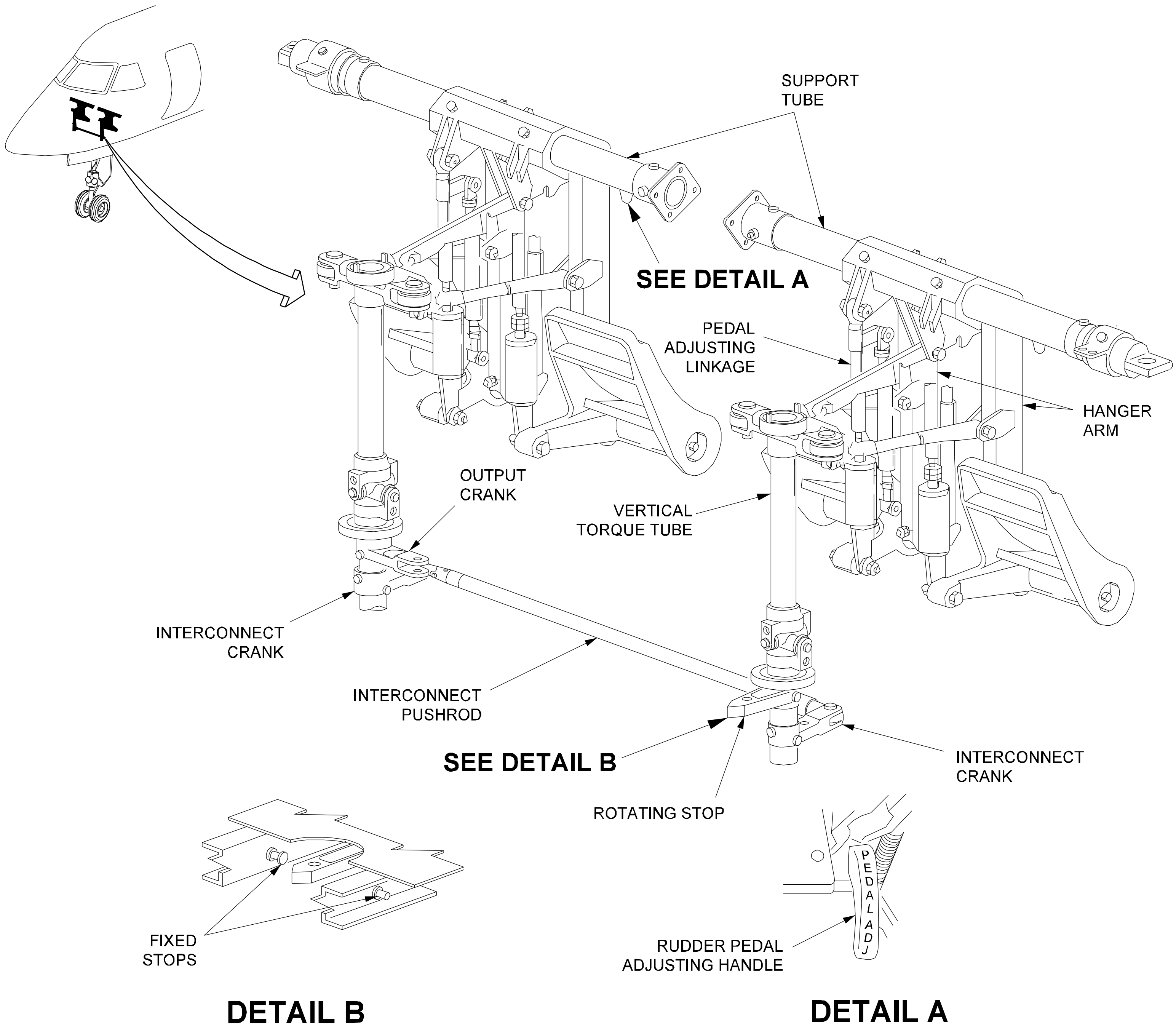
Mach trim compensation must be ON during all flight operations except as provided for in Section 05-03-40, Mach Trim Compensation Failure.

(2) If mach trim compensation failure is coupled with yaw damper failure:

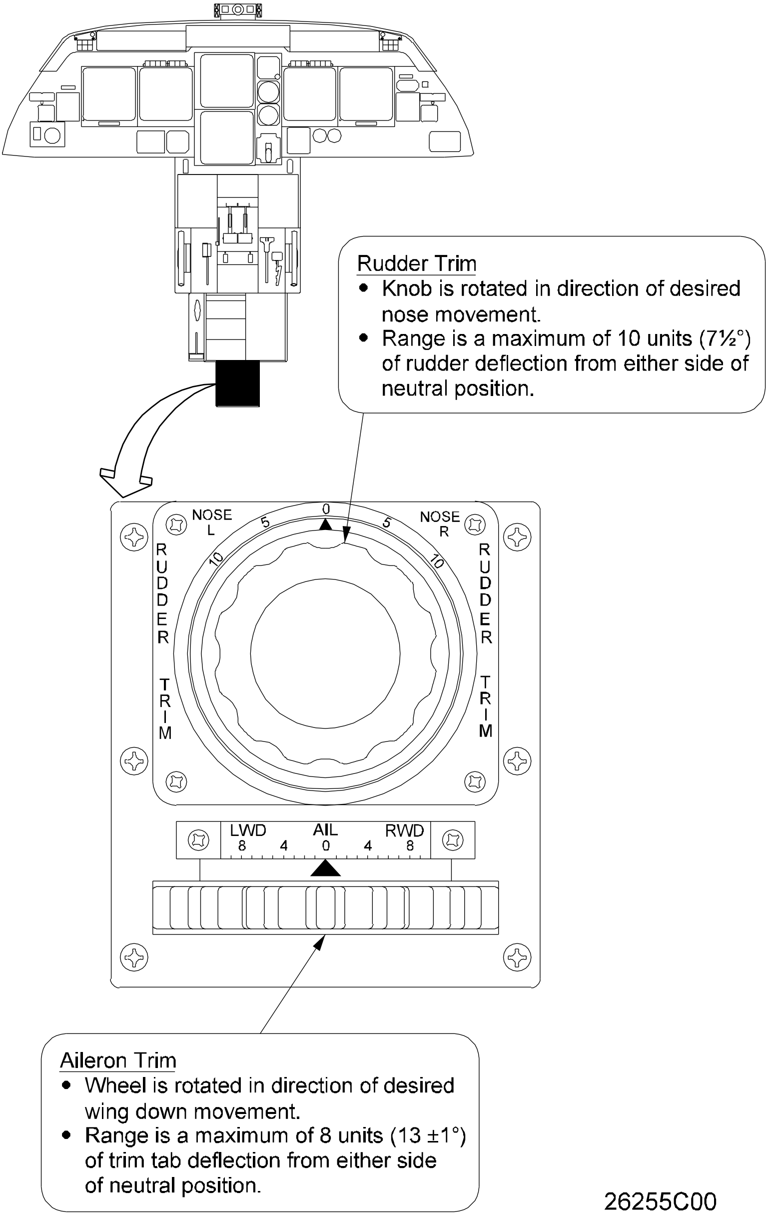
When mach trim compensation failure is coupled with yaw damper failure, observe speed limitations for both failures and limit altitude to 41,000 ft.



Yaw Flight Control System
Simplified Block Diagram
Figure 12



Rudder Pedals / Forward Linkage
Figure 13



Yaw Trim / Roll Trim Control Wheels
Figure 14

2A-27-40: Roll Flight Control System

1. General Description:

Aircraft movement about the longitudinal axis (roll) is controlled by the position of the ailerons. The ailerons are manually (control wheel) or electrically (autopilot servo) controlled, mechanically actuated and hydraulically boosted airfoils mounted on the trailing edge of each wing. Total aileron travel ranges from 10° trailing edge up to 10° trailing edge down.

Each aileron is positioned by a tandem type hydraulic actuator. The actuator receives hydraulic operating pressure simultaneously from both the Combined and Flight hydraulic systems during normal operations. Loss of a single hydraulic system has no effect on operation of the ailerons, as the remaining system is capable of maintaining actuator load capacity. In the event of total loss of hydraulic pressure in both systems, the ailerons revert to manual operation. Manual reversion is also possible through use of a flight power shutoff valve and its pedestal-mounted control handle.

Flight spoilers are incorporated into the roll flight control system to improve aircraft roll response. During roll, the spoilers respond to upward movement of the ailerons while the opposite spoilers remain flush to the wing as the opposite aileron travels downward. Spoiler travel varies in proportion to the degree of the roll, to maximum extension of 26 ±2° (55 +4/-3° with speed brakes extended). The flight spoiler system is solely a hydraulically powered system, thus reversion to manual control is not possible.

A roll trim system is used to position a trim tab attached to the trailing edge of the left aileron. The tab is positioned manually by a pedestal-mounted control wheel.

On CAA certified aircraft, a flight control automatic failure detection system monitors forces within the roll flight control system. If a malfunction is detected, the system shuts off both hydraulic power sources to the affected actuator.

The roll flight control system consists of the following subsystems, units and components:

- Control wheel system
- Mechanical actuation system
- Hydraulic boost system
- Manual reversion system
- Flight spoiler system
- Roll trim system
- Failure detection system (CAA aircraft only)

2. Description of Subsystems, Units and Components:

A. Control Wheel System:

(See Figure 7.)

A control wheel mounted on top of the left and right control column is used for roll control. Each control wheel has the following controls incorporated:

- Pitch trim (NOSE UP / NOSE DOWN)
- Map light (MAP)
- Transponder identification (IDENT)
- Touch control steering (TCS)

- Autopilot / stall barrier disconnect (A/P DISC / BARR DISC)
- Radio / intercom push-to-talk (RADIO / ICS)

A torque shaft in the top of the control column transmits left and right control wheel movement through a universal joint to a vertical torque tube that descends through the column. The torque tube, in turn, connects to a two-armed crank at the base of the column. On the rear facing arm of the crank, a pushrod assembly interconnects the pilot's and copilot's control wheels. The aileron control cables are connected to the outboard facing arm of the crank. Non-adjustable stops in the control column limit control wheel movement to 90° left or right of neutral.

B. Mechanical Actuation System:

(See Figure 15.)

Control wheel inputs are transmitted rearward through a synchronized system of pushrods, bellcranks, cables and sector assemblies. The system transmits the inputs to the outboard sector assembly. The outboard sector assembly operates the pushrods and bellcranks that form the input and output cranks. Cradled between these cranks is the aileron actuator. The leverage ratio of the input and output cranks provides a 5:1 boost ratio, i.e., one unit of work supplied by the flight crew through the control wheel results in five units of work provided by the actuator. Movement of the input and output cranks about a common pivot point displaces the aileron actuator servo control valve input lever.

C. Hydraulic Boost System:

The aileron actuator is a dual tandem actuator consisting of two pistons secured to a common shaft. The pistons move inside a common cylinder divided to create two separate cylinders. One cylinder receives Flight hydraulic system pressure while the other cylinder receives Combined hydraulic system pressure.

Mechanical movement of the actuator input lever moves the servo control valve from its neutral position. The servo control valve then directs hydraulic pressure to one of the actuator's two cylinders and connects each cylinder's opposite side to return. When the aileron reaches the desired deflection, the servo control valve shifts to its neutral position to lock hydraulic pressure within the actuator, in effect preventing further surface movement.

D. Manual Reversion System:

During normal flight operations, the Combined and Flight hydraulic systems each supply and maintain 3000 psi to the aileron actuators. Loss of system pressure due to a single system failure has no effect on operation of the roll flight control system.

Loss of system pressure from both hydraulic systems will automatically revert the roll flight control system to manual control. As pressure at each actuator drops below 60 psi, bypass valves within the actuator open to allow the actuator piston to idle. With the actuator piston idling, the system is said to be in manual reversion. The ailerons are now controlled solely by mechanical means; the flight spoilers are inoperative.

Manual reversion of the roll flight control system is also possible by closing a normally open flight power shutoff valve. The flight power shutoff valve is

a mechanically operated shutoff valve located between the Combined and Flight hydraulic system pressure sources and the aileron actuators (as well as the elevator, rudder and flight / ground spoiler actuator) pressure lines. The valve consists of two mechanically connected but hydraulically isolated sections. A controlex cable connects the valve to a FLIGHT POWER SHUT OFF handle located on the left aft side of the cockpit center pedestal. See Figure 8.

Moving the FLIGHT POWER SHUT OFF handle up from its stowed (horizontal) position to the vertical position mechanically closes the flight power shutoff valve. With the valve closed, operating pressure is removed from the actuator, allowing the piston to idle.

The resultant advantage of the flight power shutoff provision is the ability to bypass a malfunctioning actuator (such as would be the need in the unlikely event of an actuator jam) and manually fly the aircraft. Although control wheel effort and response time to inputs are increased while in manual reversion, the aircraft remains capable of positive and harmonious control.

E. Flight Spoiler System:

(See Figure 15.)

A mixing linkage between the aileron and flight spoiler control mechanisms mixes control wheel inputs to provide aileron / flight spoiler coordination. This allows the two outboard spoiler panels on the same side as an upward moving aileron to extend up to a maximum of 26 ±2° for roll control. If the speed brakes are extended, the flight spoilers extend up to a maximum of 55 +4/-3°.

As the aileron control mechanism commands an aileron to deflect upward, the mixing linkage transmits an extend command to the flight spoiler actuator control valve through push-pull rods, bellcranks and cables. The control valve then shifts to direct pressure to the flight spoiler actuator. The normally-extended actuator retracts and the flight spoiler panels follow aileron movement to assist roll control. The flight spoiler panels on the opposite wing remain retracted.

F. Roll Trim System:

(See Figure 14 and Figure 15.)

Trim control about the longitudinal axis is controlled by displacing a trim tab attached to the trailing edge of the left aileron. This is accomplished through the use of an aileron trim control wheel mounted on the upper aft end of the center pedestal.

Rotation of the aileron trim control wheel left or right rotates a cable drum through the use of a torque tube. As the drum rotates, cables transmit the movement to the aileron trim actuator. Mechanical linkage attached to the actuator causes the trailing edge of the aileron trim tab to be raised or lowered.

Movement of the aileron trim control wheel also drives an incorporated pointer that indicates the units of Left Wing Down (LWD) or Right Wing Down (RWD) trim from neutral. Total trim wheel travel is approximately 4½ turns from stop to stop. This yields a maximum of 8 units (13 ±1°) of trim tab deflection from either side of the neutral position.

If an attempt is made to force the aileron trim control wheel past its mechanical stops, a shear pin breaks to disconnect the control wheel from the torque tube, disabling roll trim.

G. Failure Detection System:

CAA Certified Aircraft Only: A flight control automatic failure detection system monitors force applied to the aileron input / manual reversion rod, providing detection of a jammed or malfunctioning aileron actuator. If the system detects a failure, it automatically shuts off all hydraulic pressure to that actuator (the opposite actuator remains unaffected) and triggers the appropriate warning on the Crew Alerting System (CAS).

With the roll flight control system operating normally, forces on the aileron input / manual reversion rod remain relatively low because there is no lag between control wheel input and actuator output. If an actuator malfunctions resulting in an aileron hardover action, force levels on the rod increase significantly. When this force exceeds 150 pounds, a force-link limit switch closes to energize an associated time delay relay.

Similarly, a jammed actuator results in increased control inputs being applied by the flight crew. If these control inputs exceed approximately 20 pounds, the force-link limit switch closes to energize an associated time delay relay.

If the time delay relay remains energized for more than ½ second, it energizes a shutoff relay. The shutoff relay, in turn, powers the actuator's Combined and Flight hydraulic system shutoff valves to the closed position. Closing of the shutoff valves also causes an amber L-R AIL HYD OFF message to be displayed on CAS.

3. Controls and Indications:

(See Figure 14.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
L AIL HYD S/O (1)	CPO	A-14	Essential DC Bus
R AIL HYD S/O (1)	CPO	B-14	Essential DC Bus

NOTE(S):

(1) CAA certified aircraft only.

B. Caution (Amber) Messages and Annunciations:

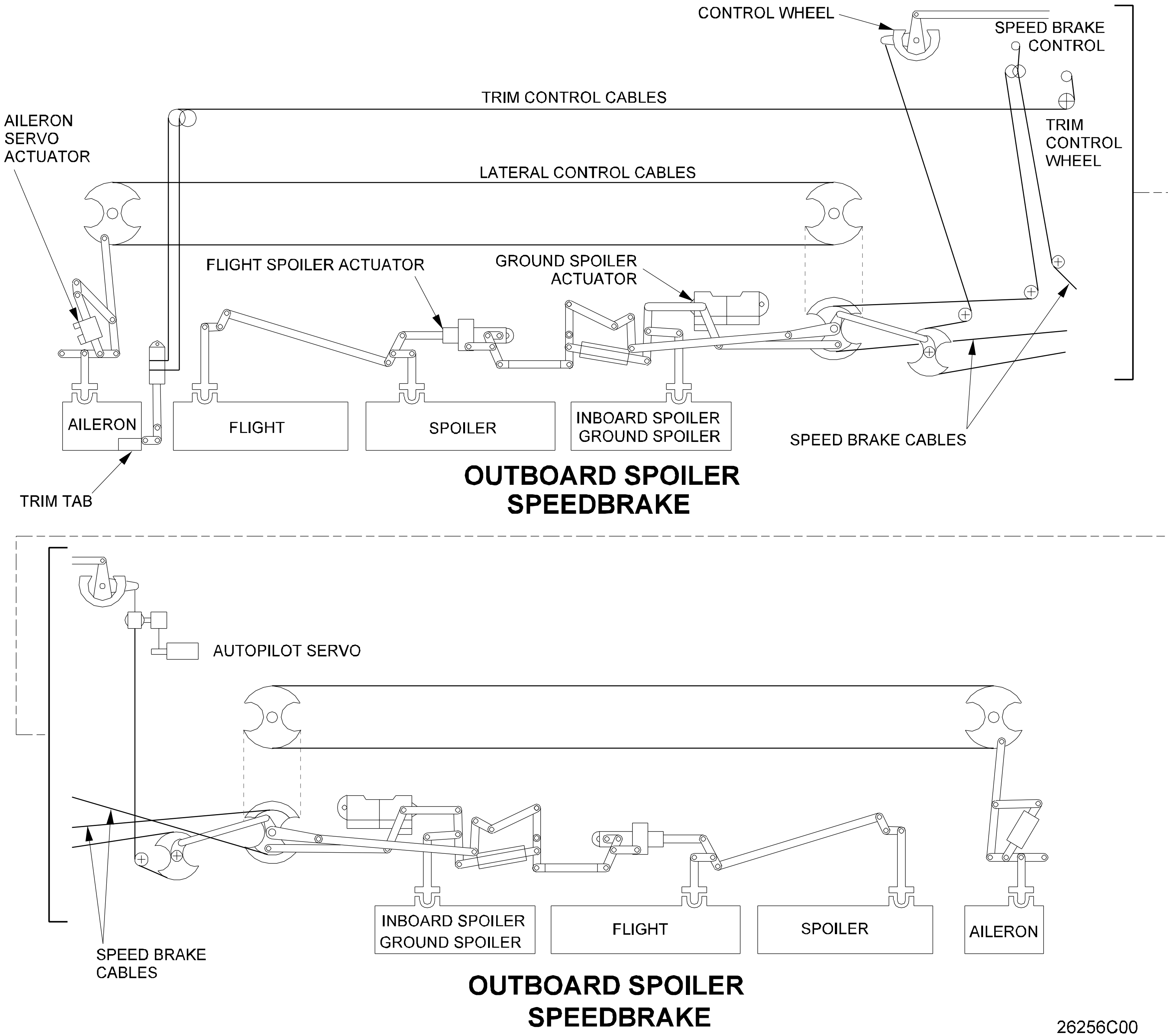
CAS Message:	Cause or Meaning:
L AIL HYD OFF (1)	The flight control automatic failure detection system has shut off Combined and Flight hydraulic system pressure to the left aileron actuator.
R AIL HYD OFF (1)	The flight control automatic failure detection system has shut off Combined and Flight hydraulic system pressure to the right aileron actuator.
RETRIM L-R WING DOWN	Lateral axis out of trim.

NOTE(S):

(1) CAA certified aircraft only.

4. Limitations:

There are no limitations established for the roll flight control system at the time of this revision.



Roll Flight Control System
Simplified Block Diagram
Figure 15

2A-27-50: Horizontal Stabilizer System

1. General Description:

A movable horizontal stabilizer provides automatic longitudinal trim assistance to compensate for the nose pitchdown moment (tuck-in) associated with flap extension. Stabilizer position (leading edge angle of incidence) is a direct function of flap position, as the stabilizer is mechanically driven from the flap central gearbox. If a malfunction occurs rendering the stabilizer immovable, a safe landing can be made using normal pitch trim.

Horizontal stabilizer position is displayed on the FLAP / STAB position indicator located on the copilot's skirt panel.

The horizontal stabilizer system is composed of the following subsystems, units and components:

- Stabilizer drive system
- Torque limiter
- Dwell box
- Stabilizer actuator
- Stabilizer position and warning system

2. Description of Subsystems, Units and Components:

(See Figure 16 and Figure 17.)

A. Stabilizer Drive System:

The stabilizer drive system consists of torque shafts installed beginning from the aft side of the flap central gearbox, rearward through the tail compartment via a dwell box, then upward through the vertical stabilizer to the stabilizer actuator. Bearing housings and pillow blocks installed at defined intervals support the torque shafts, preventing whiplash as they rotate.

B. Torque Limiter:

A torque limiter is installed at the input side of the central gearbox. If the gearbox, dwell box or stabilizer actuator should jam or fail, the torque limiter spring jams to lock the torque shafts. This stalls the central gearbox hydraulic motor, stopping further flap and horizontal stabilizer movement.

If a torque shaft fails, the flaps are still operable throughout their full range. The stabilizer, however, remains at its last position prior to shaft failure. The flight crew would then use pitch trim as necessary to compensate for the immovable stabilizer.

C. Dwell Box:

A dwell box, located in the tail compartment, is incorporated in the stabilizer drivetrain to control stabilizer movement during flap extension and retraction. It contains an input side and an output side.

During flap extension, the shaft connected to the input side of the dwell box rotates but the output side does not rotate until the flaps reach a specified position. At the final stages of central gearbox rotation, the shaft connected to the output side of the dwell box rotates to drive the stabilizer actuator.

During flap retraction, the shaft connected to the output side of the dwell box immediately begins rotation to drive the stabilizer actuator. When the flaps reach a specified position, output shaft rotation ceases, although

input shaft rotation continues during flap retraction.

D. Stabilizer Actuator:

The irreversible stabilizer actuator consists of a double Acme™ screw with a reduction gear. One end of the actuator connects to the leading edge of the horizontal stabilizer and the other end connects to the vertical stabilizer structure. The reduction gear reduces the input torque shaft speed of 730 RPM to an actuator screw speed of 33 RPM to remain commensurate with the speed of flap movement. Should a malfunction, jam or failure occur, the actuator remains in its last commanded position.

E. Stabilizer Position and Warning System:

A transmitter at the aft upper end of the stabilizer surface supplies stabilizer position information to the FLAP / STAB position indicator on the copilot's skirt panel. Although power for the system is normally supplied by the Essential 28 VDC bus, position indication remains functional down to, and including, emergency battery operation. Horizontal stabilizer angle of incidence and FLAP / STAB indicator position relative to flap position is outlined in the following table:

FLAP POSITION	FLAP / STAB INDICATOR	STABILIZER ANGLE OF INCIDENCE
0° (UP)	UP	1° leading edge down
10°	T/O	2.2° leading edge down
20° (T/O APPR)	APP	3.2° leading edge down
39° (DOWN)	LDG	4.6° leading edge down

A limit switch below the stabilizer position transmitter provides a warning if the stabilizer fails to move to the proper position with flap retraction. If the stabilizer fails to reach the 1° leading edge down position with the flaps retracted, the limit switch causes an amber FLAP/STAB FAIL caution message to be displayed on the Crew Alerting System (CAS).

3. Controls and Indications:

(See Figure 16 and Figure 17.)

A. Circuit Breakers (CBs):

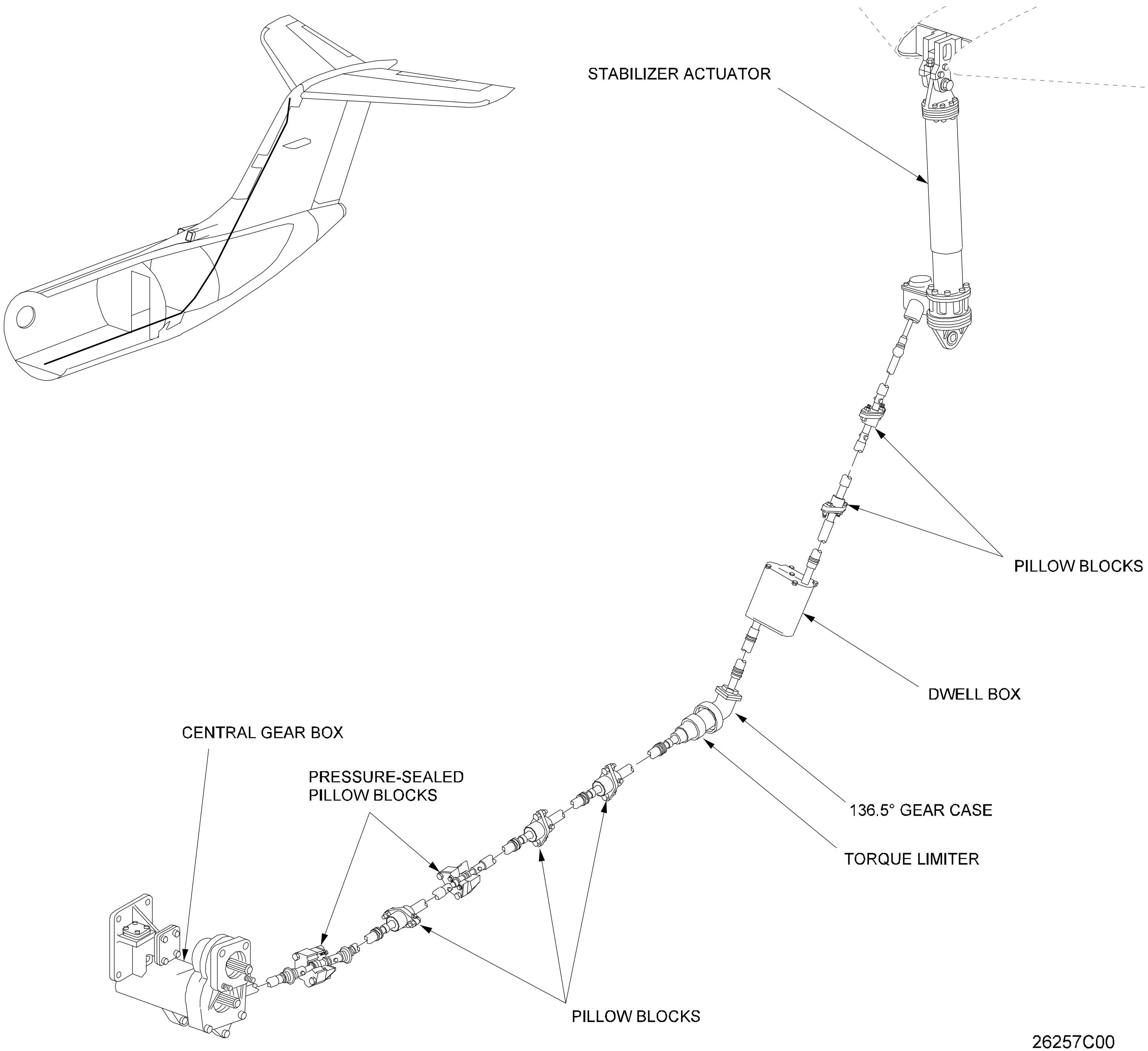
Circuit Breaker Name:	CB Panel:	Location:	Power Source:
FLAP/STAB POS	CP	E-5	Emergency DC Bus
FLAP/STAB WARN	P	D-3	Essential DC Bus

B. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
STAB-FLAP FAIL	Flaps UP (0°) and stabilizer not UP.

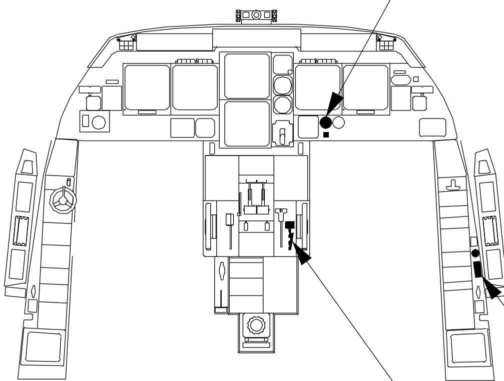
4. Limitations:

There are no limitations established for the horizontal stabilizer system at the time of this revision.



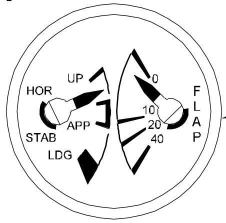
Horizontal Stabilizer
System Simplified Block
Diagram
Figure 16

SEE DETAIL A



SEE DETAIL B

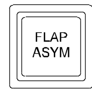
SEE DETAIL C



Flap/Stab Position Indicator

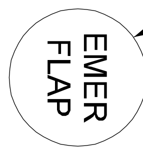
Pointers move corresponding to flap and stabilizer position as follows:

STAB	= Stabilizer Position	= FLAP	= Flap Position
UP	1.0° L.E. Down	0°	0° (UP)
T/O	2.2° L.E. Down	10°	10°
APP	3.2° L.E. Down	20°	20° (T/O APPR)
LDG	4.6° L.E. Down	39°	39° (DOWN)

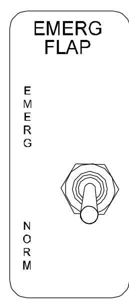


FLAP ASYM
(Aircraft not having ASC 345, but having ASC 69A.)
Illuminates amber to indicate flap asymmetry.

DETAIL A

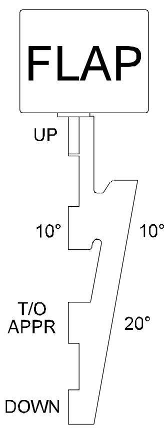


EMER FLAP
- To use handle, EMERG FLAP switch must be selected to EMERG or FLAP CONT circuit breaker must be open.
- Moving handle from NEUTRAL to FLAPS UP or FLAPS DOWN position directs hydraulic pressure through flap selector valve to hydraulic motor.
- When flaps reach desired position, moving EMER FLAP handle to NEUTRAL stops flaps.



EMERG FLAP
- EMERG:
  - • Deactivates normal FLAP handle and follow-up switches; allows direct control of flap selector valve.
  - • EMER FLAP handle is used to position flaps.
  - • Red X appears on both normalized AOA displays. Flap asymmetry protection is still provided.
- NORM:
  - • Normal operating position.
  - • Normal FLAP handle is used to position flaps.

DETAIL B



DETAIL C

26258C00

Flaps / Horizontal
Stabilizer System Controls
and Indications
Figure 17

2A-27-60: Flaps System

1. General Description:

The flaps system for the Gulfstream IV provides a means for the flight crew to control the position and movement of the trailing edge flaps, in order to allow steeper approach and climb angles, and lower takeoff and landing speeds.

The flaps are manually operated, electrically controlled, hydraulically powered, mechanically actuated airfoils located on the inboard trailing edge of the left and right wings. Designed as single-slotted Fowler-type flaps, they ride on rollers along four curved flap tracks attached to the rear beam of each wing.

Mechanical actuators (commonly referred to as screwjacks), driven by a hydraulic motor through a central gearbox, extend and retract the flaps. As the screwjacks extend the flaps, the flaps move aft and downward to increase wing area and camber.

An asymmetry protection system monitors flap position to prevent unequal movement. If one flap moves further than the other, the protection system stops flap movement by shutting off the hydraulic pressure source to the hydraulic motor.

A FLAP handle, located on the center pedestal, controls normal flap extension and retraction to four positions. Emergency flap control is provided by an EMER FLAP handle located on the copilot's right console. Flap position is displayed on the FLAP / STAB position indicator located on the copilot's skirt panel.

Extending the flaps normally results in a nose pitch down movement as the flaps increase wing lift. To compensate for nose pitch down, the horizontal stabilizer leading edge moves down corresponding to flap extension. Horizontal stabilizer operation is described in Section 2A-27-50, Horizontal Stabilizer System.

Units and components in the flaps system receive electrical power from the Essential DC bus. The primary source of hydraulic power for the system is the Combined hydraulic system, with the Utility hydraulic system available as a backup source. If both the Combined and Utility hydraulic systems are not available, the Auxiliary hydraulic system can be used to position the flaps using the AUX pump.

The flap system is composed of the following subsystems, units and components:

- Flap handles
- Flap shutoff valve
- Flap selector valve
- Hydraulic motor and central gearbox
- Flap actuators
- Flap follow-up switches
- Flap asymmetry protection system
- Flap position indication system
- Aircraft configuration warning system
- Flap control circuit breakers

A description of flap operation is also provided in this section.

2. Description of Subsystems, Units and Components:

(See Figure 17 through Figure 20.)

A. Flap Handles:

Normal positioning of the flaps is accomplished by a FLAP handle located on the right side of the cockpit center pedestal. It has four detents that set the flaps to each of its positions: UP (0°), 10°, Takeoff / Approach (T/O / APPR [20°]) and DOWN (39°). The handle moves vertically into each detent. At each detent, operating limit switches control electrical power to the flap circuitry.

An EMER FLAP handle on the copilot's side console mechanically controls the flap selector valve if the normal flap control system malfunctions. The handle moves vertically to select one of three detents: FLAPS UP, NEUTRAL (no movement) or FLAPS DOWN. Before using the EMER FLAP handle, however, the EMERG FLAP switch (just aft of the EMER FLAP handle) must be in the EMERG position in order to bypass the normal FLAP handle switches and flap follow-up switches.

If both the Combined and Utility hydraulic systems are not available, the Auxiliary hydraulic system can be used to position the flaps using the AUX pump. In this scenario, the normal FLAP handle would still be used.

During normal FLAP handle operation, both the FLAP CONT and MANUAL FLAP CONT circuit breakers must be set. During EMER FLAP handle operation, only the MANUAL FLAP CONT circuit breaker must be set.

B. Flap Shutoff Valve:

A normally closed (spring-loaded) shutoff valve controls hydraulic fluid flow to the flap selector valve. Electrically controlled by the flap control relay and the follow-up switches, the valve is energized open to allow fluid flow to the selector valve.

If an asymmetrical flap condition occurs, the flap asymmetry switches open the flap control relay to de-energize the flap shutoff valve and stop hydraulic pressure flow to the flap selector valve. Flap movement ceases at this point.

C. Flap Selector Valve:

The solenoid-operated flap selector valve directs hydraulic pressure flow to the hydraulic motor. When electrically actuated by the FLAP handle, one of the valve's two solenoids energizes to supply hydraulic pressure to the hydraulic motor to drive it in the required direction to extend or retract the flaps. When the flaps reach the commanded position, flap follow-up switches de-energize the flap selector valve through the control relay. Hydraulic pressure flow to motor is shut off and the motor stops.

Two flow regulators limit hydraulic fluid flow to the hydraulic motor to 2.6 Gallons Per Minute (GPM) and from the motor to return at 2.9 GPM.

D. Hydraulic Motor and Central Gearbox:

A fixed-displacement hydraulic motor powered by the Combined, Utility or Auxiliary hydraulic systems provides power to the flap system central gearbox. The gearbox employs 8.22:1 reduction gears to provide a 730 RPM output that drives the flap actuators.

The gearbox has three output shafts that drive the left and right flap actuators and the horizontal stabilizer actuator. An additional output shaft drives the flap follow-up switch assembly and flap position transmitter.

Multi-section torque shafts transmit rotational force from the gearbox to the

flap actuators and the horizontal stabilizer actuator. To compensate for wing flexing and allow free running, universal joints connect the torque shaft sections. Pillow blocks and bearings support the torque shafts to prevent whiplash.

E. Flap Actuators:

Each flap actuator consists of a gearcase with a rotating jackshaft and a ball nut connected to the flap structure. As the gearcase rotates the screwjack, the ball nut is driven forward and aft to extend or retract the flaps. Non-jamming mechanical stops are installed on each end of the screwjack. Because the inboard and outboard actuator screwjacks are different lengths, they rotate at different speeds to provide equal flap movement.

An internal torque limiter in each screwjack prevents damage to the flap by locking the input shaft if the flaps are extended at excessive airspeeds. By locking the input shaft, the hydraulic motor is stalled and flap movement stops. (This should not be confused with an asymmetrical flap condition, in which flap movement is restrained.) Once airspeed is reduced to the proper extension speed, flap movement can be resumed by moving the FLAP handle to the next upward or downward position, then back to the desired position.

If a torque shaft fails with the flaps extended, a “no-back” device in the gearcase prevents the flaps from creeping up (retracting) under airloads. The device consists of a clutch-type brake and a friction plate that senses compression loads on the screwjack. During normal screwjack movement, the brake remains released to allow flap retraction.

F. Flap Follow-Up Switches:

Twelve follow-up switches, actuated by a cam on the central gearbox, open and close as the flaps extend and retract. Once the flaps reach a selected position, the cam opens one of the switches to control the flaps and provide signals to other systems that require flap position information.

Six of the twelve switches (S1 through S6) control flap position in conjunction with the four flap handle switches. Once the flaps reach a selected position, the cam opens the respective follow-up switch, breaking the circuit to the flap control relay. The relay de-energizes to close the flap shutoff valve. Hydraulic pressure stops and the flaps stop moving.

G. Flap Asymmetry Protection System:

A flap asymmetry protection system monitors left and right flap movement to prevent an asymmetrical flap condition. Each outboard actuator has an asymmetry switch assembly connected to the opposite flap asymmetry switch to form an electrical circuit. The circuit serves to detect one flap moving faster than the other or movement of only one flap. With the flaps operating normally, the switches are in phase. If flap movement separates ¼ inch or more in any flap position, the circuit to the flap control relay is broken. The relay de-energizes to close the flap shutoff valve. Hydraulic pressure stops and the flaps stop moving. In order to maintain lateral stability, once flap operation is stopped due to asymmetry, further flap movement in either direction is not possible.

For SPZ-8400 equipped aircraft having ASC 69A (Flap Asymmetry Indicator Installation) incorporated: When the flap control relay

de-energizes to close the flap shutoff valve, a signal is sent causing an amber FLAP ASYMMETRY caution message is displayed on the Crew Alerting System (CAS).

For SPZ-8000 equipped aircraft having ASC 69A incorporated: When the flap control relay de-energizes to close the flap shutoff valve, a signal is sent causing an amber FLAP ASYM indicator (below the FLAP / STAB position indicator) to illuminate.

H. Flap Position Indication System:

A flap position transmitter is installed on the central gearbox. As the flaps extend and retract, an output shaft on the gearbox rotates the transmitter shaft, sending an electrical signal to the FLAP pointer on the FLAP / STAB position indicator.

I. Aircraft Configuration Warning System:

(See Figure 21.)

An aircraft configuration warning system monitors speed brake, power lever and flap position to provide visual and aural warnings of an unsafe aircraft configuration both on the ground and in flight.

The system receives inputs from the landing gear handle, speed brake handle, power lever takeoff power, flap follow-up switches and nutcracker switches. The Fault Warning Computers (FWCs) receive these inputs and generate the appropriate warnings if an unsafe configuration is detected.

A red ACFT CONFIGURATION warning message will be displayed on CAS, along with the associated warning tone, whenever the following conditions exist:

- Extending the flaps to DOWN (39°) in flight with the SPEED BRAKE handle not in the RETRACT detent
- Extending the landing gear in flight with the SPEED BRAKE handle not in the RETRACT detent
- Advancing either power lever above 80% HP RPM on the ground with the SPEED BRAKE handle not in the RETRACT detent
- Advancing either power lever above 80% HP RPM on the ground with the FLAP handle set to UP (0°) or DOWN (39°), i.e., flaps not in the takeoff range of 10° or T/O APPR (22°)

For aircraft having a Standby Warning Lights Panel (SWLP) installed, a red ACFT CONFIG light will be illuminated whenever the following conditions exist:

- Extending the flaps past T/O APPR (22°) in flight with the SPEED BRAKE handle not in the RETRACT detent
- Extending the landing gear in flight with the SPEED BRAKE handle not in the RETRACT detent
- Advancing either power lever above the takeoff power range on the ground with the SPEED BRAKE handle not in the RETRACT detent
- Advancing either power lever above the takeoff power range on the ground with the FLAP handle set to UP (0°) or DOWN (39°), i.e., flaps not in the takeoff range of 10° or T/O APPR (22°)

J. Flap Control Circuit Breakers:

The FLAP CONT circuit breaker incorporates a switch which closes if the breaker opens (manually pulled or popped). Power through the closed MANUAL FLAP CONT circuit breaker and asymmetry switches will then energize the flap control relay. The flap control relay, in turn, energizes the flap shutoff valve open and hydraulic pressure flows to the flap selector valve. In this configuration, the EMER FLAP handle can be used to select flap position. If the MANUAL FLAP CONT circuit breaker opens (manually pulled or popped), both normal and emergency operation of the flaps is inhibited.

K. Flap Operational Description:

(1) Normal Operation:

Moving the FLAP handle from one detent to another actuates one of the four handle switches associated with the selected flap position. The switch closes and power is routed through a follow-up switch associated with the desired flap position and direction (e.g., down to 10°) to the flap control relay through flap asymmetry switches.

If the flaps are symmetrical, the flap control relay energizes. The flap shutoff relay energizes to the open position and the flap selector valve shifts to the extend or retract position. Combined, Utility or Auxiliary hydraulic system pressure then flows through the open shutoff valve to the selector valve, which directs pressure through flow regulators to the appropriate side of the hydraulic motor.

As the hydraulic motor turns, it drives the flaps in the desired direction through the central gearbox, torque shafts and screwjacks. The central gearbox at the same time also drives the horizontal stabilizer to the desired setting to maintain longitudinal stability.

When the flaps reach the desired position, the follow-up switch opens, breaking the circuit to the flap control relay. The relay de-energizes to close the flap shutoff valve and the flap selector valve is de-energized closed. Hydraulic pressure stops and the flaps stop moving.

(2) Emergency Operation:

Placing the EMERG FLAP switch to EMERG deactivates the normal FLAP handle and follow-up switches to allow direct control of the flap control relay. A red X appears on both the pilot's and copilot's normalized AOA display (lower left portion of the Primary Flight Display [PFD]). The flap control relay energizes to open the flap shutoff valve and hydraulic pressure flows to the flap selector valve. Flap asymmetry protection is still provided because flap control relay power is routed through the asymmetry switches.

Moving the EMER FLAP handle from NEUTRAL to the FLAPS UP or FLAPS DOWN position directs hydraulic pressure through the flap selector valve to the hydraulic motor. The motor operates and the flaps move. When the flaps reach the desired position, moving the EMER FLAP handle to NEUTRAL stops the flaps.

3. Controls and Indications:

(See Figure 17.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
FLAP CONT	CPO	A-1	Essential DC Bus
FLAP/STAB POS	CP	E-5	Emergency DC Bus
MANUAL FLAP CONT	CPO	B-1	Essential DC Bus
SPD BRAKE/FLAP ALARM	P	D-4	Essential DC Bus

B. Warning (Red) Messages and Annunciations:

CAS Message:	SWLP Indication	Cause or Meaning:
ACFT CONFIGURATION	ACFT CONFIG	Position of one or more of the following controls is not correct: • FLAP Handle • SPEED BRAKE Handle • Landing Gear Control Handle

C. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
FLAP ASYMMETRY (1)	Flap position asymmetry detected. Flaps have stopped moving toward selected position.

NOTE(S):

(1) For SPZ-8400 equipped aircraft having ASC 69A (Flap Asymmetry Indicator Installation).

Annunciation:	Cause or Meaning:
FLAP ASYM indicator under FLAP / STAB position indicator illuminated amber. (1)	Flap position asymmetry detected. Flaps have stopped moving toward selected position.

NOTE(S):

(1) For SPZ-8000 aircraft having ASC 69A (Flap Asymmetry Indicator Installation).

4. Limitations:

A. Maximum Landing Flaps Extended Operating Altitude:

Maximum operating altitude for extending landing flaps (39°), or flying with landing flaps extended is 20,000 ft MSL.

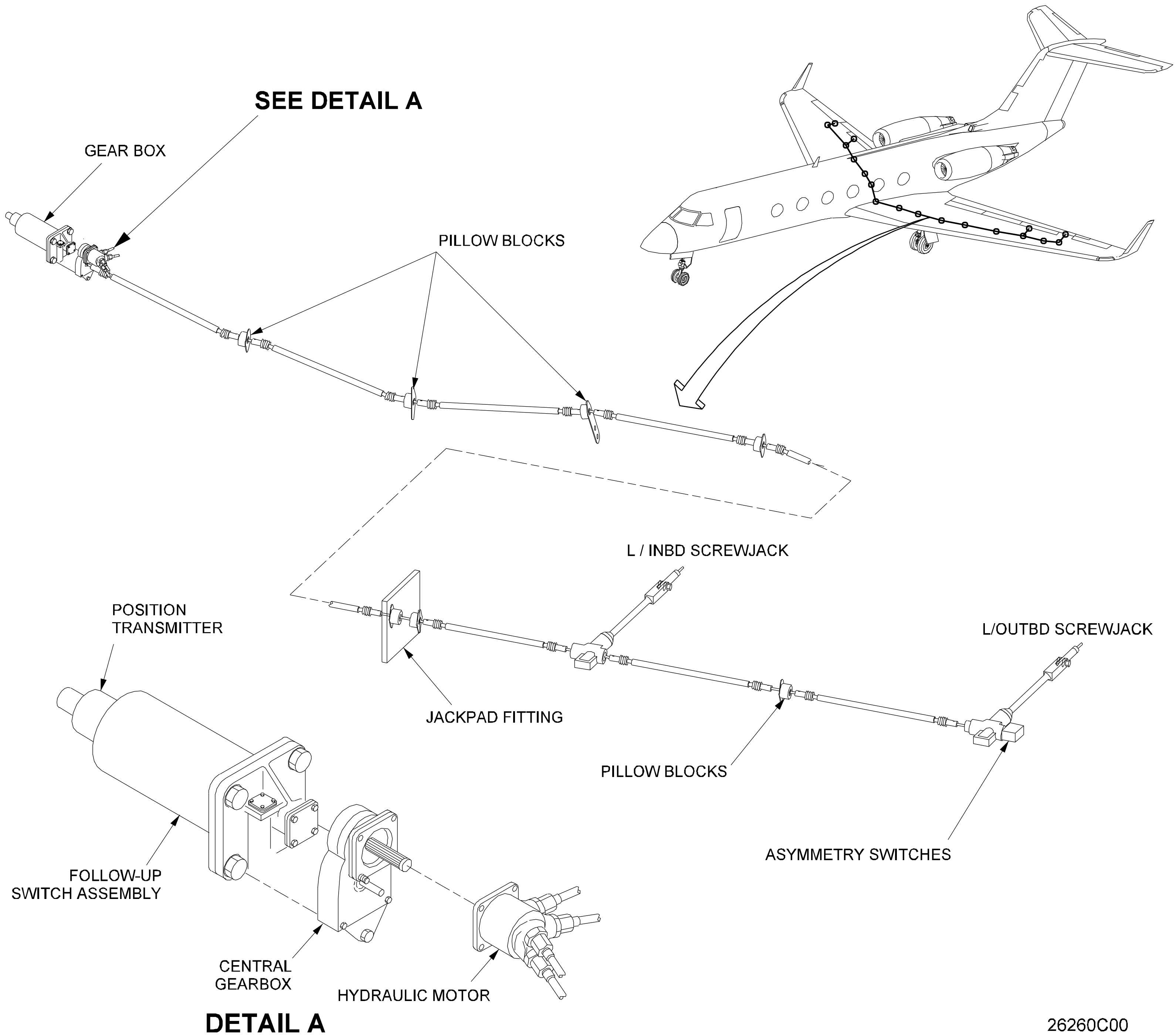
B. Flaps Extended Speeds (V_{FE} / M_{FE}):

(1) **Takeoff (10°):** 250 KCAS / 0.60 MT

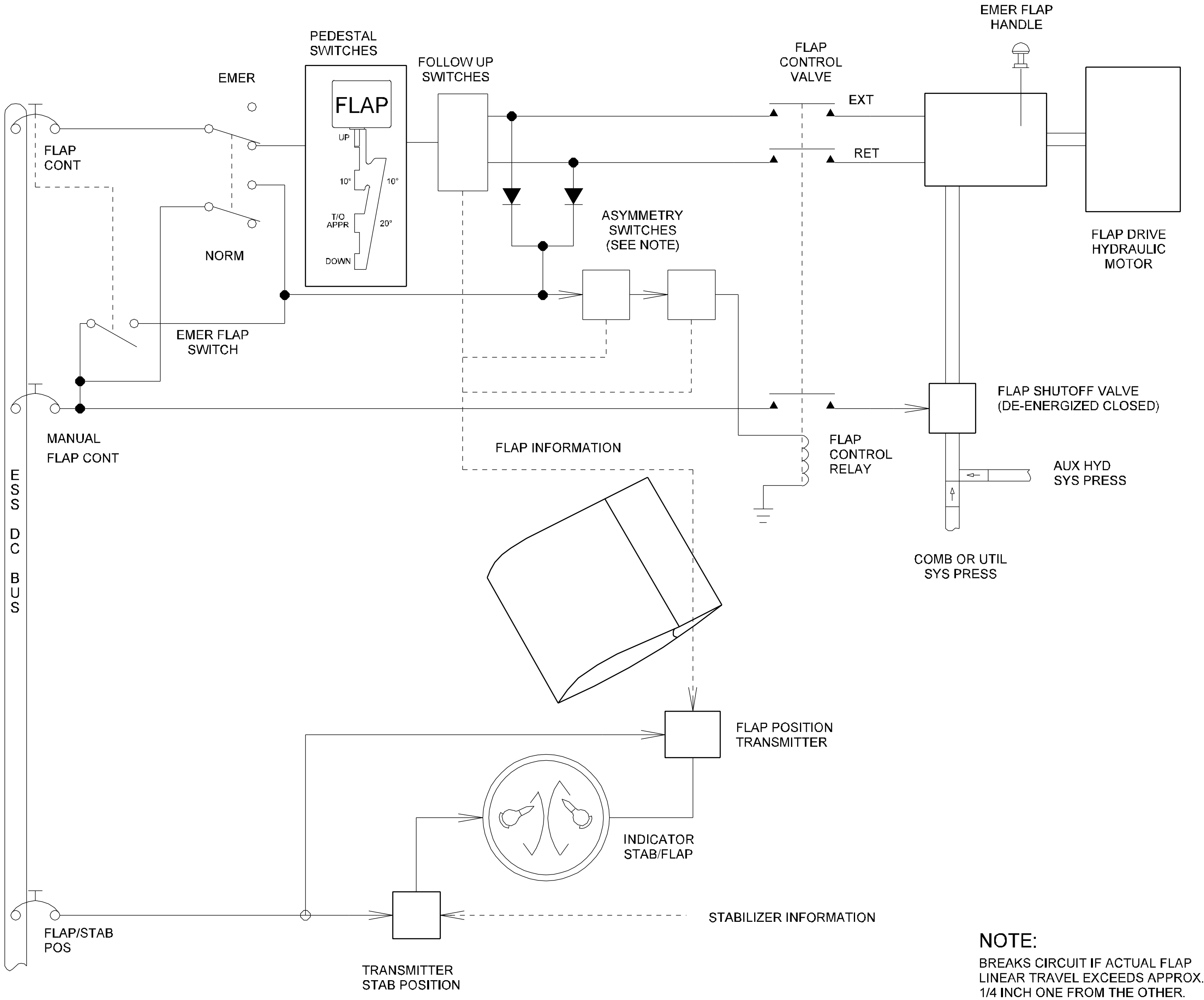
(2) **T/O APP (20°):** 220 KCAS / 0.60 MT

(3) **DOWN (39°):**

- 170 KCAS / 0.60 MT — SN 1000 through 1213 without ASC 190
- 180 KCAS / 0.60 MT — SN 1214 and subs, SN 1000 through 1213 with ASC 190



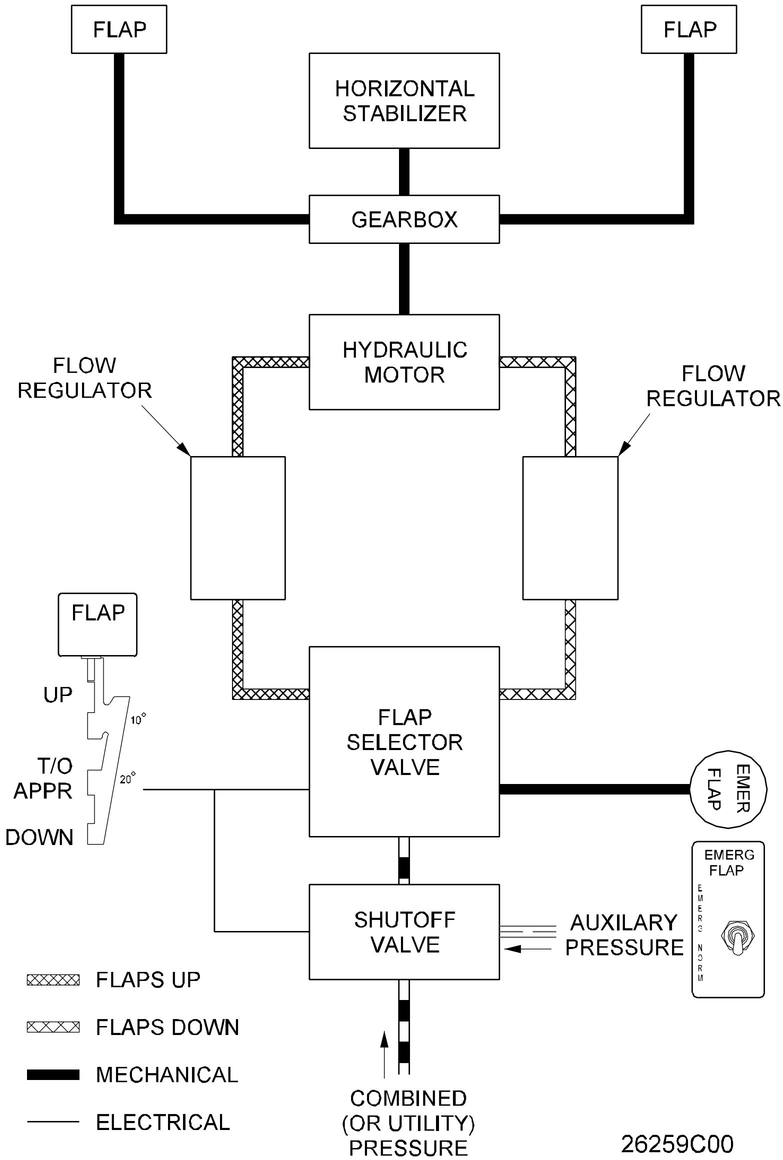
Flaps Simplified Block
Diagram
Figure 18



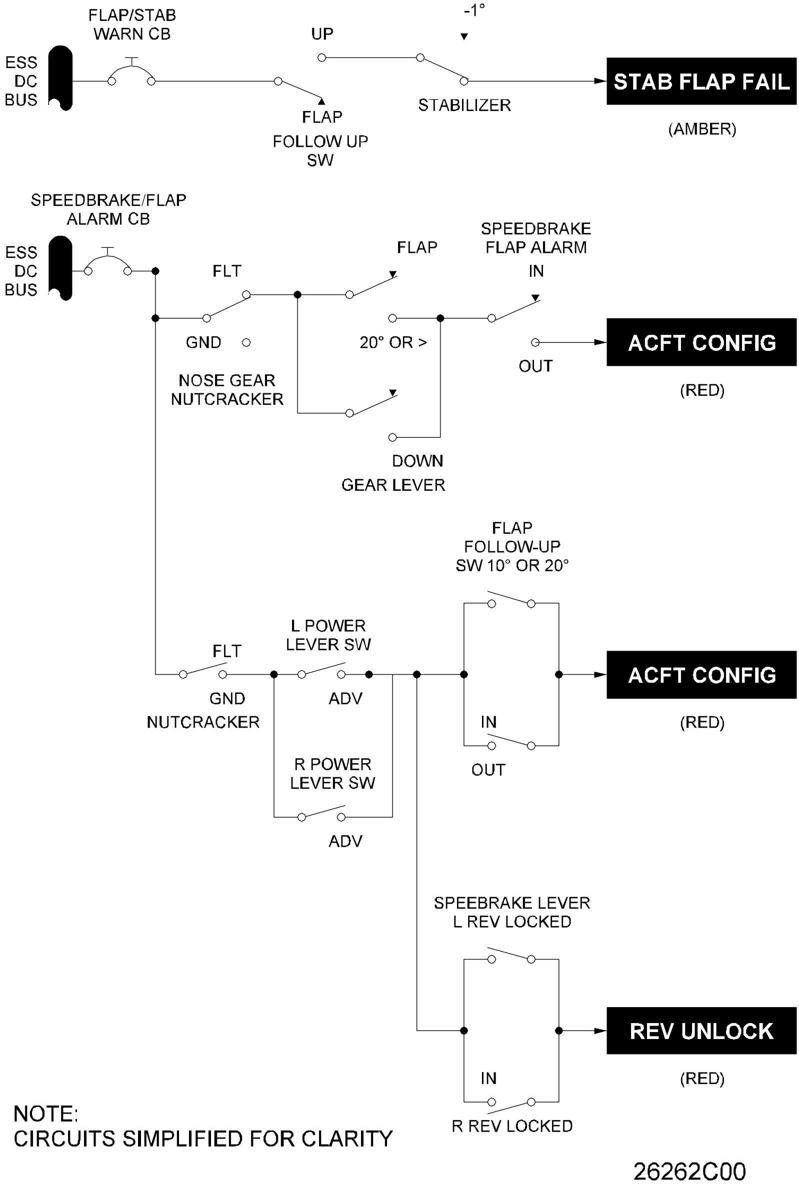
NOTE:
BREAKS CIRCUIT IF ACTUAL FLAP LINEAR TRAVEL EXCEEDS APPROX. 1/4 INCH ONE FROM THE OTHER.

26261C00

Flaps Simplified Electrical Diagram
Figure 19



Flaps / Horizontal Stabilizer System Hydraulic Control Diagram
Figure 20



Configuration Warning System Simplified Electrical Diagram
Figure 21

2A-27-70: Spoiler System

1. General Description:

The spoiler system for the Gulfstream IV assists the flight crew in maintaining roll control of the aircraft, functioning as flight spoilers. Additionally, while in flight, the spoilers extend to decrease airspeed and increase descent rate, functioning as speed brakes. On the ground after landing, they function as ground spoilers, extending to help dump any remaining lift and increase braking effectiveness.

The aircraft has three spoilers on the upper trailing edge of each wing (from inboard to outboard): a ground spoiler, an inboard flight spoiler and an outboard flight spoiler. The spoilers are manually and electrically controlled, hydraulically powered, and mechanically actuated. They are hinged to open forward when extended and close aft when retracted. Four hydraulic actuators convert hydraulic pressure to a linear mechanical force to position the spoilers.

Spoiler panel function and position depends upon the control input. Moving the SPEED BRAKE handle to the extend position extends all six spoiler panels simultaneously to function as speed brakes. Rotating the control wheel from the neutral position extends the inboard and outboard spoiler panels on the same wing as the raised aileron, functioning as flight spoilers to assist with roll control. When armed, all six spoiler panels extend automatically extend on touchdown to dump lift and increase braking effectiveness, functioning as ground spoilers. Panel function and position is summarized in the following table:

INPUT CONDITION	PANEL POSITION
Maximum Aileron, No Speed Brakes	Two Down-Wing Flight Spoilers Extended 23°, All Other Panels Retracted
Maximum Speed Brakes, No Aileron	All Six Panels Extended 26°
Maximum Speed Brakes, Maximum Aileron	Two Down-Wing Flight Spoilers Extended 55°, Two Ground Spoilers Extended 26°, Two Up-Wing Flight Spoilers Extended 26°,
Automatic Ground Spoilers	All Panels Extended 55°

The spoiler system is composed of the following subsystems, units and components:

- Flight Spoiler System
- Speed Brake System
- Ground Spoiler System
- Flight Power Shutoff System

2. Description of Subsystems, Units and Components:

A. Flight Spoiler System:

(See Figure 25.)

Flight spoilers are incorporated into the roll flight control system to improve aircraft roll response. Spoiler travel varies in proportion to the degree of roll input. The flight spoiler system is solely a hydraulically powered system, thus reversion to manual control is not possible.

As the aileron control system commands an aileron to deflect upward, a mixing linkage between the aileron and flight spoiler control systems transmits an extend command to the flight spoiler actuator servo control valve. The servo control valve then shifts to direct pressure to the flight

spoiler actuator. This allows the two outboard spoiler panels on the same side as the raised aileron to extend commensurate with the amount of roll input, up to a maximum of 23 ±2°. The opposite spoilers remain flush to the wing as that aileron travels downward.

If the speed brakes are extended, the flight spoilers may extend up to a maximum of 55 +4/-3°. The opposite spoilers remain in the position commanded as speed brakes.

B. Speed Brake System:

(See Figure 23 and Figure 25.)

The speed brake system provides a method for manual symmetrical deployment of all six spoiler panels in flight to decrease airspeed and increase descent rate. They may also be manually extended on the ground to increase braking effectiveness, should the ground spoilers not automatically extend upon landing or an aborted takeoff. This is accomplished by mechanical control of the left and right flight spoiler actuators through the use of the SPEED BRAKE handle located on the cockpit center pedestal. The speed brake system is solely a hydraulically powered system, thus reversion to manual control is not possible.

Moving the SPEED BRAKE handle out of the RETRACT detent provides simultaneous mechanical input to the left and right flight spoiler mixing linkages. Each mixing linkage then shifts the associated flight spoiler actuator servo control valve to the extend position. The servo control valve then directs Combined and Flight hydraulic system pressure to the actuator piston. The pistons extends and the attached mechanical linkage drives all six spoiler panels to a position commensurate with SPEED BRAKE handle position. Speed brake position is infinitely variable between fully retracted (0°) and fully extended (26 ±2°), depending on handle position.

With speed brakes extended, rotation of the control wheel repositions the mechanical linkage between the aileron and flight spoiler control mechanisms. Movement in either direction will further extend the outboard two (inboard and outboard flight spoiler) panels on the side of the raised aileron, up to a maximum of 55 +4/-3°. These panels will return to their original speed-brakes-extended position when the control wheel is returned to neutral.

When speed brakes are extended, a blue SPD BRAKE EXTNDED advisory message is displayed on the Crew Alerting System (CAS). In addition, the SPEED BRAKE handle will illuminate blue.

On airplanes SN 1000 through 1472 having ASC 415B and SN 1473 and subs, an amber SPD BRAKE EXTNDED caution message is displayed on CAS any time the speed brakes (flight spoilers) are extended with one or both power levers above idle. Should this message be displayed, the flight crew normally would either retract the speed brakes or reduce engine thrust to extinguish the message.

Should the fault warning computer and flight guidance computer disagree on the position of the speed brakes, a blue SPD BRAKE SWITCH advisory message is displayed on CAS.

A red ACFT CONFIGURATION warning message will be displayed on CAS, along with the associated warning tone, whenever the SPEED BRAKE handle is not in the RETRACT detent and the following conditions

occur:

- Advancing either power lever above 80% HP RPM on the ground
- Extending the flaps to DOWN (39°) or extending the landing gear in flight

For aircraft having a Standby Warning Lights Panel (SWLP) installed, a red ACFT CONFIG light will be illuminated whenever the SPEED BRAKE handle is not in the RETRACT detent and the following conditions occur:

- Advancing either power lever above the takeoff power range on the ground
- Extending the flaps past 22° or extending the landing gear in flight

C. Ground Spoiler System:

(See Figure 22 through Figure 24.)

(1) General:

The ground spoiler system provides the capability for full and automatic deployment of all six spoiler panels upon aircraft touchdown in order to dump any remaining lift and increase braking effectiveness. Also, if takeoff is aborted, the system provides for automatic deployment of all spoiler panels.

The inboard spoilers on each wing are used as ground spoilers and, through electrical control, are powered by the Combined or Flight hydraulic system. Ground spoiler operation is dependent on control signal pressure being available from the Combined, Utility or Auxiliary hydraulic system. When the system is armed (GND SPLR switch selected to ARMED), the ground spoilers extend to 55 +4/-3° upon touchdown as the power levers are retarded to the ground idle setting. Movement of the two ground spoiler actuators provides an input to the two flight spoiler actuators, causing flight spoiler extension to 55 +4/-3°. These two actions result in all six panels extending. After rollout is completed, all six spoiler panels are retracted by selection of the GND SPLR switch to OFF. Switch selection to OFF also causes a blue GND SPOILER UNARM advisory message to be displayed on CAS.

NOTE:

The GND SPLR switch is normally selected to ARMED on lineup before takeoff. If either power lever is advanced above idle after the spoilers are armed, the electrical circuit to the primary and secondary solenoid-operated hydraulic control valves is broken. The spoilers will remain stowed even though the GND SPLR switch is selected to ARMED.

If the aircraft is operating on Essential DC bus power only, the ground spoilers are inoperative, as power for the system is required from the Main DC bus. If Main DC bus power is not available and the GND SPLR switch is selected to ARMED, a red NO GND SPOILERS warning light (windshield center post) illuminates when the power levers are retarded to ground idle after landing. Since the ground spoilers are inoperative, no input is made to the flight

spoilers, thus they also do not extend. In this case, the speed brakes would be extended at the discretion of the flight crew.

The ground spoiler system incorporates two distinct warning functions. The first warning function occurs only on the ground, and is activated if the ground/flight spoilers do not automatically extend upon touchdown. This is annunciated by illumination of the red NO GND SPOILERS warning light on the windshield center post. The second warning function can occur both on the ground and in flight, and is activated if there is a failure within the ground spoiler system which might result in inadvertent spoiler extension. This is annunciated by a red GND SPOILER warning message displayed on CAS and, if installed, the SWLP.

(2) Hydraulic Operation:

The ground spoiler system requires both control pressure and operating pressure. Control pressure, also known as servo pressure, is normally supplied by the Combined hydraulic system, but can be supplied by either the Utility or Auxiliary hydraulic systems. Operating pressure is supplied by the Combined and Flight hydraulic systems. The ground spoiler system is solely a hydraulically powered system, thus reversion to manual control is not possible.

The ground spoiler hydraulic system contains primary and secondary solenoid-operated hydraulic control valves located in the main wheel well. Two ground spoiler actuators (one per side) are located at the inboard ends of the left and right wing rear beam. During normal operation, Combined and Flight hydraulic system pressure is supplied through the open flight power shutoff valve to the retract side of the ground spoiler actuators.

The ground spoiler automatic deployment feature is accomplished by a common hydraulic signal that overrides any input to the servo valve through the SPEED BRAKE handle. The override signal is supplied by Combined (or Utility or Auxiliary) hydraulic pressure to both actuators through normally-closed solenoid valves. Since the override signal is common to both the left and right actuators, the possibility of asymmetrical extension is virtually eliminated.

In the unlikely event that the ground spoilers inadvertently extend during low-speed flight, the ground spoiler control system will extend all six spoiler panels in unison to 55 +4/-3°. Should this occur at higher airspeeds, the panels will “blow back” to an angle that balances the aerodynamic load against the panels with the force applied by the actuators.

If both Combined and Flight hydraulic system pressure is lost, a bypass feature in the actuators allows the spoiler panels to “blow down” to a trail position.

(3) Operational Logic:

The ground spoiler control system will automatically extend all six spoiler panels in unison to 55 +4/-3° when the following parameters are satisfied:

- Main DC bus power is available to provide electrical power

for the system

- Combined (or Utility or Auxiliary) hydraulic pressure is available to provide servo pressure for spoiler control
- Combined or Flight hydraulic pressure is available to provide operational pressure to extend the spoilers
- GND SPLR switch is selected to ARMED
- Both power levers are retarded to ground idle
- Main landing gear Weight-On-Wheels (WOW) is sensed by the nutcracker system
- At wheel spinup greater than 57 knots (48 knots for aircraft with ASC 307 incorporated) when:
    - Flaps position is greater than 22° **OR:**
    - Flaps position is less than 22° and the GND SPLR FLAP ORIDE switch is selected to ON

(4) Ground Spoiler Warning System:

The ground spoiler system is monitored by a warning circuit that will detect certain in-flight and on-ground malfunctions within the ground spoiler system.

(a) A red GND SPOILER warning message will be displayed on CAS and, if installed, the SWLP, should any of the following events occur while in flight:

- One or both ground spoilers not fully retracted with SPEED BRAKE handle in RETRACT detent
- Primary solenoid-operated hydraulic control valve failure with secondary solenoid-operated hydraulic control valve pressurized
- Secondary solenoid-operated hydraulic control valve electrically energized
- Failure of the primary solenoid-operated hydraulic control valve nutcracker relay No. 6
- Failure of the secondary solenoid-operated hydraulic control valve nutcracker relay No. 1

Additionally, the red GND SPOILER warning message will be displayed should any of the following events occur while on the ground:

- Primary solenoid-operated hydraulic control valve failure with the power levers not in ground idle
- Unlocked ground spoiler with power levers at takeoff power

(b) A red NO GND SPOILERS warning light, located adjacent to the pilot's AOA indexer on the windshield center post, will illuminate if the ground spoilers do not automatically extend upon touchdown.

(5) Ground Spoiler System Check:

The following ground spoiler system check should be performed during the After Starting Engines checklist prior to the first departure

of each day. It provides a complete functional check of the automatic ground spoiler system and ground spoiler warning system.

(a) Verify the GND SPLR switch is selected to OFF.

(b) Verify left and right power levers are positioned to idle. Verify the following indications:

- Ground spoilers are stowed (visually check)
- NO GND SPOILERS light is extinguished
- GND SPOILER warning message is not displayed on CAS
- GND SPOILER UNARM advisory message is displayed on CAS

(c) Select the GND SPLR switch to ARMED. Verify the following indications:

- Ground spoilers are extended (visually check)
- NO GND SPOILERS light is extinguished (may flash momentarily)
- GND SPOILER warning message is not displayed on CAS
- GND SPOILER UNARM advisory message is not displayed on CAS

(d) Advance the left power lever out of idle. (See Note 1.) Verify the following indications:

- Ground spoilers are stowed (visually check)
- NO GND SPOILERS light is extinguished
- GND SPOILER warning message is not displayed on CAS (may appear momentarily)

(e) Advance the right power lever out of idle.

(f) Retard the left power lever to idle. Verify the following indications:

- Ground spoilers remain stowed (visually check)
- NO GND SPOILERS light is extinguished
- GND SPOILER warning message is not displayed on CAS

(g) Select the GND SPLR switch to OFF. Verify the following indications:

- GND SPOILER UNARM advisory message is displayed on CAS

(h) Press and hold the GND SPLR TEST switch. (See Notes 2 and 3.) Verify the following indications:

- Ground spoilers remain stowed (visually check)
- NO GND SPOILERS light is illuminated
- GND SPOILER warning message is displayed on CAS
- Both MASTER WARN lights illuminate
- GND SPOILER UNARM advisory message is

displayed on CAS

- GND SPOILER light on SWLP (if installed) is illuminated

(i) Release the GND SPLR TEST switch. Verify the following indications:

- Ground spoilers remain stowed (visually check)
- NO GND SPOILERS light is extinguished
- GND SPOILER warning message is not displayed on CAS
- Both MASTER WARN lights are extinguished
- GND SPOILER UNARM advisory message is displayed on CAS
- GND SPOILER light on SWLP (if installed) is extinguished

(j) Retard the right power lever to idle. Verify the following indications:

- NO GND SPOILERS light is extinguished
- GND SPOILER warning message is not displayed on CAS
- GND SPOILER UNARM advisory message is displayed on CAS

NOTE:

(1) When the power lever is advanced, the red GND SPOILER warning message may be displayed momentarily and then extinguish. If the message remains extinguished, continue the test.

(2) The absence of either the NO GND SPOILERS light or the GND SPOILER message during Step 6 of the ground spoiler system check, or any incorrect indication, constitutes an unsuccessful ground spoiler system check.

(3) The ground spoilers cannot always be observed from the cockpit. The correct GND SPOILER message and NO GND SPOILERS light indications are sufficient for satisfactory preflight functional verification.

D. Flight Power Shutoff System:

The flight power shutoff valve is a mechanically operated shutoff valve located between the Combined and Flight hydraulic system pressure sources and the flight and ground spoiler actuator pressure lines. The valve consists of two mechanically connected but hydraulically isolated sections. A controlex cable connects the valve to a FLIGHT POWER SHUT OFF handle located on the left aft side of the cockpit center pedestal. See Figure 8.

Moving the FLIGHT POWER SHUT OFF handle up from its stowed (horizontal) position to the vertical position mechanically closes the flight power shutoff valve. With the valve closed, operating pressure is removed from the spoiler actuators and use of the system is not possible.

In accordance with limitations established in the GIV Airplane Flight Manual, pulling the FLIGHT POWER SHUTOFF handle with speed brakes extended is prohibited.

The resultant advantage of the flight power shutoff provision is the ability to bypass a malfunctioning actuator (such as would be the need in the unlikely event of an actuator jam) and manually fly the aircraft. Although control column effort and response time to inputs are increased while in manual reversion, the aircraft remains capable of positive and harmonious control.

3. Controls and Indications:

(See Figure 23 and Figure 24.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
GND SPOILER	CPO	C-6	Left Main DC Bus
SPD BRAKE/FLAP ALARM	P	D-4	Essential DC Bus

B. Warning (Red) Messages and Annunciations:

CAS Message:	SWLP Indication	Cause or Meaning:
ACFT CONFIGURATION	ACFT CONFIG	Position of one or more of the following controls is not correct: • FLAP Handle • SPEED BRAKE Handle • Landing Gear Control Handle
GND SPOILER	GND SPOILER	Failure of ground spoiler component or deployed ground spoiler panel.

Annunciation:	Cause or Meaning:
Red NO GND SPOILERS light on windshield center post.	Failure of ground spoiler component or deployed ground spoiler panel.

C. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
SPD BRAKE EXTNDED	Speed brakes (flight spoilers) are deployed with power lever(s) above idle.

D. Advisory (Blue) Messages and Annunciations:

CAS Message:	Cause or Meaning:
GND SPOILER UNARM	Ground spoiler system is not armed.

CAS Message:	Cause or Meaning:
SPD BRAKE EXTNDED	Speed brakes are extended.
SPD BRAKE SWITCH	FWC and FGC disagree on position of speed brakes.

Annunciation:	Cause or Meaning:
SPEED BRAKE handle illuminated (pale blue).	SPEED BRAKE handle not in RETRACT detent.

4. Limitations:

A. Flight Manual Limitations:

(1) Use of Speed Brakes:

Speed brakes are not approved for extension with flaps at 39° (DOWN) or with landing gear extended in flight.

(2) Use of FLIGHT POWER SHUTOFF:

Do NOT pull FLIGHT POWER SHUTOFF handle with speed brakes extended.

(3) Automatic Ground Spoilers:

Takeoff is permitted with automatic ground spoilers inoperative, provided anti-skid is operative and 20° flaps are used for takeoff.

When the Standby Electrical System is in operation, the following limitations also apply:

(4) Use of Speed Brakes:

Speed brakes may be used, however, operation should be slow (approximately five [5] seconds for full range movement).

(5) Landing with Standby Electrical System Operating:

Landing is approved provided automatic ground spoilers and thrust reversers are not used for landing. See Section 05-17-50, Landing With Standby Electrical Power System Operating.

B. Other Operational Information:

(1) Speed Brake Operating Envelope:

Speed brake operation is permitted within the flight envelope up to and including V_{MO}/M_{MO}.

(2) Speed Brake Operating Altitude:

The use of speed brakes at altitudes below 2,000 feet AGL is not recommended.

(3) Effects of Speed Brake Operation:

A mild nose-up trim change and some buffet may be expected when speed brakes are fully extended. For passenger comfort, it is recommended that speed brakes be extended and retracted slowly, retrimming as required.

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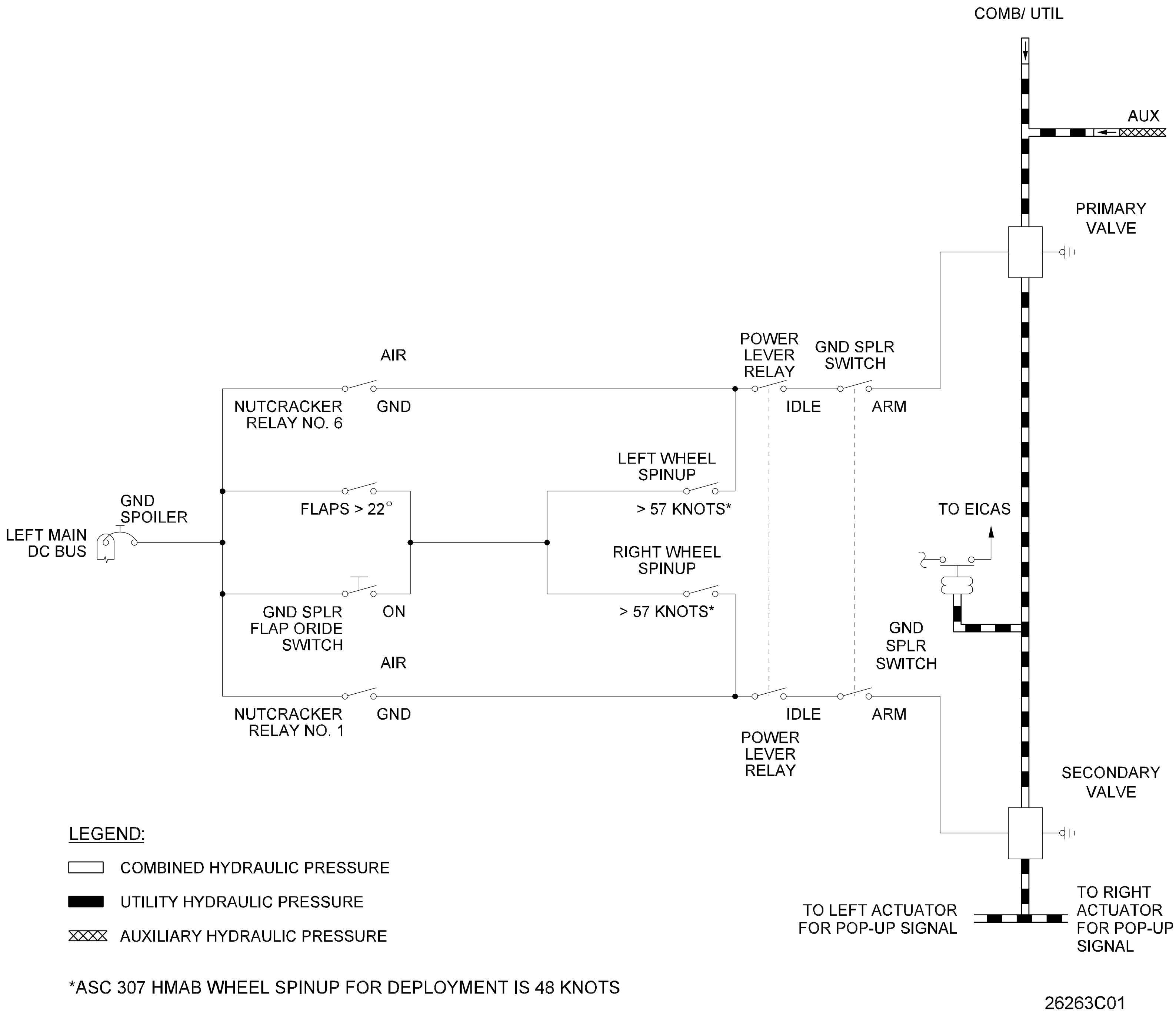
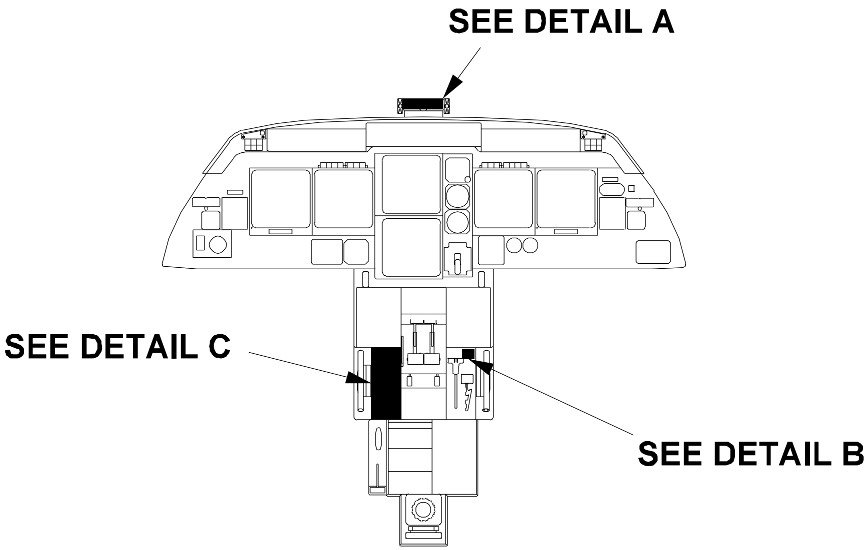
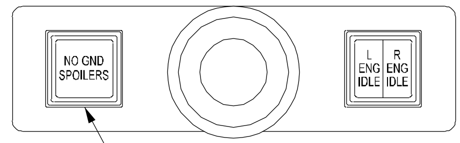


Spoiler Control System
Simplified Electrical
Diagram
Figure 22





NO GND SPOILERS
Illuminates red to warn the flight crew that the ground spoilers did not automatically extend upon touchdown.

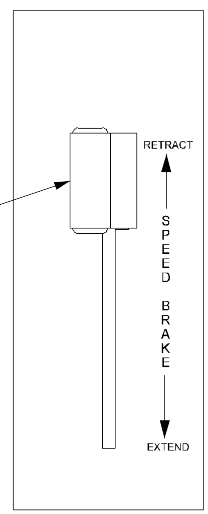
DETAIL A

SPEED BRAKE

- Handle not in RETRACT detent:
  - • All six spoiler panels extend commensurate with SPEED BRAKE handle position. Speed brake position is infinitely variable between fully retracted (0^) and fully extended (26 | 2^), depending on handle position.
  - • SPEED BRAKE handle illuminates blue.
  - • A blue SPD BRAKE EXTNDED advisory message is displayed on CAS.

Note: On SPZ-8400 equipped airplanes with ASC 415B incorporated, an amber SPD BRAKE EXTNDED caution CAS message will be displayed if speed brakes (flight spoilers) are deployed with power lever(s) above idle.

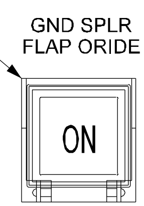
- Handle in RETRACT detent:
  - • All six spoiler panels are retracted.



DETAIL C

GND SPLR FLAP ORIDE

- ON:
  - • Amber ON legend is illuminated.
  - • Automatic ground spoiler extension will occur with flaps less than 22^, provided all other conditions are satisfied.
- Off:
  - • Amber ON legend is extinguished.
  - • Automatic ground spoiler extension will NOT occur with flaps less than 22^, even with all other conditions satisfied.



DETAIL B

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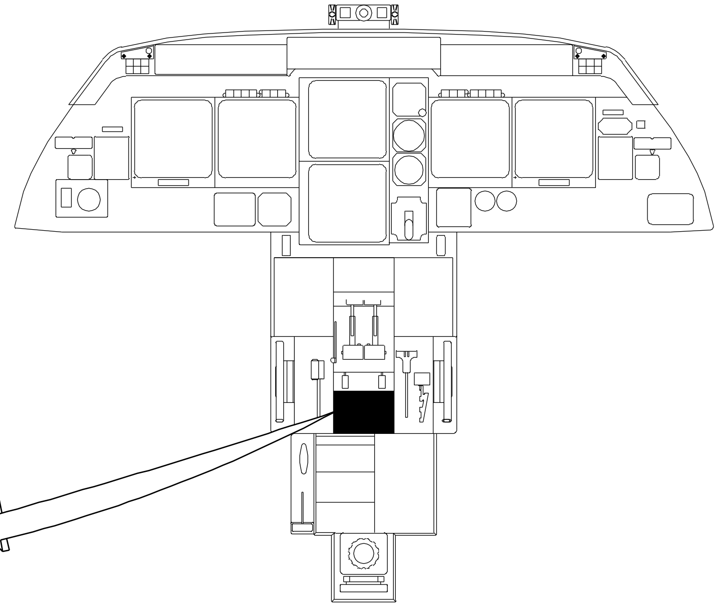
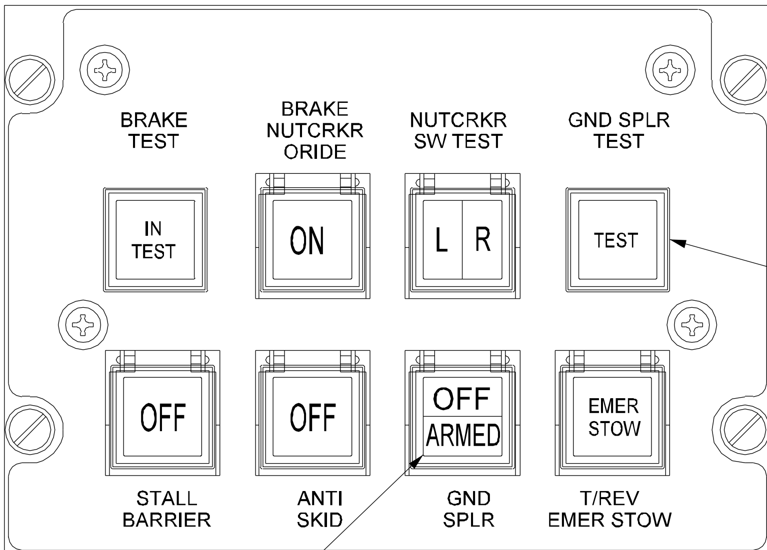
Spoiler/Speed Brake Controls and Indications
Figure 23





GND SPLR TEST
Used to test ground spoiler annunciations during ground spoiler checkout. The ground spoilers remain stowed. When depressed and held, the following annunciations are displayed:
- Blue TEST legend in switch illuminates.
- Red NO GND SPOILERS light is illuminated.
- Red GND SPOILER warning message is displayed on CAS.
- Red GND SPOILER light on SWLP (if installed) is illuminated.
- Both red MASTER WARN lights illuminate.
- Blue GND SPOILER UNARM advisory message is displayed on CAS.

GND SPLR
- OFF:
  - • Amber OFF legend in switch is illuminated.
  - • Ground spoilers are disabled.
  - • If extended, ground spoilers retract.
  - • A blue GND SPOILER UNARM advisory message is displayed on CAS.
- ARMED:
  - • Amber OFF legend in switch is extinguished.
  - • ARMED legend in switch is illuminated.
  - • Ground spoilers are armed.
  - • If all conditions are satisfied, ground spoilers extend when power levers are retarded to ground idle.

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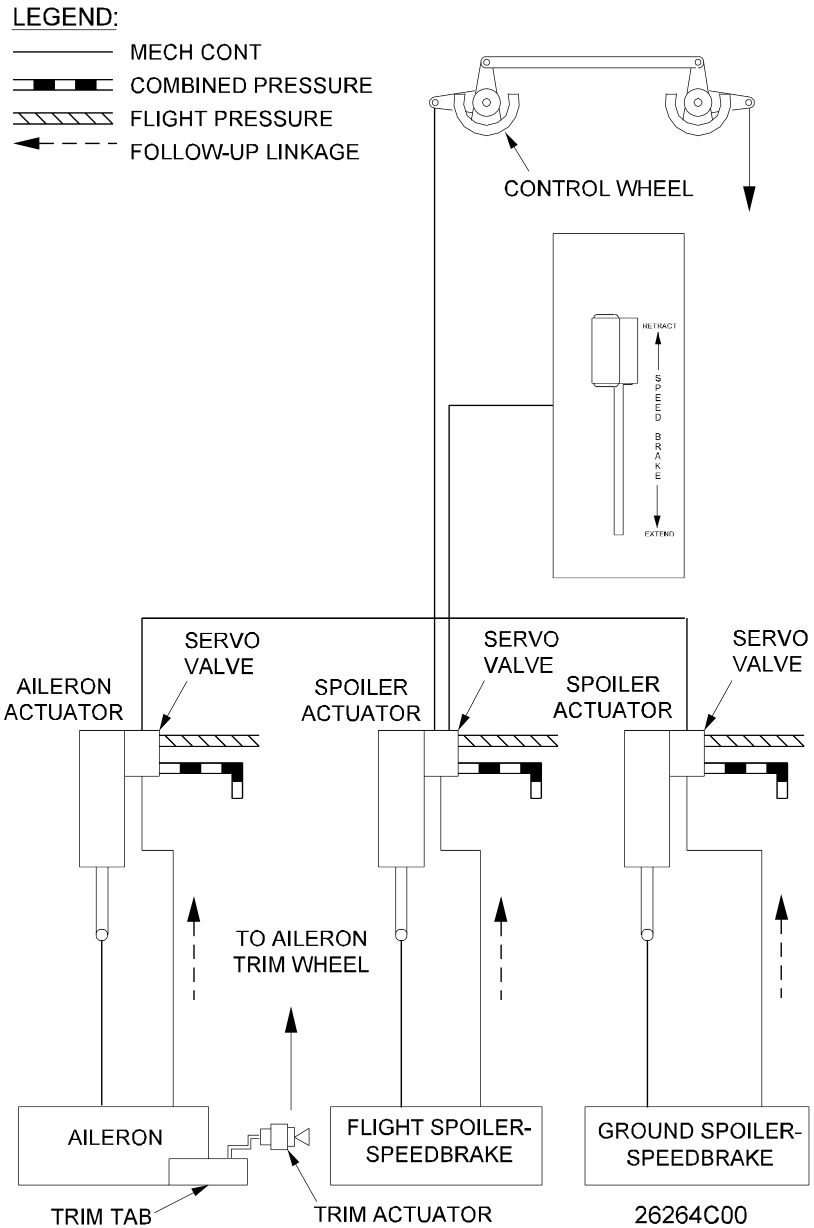
Ground Spoiler Controls
and Indications
Figure 24



Spoiler/Speed Brake Control System Simplified Block Diagram
Figure 25

2A-27-80: Gust Lock System

1. **General Description:**

The gust lock system for the Gulfstream IV provides a means for the flight crew to manually protect the unpowered flight control surfaces from movement by wind gusts while the aircraft is on the ground.

The gust lock is a mechanical ground safety system that neither affects the flight performance of the aircraft nor receives any flight loads. The ailerons, elevators and rudder are locked against gust loads by mechanical latches operated by the GUST LOCK handle located on the cockpit center pedestal.

2. **Description of Subsystems, Units and Components:**

   A. **Surface Lock System:**

   (See Figure 26.)

   A single T-shaped handle, located on the right side of the cockpit center pedestal and labeled GUST LOCK, controls the gust lock system. A spring loaded trigger is incorporated in the gust lock handle to prevent the handle from inadvertently being pulled. Releasing the trigger and then raising and pulling the GUST LOCK handle aft actuates conventional mechanical linkage consisting of cables, springs, latches and a bungee rod. Moving the ailerons and rudder to the neutral position and the elevator to the trailing edge down position allows the gust lock to engage and lock the flight controls as their linkages reach the locking position. Releasing the trigger and then lowering the GUST LOCK handle releases the gust lock.

   Safety features prevent the gust lock from inadvertently engaging or a failure of the system preventing gust lock release. With the gust lock released, the bungee rod acts as a fixed rod to prevent inadvertent flight control locking. If the gust lock fails when engaged, the springs will unlock the gust lock.

   B. **Mechanical Power Lever Interlock:**

   A mechanical interlock is incorporated in the GUST LOCK handle mechanism that restricts simultaneous movement of the power levers to a maximum of six percent above ground idle with the gust lock engaged. Force applied to advance both power levers simultaneously cannot override the interlock. To prevent any hydraulic forces acting upon an engaged gust lock, the gust lock should be released prior to engine starting and not engaged until all hydraulic pressures read zero.

3. **Controls and Indications:**

(See Figure 26.)

4. **Limitations:**

   A. **Flight Manual Limitations:**

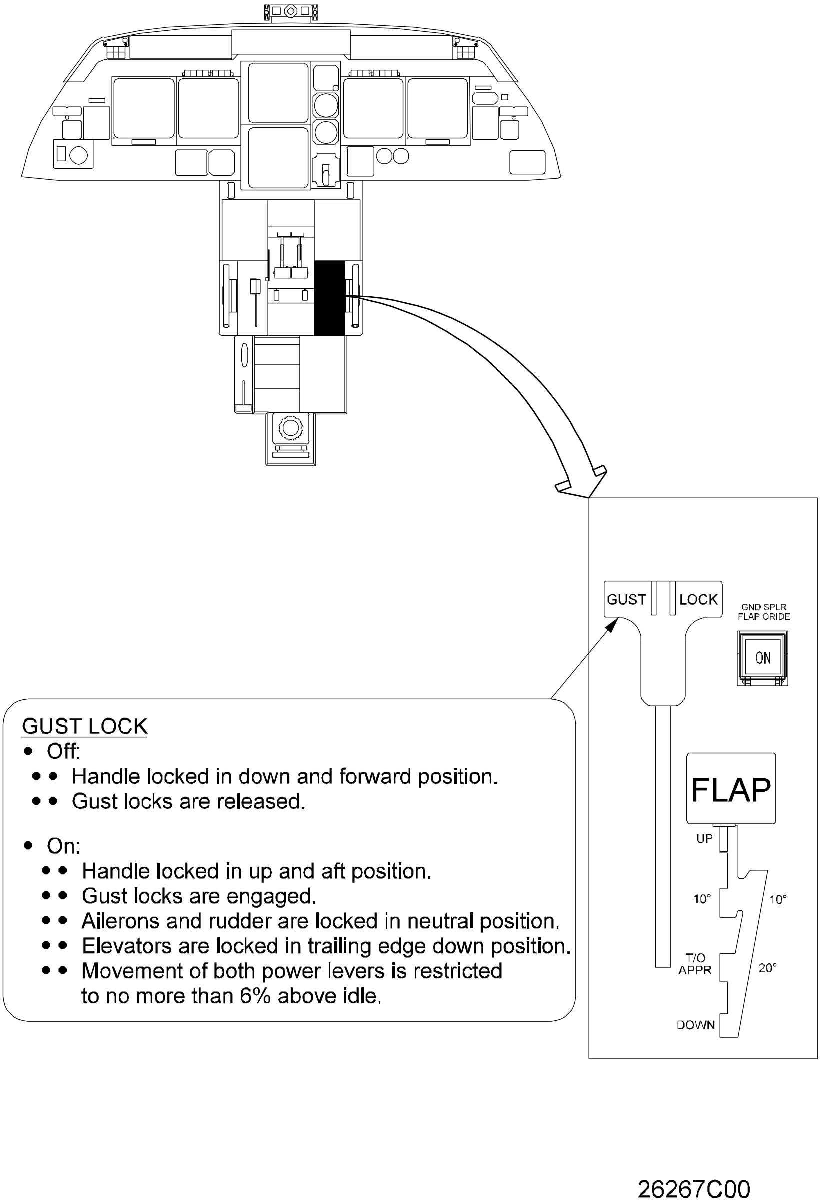
   There are no limitations for the gust lock system at the time of this revision.

B. Other Operational Limitations:

The gust lock is effective in protecting the flight controls in wind gusts up to 60 knots.

CAUTION

ENSURE HYDRAULIC PRESSURE IS DEPLETED PRIOR TO ENGAGING GUST LOCK. CYCLE THE CONTROLS WITH THE CONTROL COLUMN, CONTROL WHEEL AND RUDDER PEDALS TO DEPLETE ANY RESIDUAL PRESSURE.



GUST LOCK Handle
Figure 26

2A-28-10: General

1. Introduction:

The fuel system contains and supplies all the necessary fuel for the two turbofan engines and the Auxiliary Power Unit (APU). There are two integral (wet wing) fuel tanks, each formed by the respective wing's structure. Low pressure fuel is delivered from the tanks to the engine-driven pumps and the APU.

The fuel tanks can be fueled by the conventional "over-the-wing" method or from a single-point pressure fueling adapter. Defueling may be accomplished by suction or pumping; any remaining pockets of fuel may then be drained.

Each wing tank contains a fuel hopper as a separate compartment within the tank. Low pressure fuel is pumped from the hopper through a fuel feed line to the engine-driven pumps by fuel boost pumps (two per wing). A hydraulic fluid-to-fuel heat exchanger inside each hopper is used for cooling hydraulic fluid.

An intertank valve in the right fuel hopper allows simultaneous defueling of both tanks. A crossflow shutoff valve in the left hopper allows fuel balance to be adjusted.

A fuel temperature bulb installed in the left hopper provides a signal for cockpit indications of fuel temperature.

A tank vent system provides adequate venting while the aircraft is on the ground. In flight, this same system slightly pressurizes the tanks.

2. Subsystems Within the Fuel System:

The fuel system is divided into the following subsystems:

- 2A-28-20: Fuel Storage System
- 2A-28-30: Fuel Distribution System
- 2A-28-40: Fuel Indication System

3. Engine Fuel Grades:

A. Kerosene Type:

Fuel conforming to any of the following specifications is approved for use. Fuels conforming to ASTM Specification ES2-74 are also eligible. Mixing of fuels is permissible.

Kerosene Type		
American	**British**	**Canadian**
ASTM D1655-89, Jet A ASTM D1655-89, Jet A-1 MIL-T-83133A, Grade JP-8	DEF STAN 91-87 DEF STAN 91-91	CAN/CGSB-3.23-M86
French	**USSR**	**I.A.T.A.**
Air 3405/C	T-1, TS-1, RT (GOST 10227-86) T-7 (GOST 12308-66)	1988 Kerosene Type

B. Wide Cut JP-4 Type:

Fuel conforming to any of the following specifications is approved for use. Fuels conforming to ASTM Specification ES2-74 are also eligible. Mixing of

fuels is permissible.

Wide Cut JP-4 Type		
American	**British**	**Canadian**
ASTM D1655-89, Jet B MIL-T-5624N, Grade JP-4	DEF STAN 91-88	CAN/CGSB-3.22-M86
French	**I.A.T.A.**	
Air 3407/B	1987 JP-4	

C. High Flash Point JP-5 Type:

Fuel conforming to any of the following specifications is approved for use. Fuels conforming to ASTM Specification ES2-74 are also eligible. Mixing of fuels is permissible.

High Flash Point JP-5 Type		
American	**British**	**Canadian**
MIL-T-5624N, Grade JP-5	DEF STAN 91-86	CAN 3-GP-24Ma
French		
Air 3404/C		

NOTE:

The use of Wide Cut fuel, as agreed by the Operator, Rolls-Royce and the appropriate Airworthiness Authority, may result in a reduction in HP Fuel Pump life.

4. Fuel Additives:

The following fuel additives (in addition to those included in DEF STAN Specifications) are approved by Rolls-Royce, subject to limitations stated:

A. Corrosion Inhibitor / Lubricity Aids:

	Concentration Range - Lb 42,035 Gallons (US) / 35,000 Gallons (IMP)	
Additive:	**Minimum:**	**Maximum:**
HITEC 515	4 (11 mg/l)	7.5 (21 mg/l)
APOLLO PRI 19	3 (9 mg/l)	8 (23 mg/l)
TOLAD 245	7.5 (21 mg/l)	12 (34 mg/l)
DUPONT DCI-4A	3 (9 mg/l)	8 (23 mg/l)
HITEC 580	3 (9 mg/l)	8 (23 mg/l)

NOTE:

Minimum requirement is to ensure that sufficient additive is available when it is required to act as a lubricity aid.

B. Anti-Icing Additive:

- DEF STAN 68-252 **or:**

- MIL-I-27686E **or:**
- Any direct equivalent in concentrations not exceeding 0.15 percent by volume

C. Static Dissipater Additive:

- Shell A.S.A. 3 in concentrations of not more than 1.00 parts per million
- Stadis 450 maximum concentration not more than 3.00 parts per million

D. Anti-Microbiological Additive:

(1) Methyl Cellosolve:

Methyl cellosolve may be used. Refer to GIV Maintenance Manual for additive application procedures.

(2) Biobor JF:

Biobor JF may be used. Refer to GIV Maintenance Manual for additive application procedures.

(3) Kathon FP 1.5:

Kathon FP 1.5 may be used. Refer to GIV Maintenance Manual for additive application procedures.

NOTE:

Under certain conditions, solid matter may be precipitated from fuel containing Biobor JF or Kathon FP 1.5 during flight. The fuel differential pressure signals should be carefully monitored in flight immediately following its use in the airplane tanks. Refer to Rolls-Royce Tay Maintenance M-TAY-1RR for recommended procedures to be followed when using Biobor JF or Kathon FP 1.5.

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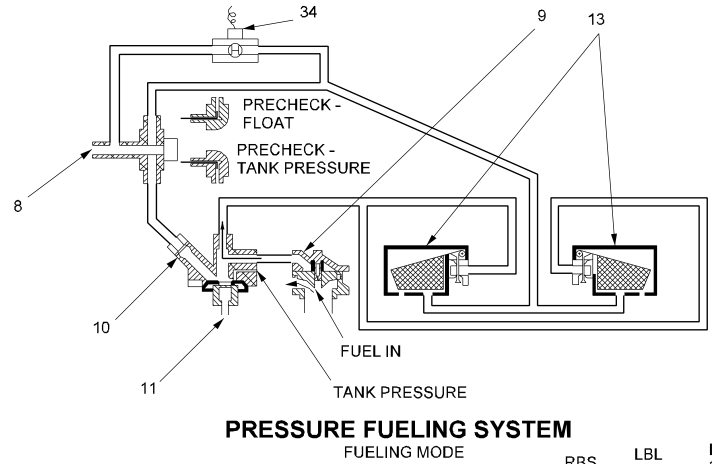
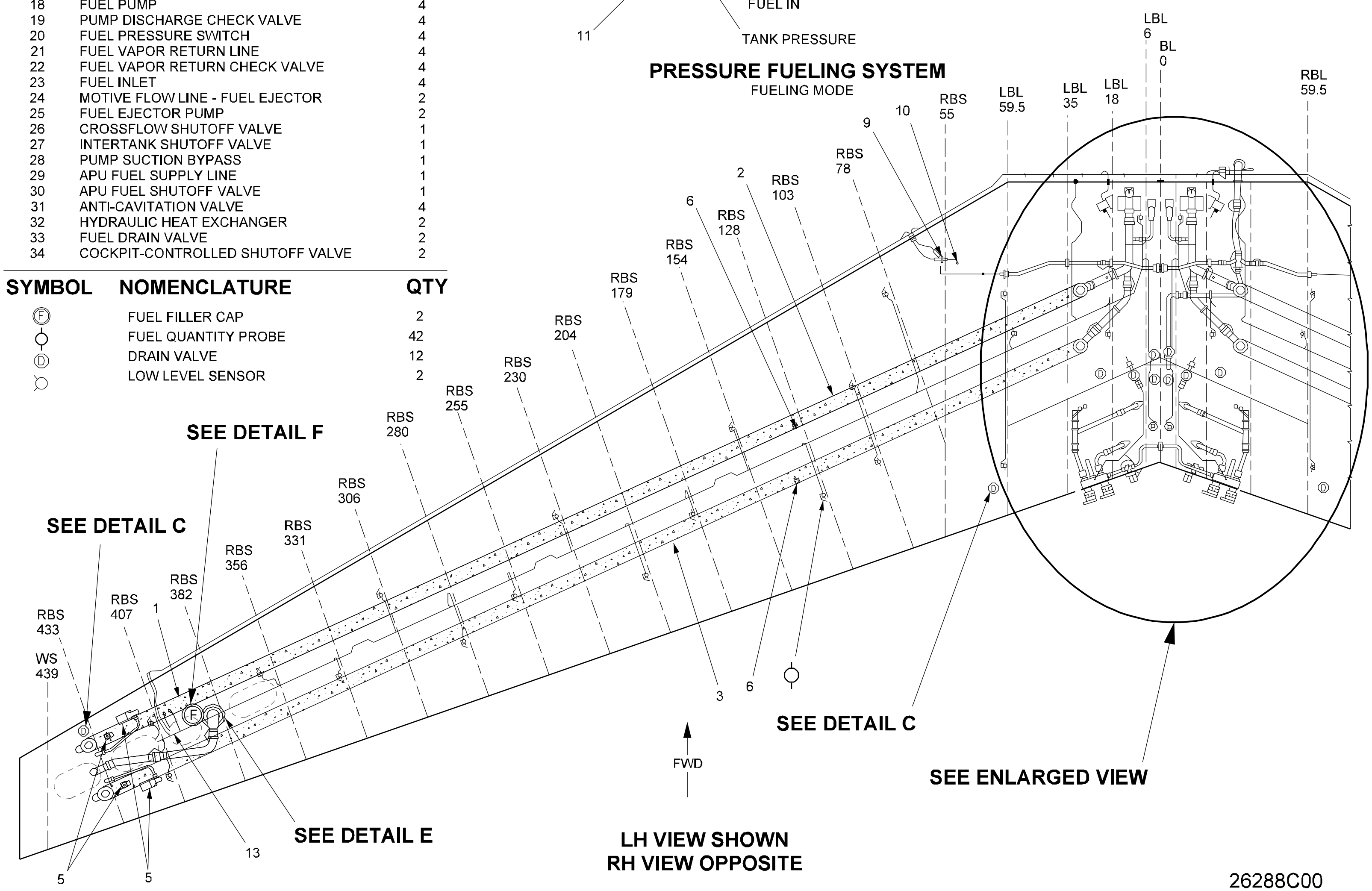
ITEM	NOMENCLATURE	QTY
1	VENT FLOAT / RELIEF VALVE	4
2	VENT LINE - FWD	2
3	VENT LINE - AFT	2
4	VENT - OVERBOARD	2
5	VENT FLOAT / NON-RELIEVING VALVE	8
6	VENT FLOAT / DRAIN VALVE	6
7	PRESSURE FUELING ADAPTER	1
8	PRESSURE FUELING PRECHECK VALVE	2
9	PRESSURE FUELING SHUTOFF VALVE	2
10	PRESSURE FUELING SENSING VALVE	2
11	PRESSURE FUELING AMBIENT PRESSURE PORT	2
12	ENGINE FUEL SHUTOFF VALVE	2
13	PRESSURE FUELING HIGH LEVEL PILOT	4
14	SUCTION DEFUEL VALVE	1
15	SUCTION DEFUEL LINE	1
16	FUEL HOPPER	2
17	FUEL PUMP MANIFOLD	2
18	FUEL PUMP	4
19	PUMP DISCHARGE CHECK VALVE	4
20	FUEL PRESSURE SWITCH	4
21	FUEL VAPOR RETURN LINE	4
22	FUEL VAPOR RETURN CHECK VALVE	4
23	FUEL INLET	4
24	MOTIVE FLOW LINE - FUEL EJECTOR	2
25	FUEL EJECTOR PUMP	2
26	CROSSFLOW SHUTOFF VALVE	1
27	INTERTANK SHUTOFF VALVE	1
28	PUMP SUCTION BYPASS	1
29	APU FUEL SUPPLY LINE	1
30	APU FUEL SHUTOFF VALVE	1
31	ANTI-CAVITATION VALVE	4
32	HYDRAULIC HEAT EXCHANGER	2
33	FUEL DRAIN VALVE	2
34	COCKPIT-CONTROLLED SHUTOFF VALVE	2

SYMBOL	NOMENCLATURE	QTY
F	FUEL FILLER CAP	2
	FUEL QUANTITY PROBE	42
D	DRAIN VALVE	12
	LOW LEVEL SENSOR	2





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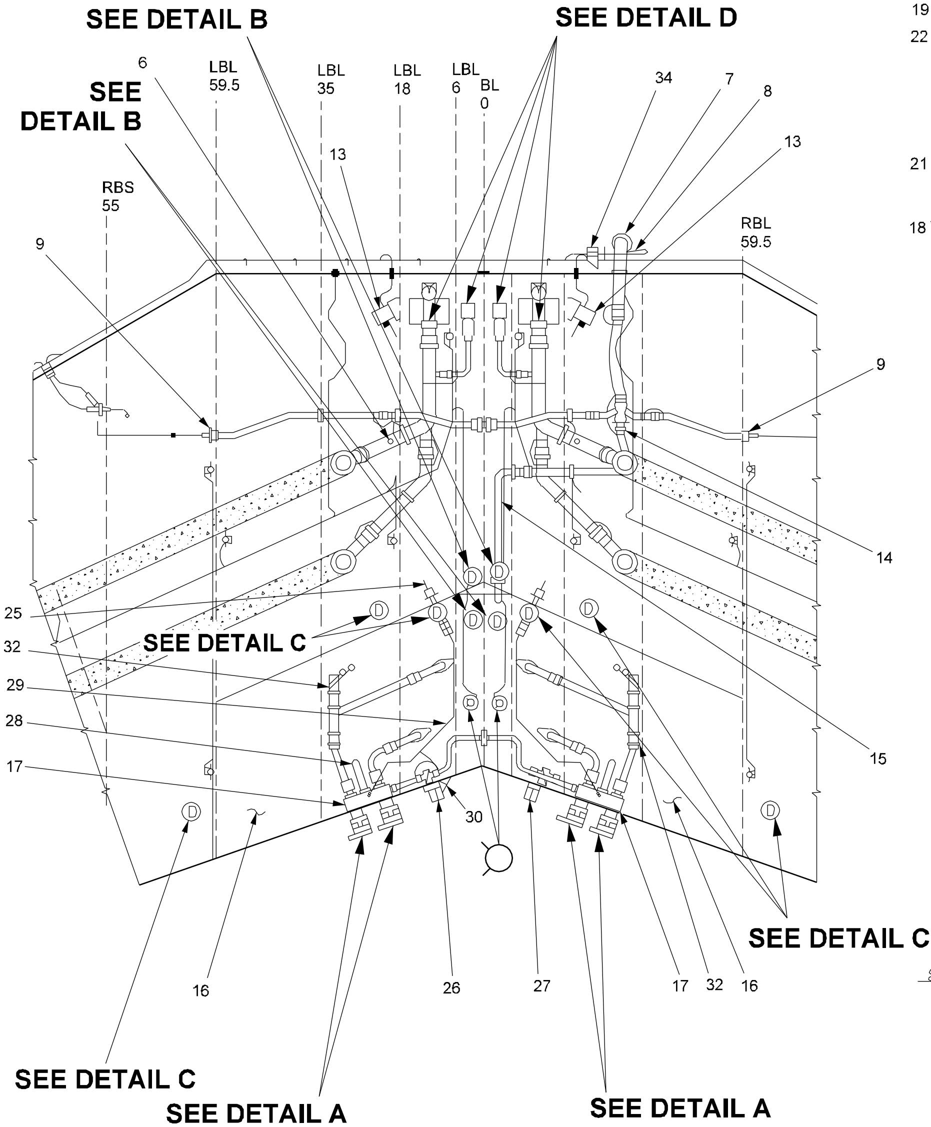
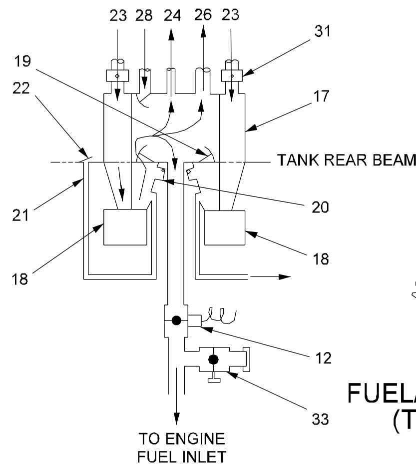
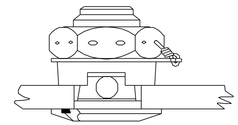
Fuel System Simplified Block Diagram, Sheet 1 of 2
Figure 1



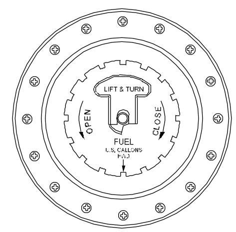
ENLARGED VIEW
LH VIEW SHOWN
RH VIEW OPPOSITE



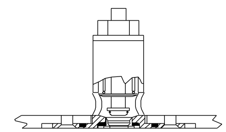
DETAIL A
(TYPICAL 2 PLACES)



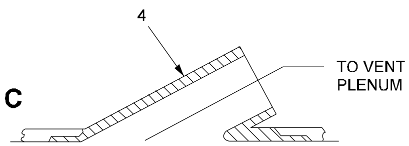
DETAIL B
FUEL/WATER DRAIN VALVE
(TYPICAL 4 PLACES)



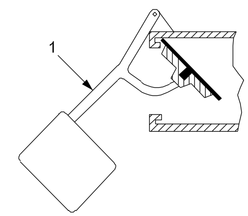
DETAIL F
(TYPICAL 2 PLACES)



DETAIL C
WATER DRAIN VALVE
(TYPICAL 8 PLACES)



DETAIL E
(TYPICAL 2 PLACES)



DETAIL D
(TYPICAL 4 PLACES)

SYMBOL	NOMENCLATURE	QTY
F	FUEL FILLER CAP	2
	FUEL QUANTITY PROBE	42
D	DRAIN VALVE	12
	LOW LEVEL SENSOR	2

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Fuel System Simplified
Block Diagram, Sheet 2 of
2
Figure 2

2A-28-20: Fuel Storage System

1. **General Description:**

   The fuel storage system contains all the necessary fuel for the aircraft engines and the Auxiliary Power Unit (APU). It is composed of the following subsystems, units and components:

   - Wing Fuel Tanks
   - Fuel Hoppers
   - Gravity Water / Fuel Drain System
   - Fuel Ventilation System
   - Over-Wing (Gravity) Fueling System

2. **Description of Subsystems, Units and Components:**

   A. **Wing Fuel Tanks:**

   The integral wing tanks include most of the internal wing structure between the forward and rear spars. The centerline fuselage rib at Buttock Line Zero (BL0) and a sealed rib at Rib Body Station (RBS) 433 form the inboard and outboard edges of each wing tank, while the wing planks form the upper and lower confines.

   The external underside of the centerline rib is protected by a non-sparking skeg panel that serves to protect the underside of the aircraft in the event of a gear-up landing.

   Partially sealed sheet metal ribs form part of the wing structure and divide each wing tank into five compartments. The ribs act as baffles to prevent fuel sloshing and sudden center-of-gravity changes that could result as fuel moves about a partially filled tank. One-way flapper valves also control fuel movement by opening to allow fuel movement inward while preventing outward movement.

   Each wing tank has a fuel filler cap on the upper wing surface near the outboard end of the fuel tank.

   B. **Fuel Hoppers:**

   A 190 U.S. gallon (719 liter) fuel hopper in each wing tank extends laterally from the centerline rib to the wing's first structural rib. The hopper's forward wall and the rear wing spar form the forward and aft confines. Flapper valves in the forward wall open to allow hopper refilling during fueling or when the wing fuel tank fuel level is higher than hopper fuel level.

   Each hopper supplies its respective engine with fuel. The left hopper also supplies the APU. Each hopper contains the following:

   - Fuel pump manifold and associated plumbing
   - Fuel tank low level sensor
   - Hydraulic fluid heat exchanger
   - Fuel quantity probe with compensator
   - Motive flow ejector pump

   A crossflow shutoff valve in the left hopper allows fuel balance to be

adjusted. A fuel temperature bulb provides a signal for cockpit indications of fuel temperature.

Simultaneous defueling of both tanks is provided through a suction defueling line and an intertank shutoff valve inside the right fuel hopper.

A motive flow ejector pump, supplied with motive fuel flow from the fuel pump manifold, transfers fuel from respective fuel tank into the hopper faster than the engine can consume it at maximum cruise power. Excess fuel then overflows the hopper back into the wing tank. With a supply pressure of 25 psig and a maximum motive flow of approximately 750 Pounds Per Hour (PPH) at the ejector pump's inlet, the ejector pump produces a minimum induced flow of approximately 4500 PPH. This assists the boost pumps in maintaining the required fuel pressure at the engine-driven pump inlets.

C. Gravity Water / Fuel Drain System:

Two types of manually operated drain valves allow fuel sampling, draining of accumulated water and gravity draining of the wing tanks and fuel hopper. The majority of the drain valves are located on the bottom of the wing, adjacent to the fuselage centerline. Each fuel tank also has a drain valve near the rear spar at the wing root and another valve in the vent plenum near the wing tip.

The four valves closest to the fuselage centerline are protruding type valves and are placarded as fuel / water low point drains. Pushing the valve up with a Phillips screwdriver, then rotating the valve 90° locks the valve open, allowing draining to occur. Rotating the valve stem 90° (or more if necessary) in the opposite direction reseats and closes the valve.

The remaining valves are flush type valves. They are opened by pushing the valve inward with an appropriate tool and rotating the valve stem 90°. Rotating the valve stem 90° (or more if necessary) in the opposite direction reseats and closes the valve.

D. Fuel Ventilation System:

The fuel ventilation system is an open vent system that slightly pressurizes the fuel during flight and prevents tank over-pressurization during fueling operations. Operation of the system is fully automatic.

Two vent ducts attached to the inside of the upper wing surface run from a Y-pipe assembly near the wing center section to the vent plenum in the wingtip. The Y-pipe assembly connects to a float-operated vent / relief valve that allows tank venting and prevents tank over-pressurization during refueling by venting the inboard part of the tank through the vent ducts to the vent plenum. Float-operated vent valves, downstream of the Y-pipe assembly, drain the fuel vent system.

A non-relieving float-operated vent valve connects each vent duct to the vent plenum. Two smaller float valves provide additional venting for the vent valves. A flush vent then connects the vent plenum to the atmosphere through a flush NACA-type vent on the wing's lower surface. In flight, the NACA duct slightly pressurizes the wing tank.

E. Over-Wing (Gravity) Fueling System:

The over-wing (gravity) fueling system provides an alternate method of fueling the tanks if pressure-fueling equipment is not available. An over-

wing (gravity) fueling adapter assembly is installed in each wing fuel tank. The adapter has a standpipe filler neck that limits wing tank capacity to approximately 2,185 gallons (8,271 liters). This equates to approximately 14,750 pounds (6,690 kg) total per tank.

During gravity fueling, fuel is allowed to travel inboard, through the ribs, to the aircraft centerline. This path fills the aircraft from the centerline rib, including the hopper, outboard.

A grounding wire receptacle is incorporated in to the wing leading edge so that the gravity fueling nozzle may be grounded during fueling. Gulfstream recommends gravity fueling take place over the leading edge of the wing, rather than the trailing edge. Procedures for over-wing (gravity) fueling are presented in Chapter 9: Handling and Servicing.

3. Limitations:

A. Unbalanced Fuel:

For maximum unbalanced fuel, see Section 01-28-10, Usable Fuel Capacities. Before the unbalance exceeds that shown, proceed with fuel balancing.

Fuel load balancing may be accomplished by using the crossflow valve or intertank valve.

When balancing fuel through use of the crossflow valve, ensure that boosted fuel pressure is always available to the engines.

CAUTION

THE ENGINE WILL RUN ON SUCTION FUEL FEED ONLY AT OR BELOW 20,000 FEET. ABOVE 20,000 FEET, THE ENGINE WILL RUN ERRATICALLY AND FLAME OUT IF THE CROSSFLOW IS NOT OPEN WITH AT LEAST ONE BOOST PUMP ON.

Balancing fuel by using the intertank valve requires the airplane to be placed in a sideslip condition. Adjusting the rudder trim in the direction of the “heavy” tank will create a wing down condition and allow fuel to flow toward the “light” tank.

B. Usable Fuel Capacities:

Left Tank	Right Tank	Total
14,750 lb (6,690 kg)	14,750 lb (6,690 kg)	29,500 lb (13,380 kg)
2,185 gal (8,271 L.)	2,185 gal (8,271 L.)	4,370 gal (16,542 L.)

NOTE:

It is possible to upload fuel in excess of 29,500 lb. This is permitted as long as the maximum ramp weight and / or the maximum takeoff weight is not exceeded, and the loaded aircraft center of gravity is within limits.

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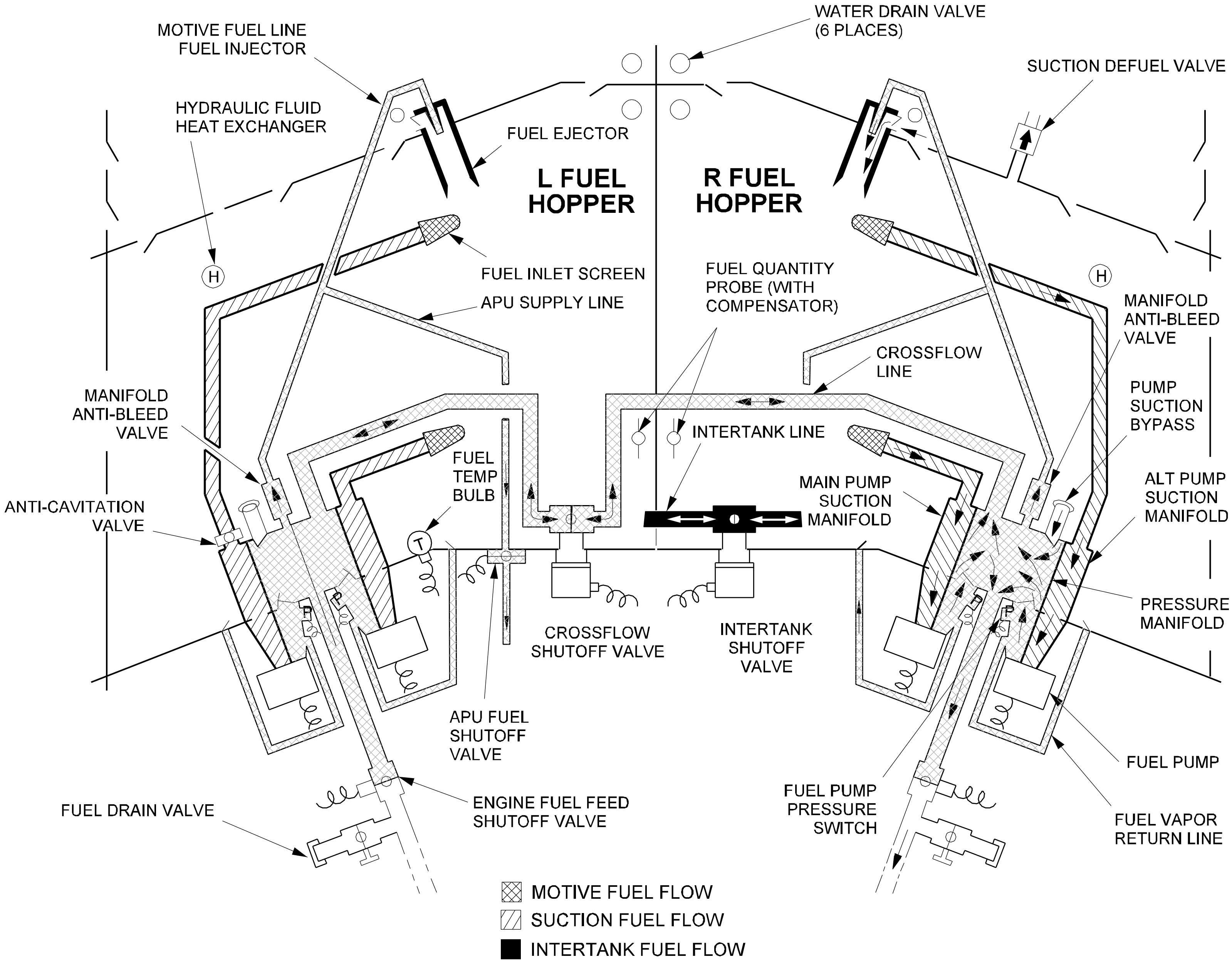


Fuel Hopper Simplified
Block Diagram
Figure 3

END

2A-28-30: Fuel Distribution System

1. **General Description:**

The fuel distribution system moves fuel from the wing fuel tanks to refill the hoppers and delivers fuel under pressure from the hoppers to the engines and APU.

Additionally, the fuel distribution system allows fueling and suction defueling through a single-point pressure fueling adapter. Defueling can also be accomplished using the boost pumps, drains or a combination of all three methods.

The fuel distribution system is composed of the following subsystems, units and components:

- Pressure Fueling System
- Defueling System
- Fuel Boost Pumps
- Fuel Ejectors
- Fuel Shutoff Valves

2. **Description of Subsystems, Units and Components:**

A. **Pressure Fueling System:**

The single-point pressure fueling system, shown in Figure 1 and Figure 4, is a hydromechanical system. It is capable of control from the cockpit (by solenoid valves) or from the adapter (by manual shutoff means). Components of the system include:

- Pressure fueling adapter
- Two precheck valves
- Two shutoff valves
- Two sensing valves
- Four high-level pilot valves
- Two fuel shutoff solenoid valves

(1) Connection and Commencement:

The fueling adapter is located underneath the right wing leading edge wing-to-fuselage fillet. After connecting the fuel nozzle to the fueling adapter, opening the nozzle supplies fuel at 35 to 55 psig (optimum for adequate prechecks) to the pressure fueling shutoff valve. Fuel pressure against the shutoff valve forces the poppet open, allowing fuel to flow to the tanks. Once fueling commences, system prechecks are completed.

(2) Precheck Selector To FLOAT:

With a tank's precheck selector in FLOAT, fuel flows through sense lines into the inboard and outboard high-level pilot valve float chambers. Flow is such that it flows into the chambers faster than it can drain through openings in the bottom of the chambers. As pilot valve fuel level rises, the float in each valve rises until it closes the

valve's poppet. The resulting back pressure in the inboard and outboard sense lines overcomes fuel pressure against the shutoff valve, which then closes to stop fueling. This sequence indicates a successful test.

(3) Precheck Selector To TANK PRESS:

Placing the precheck selector to TANK PRESS directs fuel to the tank pressure sense valve. Back pressure in the sense line then overcomes fuel pressure against the shutoff valve, which then closes to stop fueling. This sequence indicates a successful test.

(4) Precheck Selector To FUEL:

Placing the precheck selector to FUEL between each test sequence enables continued testing. After completion of the tests, placing the precheck selector to FUEL resumes normal fueling.

(5) Full Load Fueling:

As the fuel level rises, the inboard high-level pilot valves fills through the opening in the bottom of its chamber and closes before the outboard high-level pilot valve. Fueling continues while the outboard high-level pilot valve is still open. When the fuel level reaches full, the outboard high-level pilot valve closes and the resultant back pressure in the outboard sense line closes the shutoff valve. Fueling stops.

(6) Partial Load Fueling:

The preferred method for partial load fueling requires Essential DC bus power in order to use the L and / or R REMOTE FUELING SHUTOFF switches (cockpit overhead panel) to control fuel flow, and the standby fuel quantity indicator (copilot's flight panel) to monitor fuel quantity. When the desired amount of fuel has been uploaded into the wing tank(s), the REMOTE FUELING SHUTOFF switch(es) are selected to CLSD. This supplies 28 VDC to the respective fuel shutoff solenoid valve. The valve opens and the resultant back pressure against the pilot valve float system stops fuel flow to the tank(s).

NOTE:

Before fueling, it is recommended that the aircraft be positioned with the wings laterally level and with the nose gear at the low portion of the ramp.

NOTE:

The amber L-R FUEL LEVEL LOW caution Crew Alerting System (CAS) messages are prompted for display at approximately 650 lb (295 kg). These messages may erroneously appear at greater fuel loads when the boost pumps are not operating. Proper operation should resume when the boost pumps are selected on.

B. Defueling System:

The aircraft may be defueled by suction defueling or pumping. If necessary, after defueling, any remaining pockets of fuel may be removed by wing draining.

Suction defueling takes place through either the fueling adapter or main wheel well fuel drains. Suction defueling through the fueling adapter is basically the reverse of the fueling process: fuel is drawn from the fuel tanks through the fueling lines. Suction defueling through the main wheel well fuel drains requires connecting drain hoses to the drain valves. With the appropriate equipment, a fuel truck drains the tanks. Neither suction method will result in completely empty tanks. Approximately 150 gallons will remain, primarily in the left tank. This can be removed by the pumping method, if desired.

Defueling by pumping is a method whereby fuel from the wing tanks is forced through main wheel well fuel drains by the boost pumps. This method leaves much less fuel remaining than the suction methods, however, caution is required to ensure the boost pumps are not run dry for more than three minutes, to prevent pump damage.

Regardless of the defueling method, opening the INTER TANK valve (cockpit overhead panel) allows fuel to gravity flow from the left hopper to the right hopper as the right hopper level drops during defueling.

Once defueling is completed, the FUEL LOW LEVEL circuit breaker must be pulled to prevent damage to the system with power applied to the aircraft.

C. Fuel Boost Pumps:

(See Figure 3 and Figure 5.)

Two electric motor-driven boost pumps are mounted on the rear spar in each main wheel well. All four pumps are identical and interchangeable, but for the purposes of identification, the inboard pumps are designated as main pumps and the outboard pumps are designated as alternate pumps. The pumps are capable of operation for as long as fuel is available.

Each pump consists of a fuel-cooled 28 VDC motor that drives a low-pressure impeller. When operating, each pump supplies fuel at 22 psi (minimum) at a maximum flow rate of 8,000 PPH. The inlet and discharge ports of the pumps extend through the rear beam into the pump manifold. Spring-loaded flapper valves on the manifold's inlet and discharge ports close to allow pump removal without defueling the fuel tank. Operating pressure opens the discharge port flapper valve; installing the pump opens the inlet port flapper valve.

(1) Fuel Pump Manifold:

The fuel pump manifold, located in the rear of the fuel hopper, is composed of two suction chambers and a single pressure chamber. Each suction chamber connects to separate suction line with an inlet screen. An anti-cavitation float valve in each suction line prevents pump cavitation by supplying fuel through the valve if an inlet screen clogs. From the inlet screen, the pump draws fuel from the bottom of the hopper into the suction chamber, then through the suction port flapper valve, into the pump. Rotation of the pump impeller pressurizes the fuel and delivers it at low pressure through a

discharge port flapper valve into the pump manifold pressure chamber.

From the pressure chamber, fuel flows at low pressure to the following components:

(a) An engine fuel feed lines to its respective engine. Fuel flow through the line is controlled by a normally open shutoff valve installed between the boost pumps and engine-driven pumps.

(b) A crossflow line interconnected to the opposite side pump manifold. Fuel flow through the line is controlled by a normally closed shutoff valve located in the left fuel hopper.

(c) A motive flow line connected to the fuel ejector. Fuel for the APU is tapped from the left side motive flow line.

(d) Pump suction bypass lines which extend down to the bottom of the hoppers. These lines are fitted with a check valve normally held closed by boost pump pressure. If both boost pumps should fail, the engine can draw fuel directly from the hopper, using the suction bypass lines, bypassing the inoperative pumps.

(2) Pump Operation:

Depressing a MAIN PUMPS switch (cockpit overhead panel, FUEL SYSTEM section) supplies 28 VDC from the Essential DC bus to the associated pump's power relay. The relay then closes to supply 28 VDC operating power from the Essential DC bus to the pump's inverter. The inverter then supplies AC power to the pump motor. The amber OFF legend in the switch extinguishes.

Similarly, depressing an ALT PUMPS switch supplies 28 VDC from the Right Main DC bus to the associated pump's power relay. The relay then closes to supply 28 VDC operating power from the Right Main DC bus to the pump's inverter. The inverter then supplies AC power to the pump motor. The amber OFF legend in the switch extinguishes.

(3) Pump Malfunctions:

If a boost pump fails to develop 16 psi after switch selection, a pump pressure switch energizes a fuel boost pump relay. The relay then triggers the associated amber L-R MAIN FUEL FAIL or L-R ALT FUEL FAIL caution message on CAS.

If fuel pressure at the engine-driven high pressure fuel pump inlet is less than 15 psi, or both boost pumps on one side are selected OFF (or should fail) with the crossflow (X FLOW) valve closed, a pressure switch at the high pressure fuel pump inlet triggers the associated red L-R FUEL PRESS LOW warning message on CAS and on the ENGINE START system page. A respective red L / R FUEL PRESS annunciator also illuminates on the Standby Warning Lights Panel (SWLP), if installed. This message can occur regardless of engine speed and output pressure of the engine-driven low pressure fuel pump. In addition to serving as a warning to check and set the FUEL SYSTEM panel configuration, this message could serve as an early predictor of a fuel leak, in which case the

flight crew should be alert for erratic fuel flow indications.

(4) Autochange Circuit:

Should an operating boost pump on one side fail, an autochange circuit is incorporated to automatically select the other pump on that side. The autochange circuit is powered by the Essential DC bus and consists of two parts: control monitor relays and transfer relays.

The first part of the autochange circuit incorporates a control monitor relay for each boost pump. If the selected pump loses power or its power circuit breaker opens, the control monitor relay automatically selects the opposite boost pump on by energizing that pump's power relay. If a boost pump's power circuit breaker opens, an amber CB legend will illuminate in that pump's control switch.

The other part of the autochange circuit incorporates a transfer relay for each boost pump. If a boost pump is selected on but fails to develop 16 psi output, the transfer relay automatically selects the opposite boost pump on by energizing that pump's power relay. The amber OFF legend will illuminate in the failed pump's switch.

Whenever the autochange system activates a boost pump other than the one selected, the switch corresponding to the now-operative boost pump must be selected on by the flight crew. The switch corresponding to the now-inoperative boost pump should be selected to OFF.

Restated in summary, the autochange circuit will automatically select the opposite boost pump on the same side if:

- A main boost pump fails, even with the alternate boost pump not selected on
- A main boost pump power circuit breaker opens with the pump running due to actuation of the main pump control monitor relay
- An alternate boost pump fails, even with the main boost pump not selected on
- An alternate boost pump power circuit breaker opens with the pump running due to actuation of the alternate pump control monitor relay

D. Fuel Ejectors:

A fuel ejector (often referred to as an ejector pump or jet pump and shown in Figure 6) is a simple device that has no moving parts. It resembles and operates like a venturi tube. It uses high pressure / low volume fluid as an operating force to move a large volume of low pressure fluid in the desired direction.

Each fuel hopper has a fuel ejector on the forward face of the forward wall with the outlet projecting into the hopper. With a boost pump operating, high pressure / low volume fuel flowing through the motive flow fuel line from the pump manifold enters the ejector's motive flow inlet.

As the motive flow fuel rushes out of the nozzle and enters the throat of the venturi, it creates a low pressure area that in turn draws a large volume of fuel from the wing tank and moves it at low pressure into the fuel hopper through the ejector's outlet. This induced fuel flow ensures the hopper

remains full at all times as the engines consume fuel. Any excess fuel in the hopper overflows through the top of the hopper wall back into the wing tank.

E. Fuel Shutoff Valves:

Shutoff valves are incorporated in the fuel distribution system to control movement of fuel about the wing tanks, hoppers and to the engines. All shutoff valves receive power from the Essential DC bus. The valves are shown in Figure 3 and described as follows:

(1) Engine Fuel Feed Shutoff Valve:

An engine fuel feed shutoff valve is installed in each engine's fuel feed line. Each valve is normally open to allow passage of low pressure fuel to the respective engine-driven low pressure fuel pump. The valve is closed by pulling the respective FIRE handle, located on the left and right forward sides of the cockpit center pedestal.

(2) Crossflow Valve:

The fuel crossflow line is pressurized with fuel from the fuel pump manifold pressure chamber. The fuel then waits in the crossflow line at the normally-closed crossflow valve. Selection of the X FLOW switch (cockpit overhead panel, Figure 8) opens the valve, allowing fuel from the fuel pump manifold pressure chamber on one side to flow to the chamber on the other side, enabling both engines to be operated from one tank system. With the X FLOW switch selected open, a horizontal bar in the switch capsule illuminates. In addition, a blue FUEL XFLOW OPEN advisory message is displayed on CAS. When selected closed, the bar extinguishes and the CAS message is removed.

The crossflow valve is also opened for fuel balancing in accordance with the limitations set forth in Section 1 of the GIV Airplane Flight Manual.

(3) Intertank Valve:

The intertank valve is opened to combine the left and right hoppers. Located in the right fuel hopper, the intertank valve is controlled by the INTER TANK switch (cockpit overhead panel, Figure 8). When selected open, a horizontal bar in the switch capsule illuminates. In addition, a blue FUEL INT TNK OPEN advisory message is displayed on CAS. When selected closed, the bar extinguishes and the CAS message is removed.

F. Fuel Balancing Procedure:

If fuel balancing is required (either one or both engines operating), proceed as follows:

(1) When fuel tank temperature is above 0° C or fuel contains an anti-icing additive:

(a) Select both boost pumps ON for the side having the higher fuel state.

(b) Select the crossflow valve OPEN.

(c) Select both boost pumps OFF for the side having the lower fuel state.

(2) When fuel tank temperature is below 0° C or fuel contains no anti-icing additive:

(a) Select all four boost pumps ON.

(b) Select the intertank valve OPEN.

(c) Establish a small sideslip (½ ball) in the appropriate direction.

(3) After fuel is balanced:

(a) Ensure all four boost pumps are ON.

(b) Select the crossflow valve CLOSED.

(c) Select the intertank valve CLOSED.

(d) Return to normal cruise operations.

CAUTION

THE PRIMARY PURPOSE OF THE INTERTANK VALVE IS FOR SUCTION DEFUELING. THE INTERTANK VALVE IS NOT TO BE USED IN FLIGHT EXCEPT IN ACCORDANCE WITH THE ABNORMAL / EMERGENCY PROCEDURES CONTAINED IN THE LATEST APPROVED REVISION OF THE GIV AIRPLANE FLIGHT MANUAL.

G. Fuel System Check:

Test Conditions:

(1) Ensure the L and R REMOTE FUELING SHUTOFF switches are OPEN (extended). Verify the CLSD legend is extinguished.

(2) Ensure the crossflow valve is CLOSED.

(3) Ensure the L MAIN boost pump is ON.

(4) Ensure the intertank valve is CLOSED.

(5) Ensure the L FUEL PRESS LOW warning message is not displayed on CAS.

(6) Ensure the R FUEL PRESS LOW warning message is displayed on CAS.

Perform the following steps:

(7) Select the L and R REMOTE FUELING SHUTOFF switches CLOSED (depressed). Verify the CLSD legend is illuminated.

(8) Select the L and R REMOTE FUELING SHUTOFF switches OPEN.

(9) Simultaneously select the crossflow and intertank valves OPEN.

(10) Verify the following:

(a) Horizontal bar in crossflow valve switch capsule illuminates.

(b) Horizontal bar in intertank valve switch capsule illuminates.

(c) FUEL XFLOW OPEN advisory message is displayed on CAS.

(d) FUEL INT TNK OPEN advisory message is displayed on CAS.

(11) Verify the R FUEL PRESS LOW warning message is not displayed on CAS.

(12) Select the L MAIN boost pump OFF.

(a) Select the L ALT boost pump ON.

(b) Verify the L FUEL PRESS LOW and R FUEL PRESS LOW warning messages are momentarily displayed on CAS, then are removed.

(13) Select the L ALT boost pump OFF.

(a) Select the R MAIN boost pump ON.

(b) Verify the L FUEL PRESS LOW and R FUEL PRESS LOW warning messages are momentarily displayed on CAS, then are removed.

(14) Select the R MAIN boost pump OFF.

(a) Select the R ALT boost pump ON.

(b) Verify the L FUEL PRESS LOW and R FUEL PRESS LOW warning messages are momentarily displayed on CAS, then are removed.

(15) Simultaneously select the crossflow and intertank valves CLOSED.

(16) Verify the following occurs simultaneously:

(a) Horizontal bar in crossflow valve switch capsule extinguishes.

(b) Horizontal bar in intertank valve switch capsule extinguishes.

(c) FUEL XFLOW OPEN advisory message is removed from CAS.

(d) FUEL INT TNK OPEN advisory message is removed from CAS.

(17) Verify the L FUEL PRESS LOW warning messages is displayed on CAS.

(18) Select the L MAIN boost pump ON.

(19) Verify the L FUEL PRESS LOW is removed from CAS.

(20) Select the R ALT boost pump OFF.

(21) Verify the R FUEL PRESS LOW is displayed on CAS.

3. Controls and Indications:

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
ALT PUMP CONT	PO	D-4	Right Main DC Bus
FUEL INTERTANK V	PO	B-4	Essential DC Bus
FUEL PUMP IND	PO	A-4	Essential DC Bus
FUEL PUMP LOGIC (1)	PO	C-4	Essential DC Bus
FUEL X FLOW V	PO	D-2	Essential DC Bus
L ALTN BOOST PUMPS	PDB	RIGHT DC	Right Main DC Bus
L FUEL S/O	PO	A-2	Essential DC Bus
L FUELING S/O	PO	A-3	Essential DC Bus

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
L MAIN BOOST PUMPS	PDB	ESS DC	Essential DC Bus
MAIN PUMP CONT	PO	A-1	Essential DC Bus
R ALTN BOOST PUMPS	PDB	RIGHT DC	Right Main DC Bus
R FUEL S/O	PO	B-2	Essential DC Bus
R FUELING S/O	PO	B-3	Essential DC Bus
R MAIN BOOST PUMPS	PDB	ESS DC	Essential DC Bus

NOTE(S):

(1) Aircraft having SPZ-8000.

B. Warning (Red) Messages and Annunciations:

CAS Message:	SWLP Indication	Cause or Meaning:
L-R FUEL PRESS LOW	L FUEL PRESS R FUEL PRESS	Fuel pressure at inlet to high pressure fuel pump is less than 15 psi, or both fuel BOOST PUMPS on one side have been selected to OFF with CROSSFLOW valve CLOSED.

C. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
L-R ALT FUEL FAIL	Indicated alternate fuel boost pump has failed.
L-R MAIN FUEL FAIL	Indicated main fuel boost pump has failed.

D. Advisory (Blue) Messages and Annunciations:

CAS Message:	Cause or Meaning:
FUEL INT TANK OPEN	Fuel intertank valve is open.
FUEL XFLOW OPEN	Fuel crossflow valve is open.

4. Limitations:

A. Flight Manual Limitations:

All operable boost pumps shall be selected ON for all phases of flight unless fuel balancing is in progress.

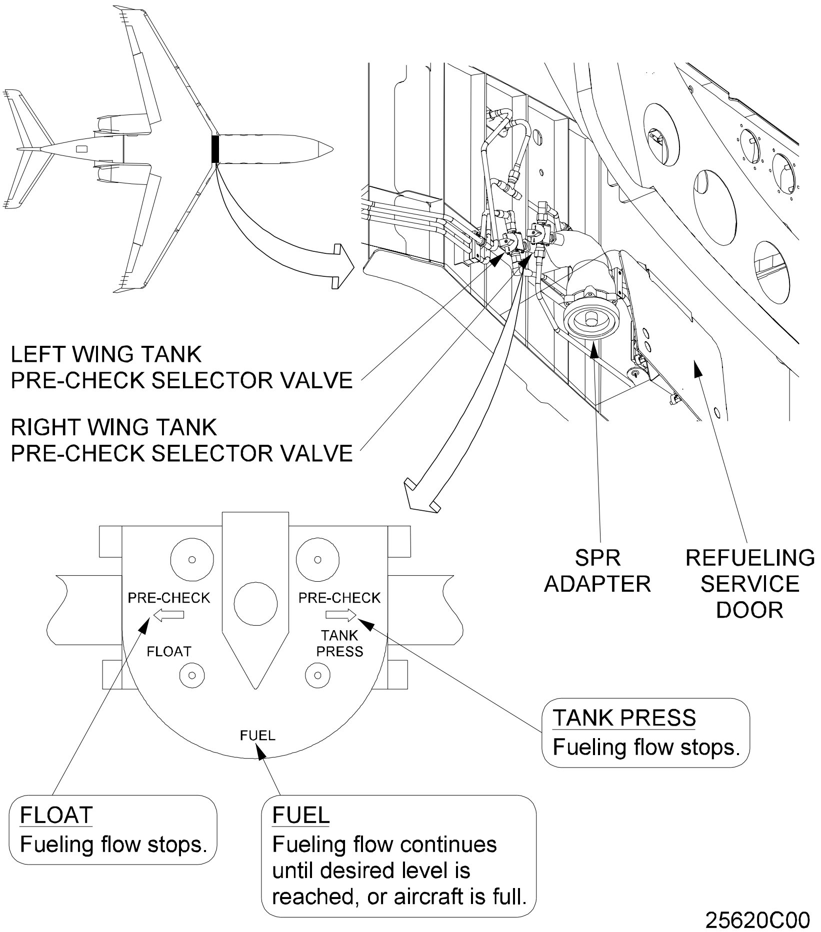
See Section 01-12-30, Engine Fuel Grades.

CAUTION

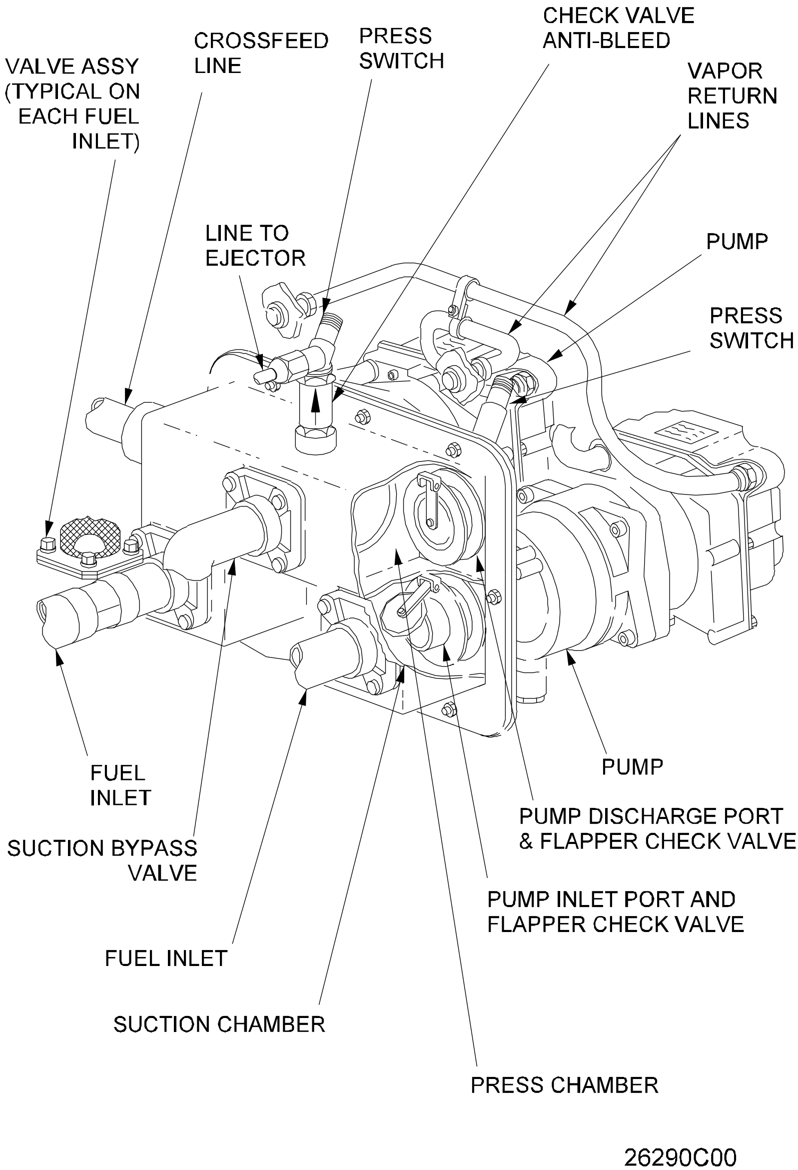
THE ENGINE WILL RUN ON SUCTION FUEL FEED ONLY AT OR BELOW 20,000 FEET. ABOVE 20,000 FEET, THE ENGINE WILL RUN ERRATICALLY AND FLAME OUT IF THE CROSSFLOW IS NOT OPEN WITH AT LEAST ONE BOOST PUMP ON.

B. Other Limitations:

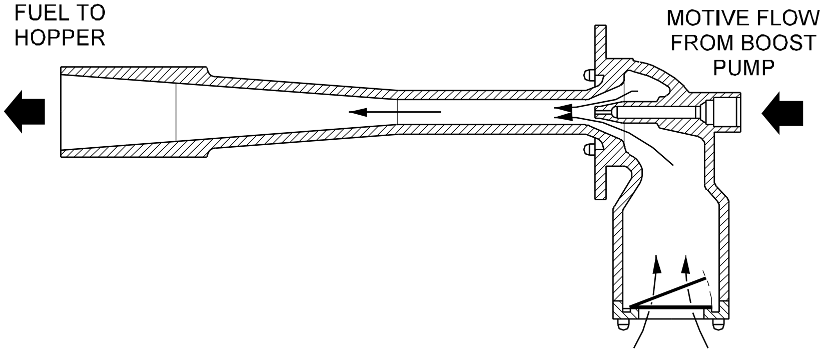
Normal operation of the fuel system is from tank to respective engine. For takeoff and landing, all operable boost pumps are selected ON.



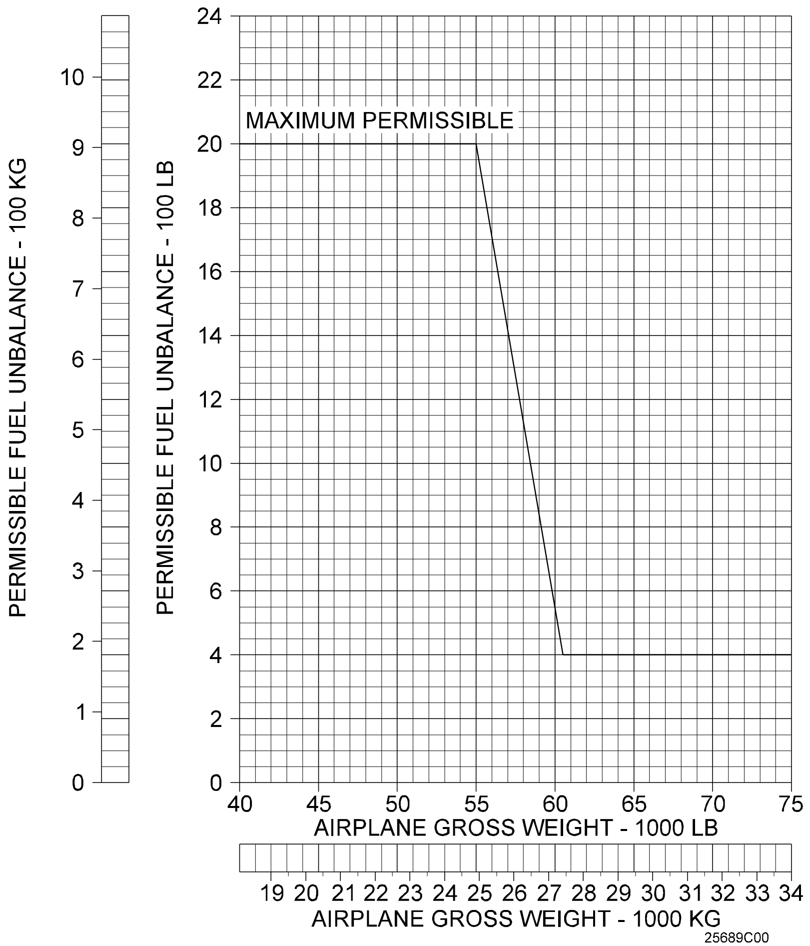
Refueling Adapter and Precheck Controls
Figure 4



Fuel Boost Pump
Figure 5



Fuel Ejector
Figure 6



Permissible Fuel Unbalance For All Flight Operations
Figure 7

END

2A-28-40: Fuel Indication System

1. General Description:

The fuel indication system provides the flight crew with continuous indication of fuel quantity and warns of a low fuel level. Continuous indication of fuel tank temperature is also provided. The fuel indication system is composed of the following subsystems, units and components:

- Fuel Quantity Indication System

- Fuel Low Level Warning System
- Fuel Temperature Indication System

Fuel flow and engine fuel temperature indications are described in Section 2A-73-10, Engine Fuel System.

2. Description of Subsystems, Units and Components:

A. Fuel Quantity Indication System:

(See Figure 8 through Figure 10.)

(1) General:

A set of 21 parallel-wired, capacitence-type fuel probes extend into various parts of each wing tank to provide an accurate measurement of tank fuel quantity. The two sets of probes are identical and independent. Each set connects to a signal conditioner through a signal junction box. The signal conditioner, powered by the Essential DC bus, uses the composite readings to drive the standby fuel quantity indicator and to provide driving signals to Data Acquisition Units (DAUs) No. 1 (left side) and No. 2 (right side) for display on the Engine Instrument and Crew Alerting System (EICAS).

(2) EICAS Fuel Displays:

Fuel quantity appears in the lower right corner of the EICAS engine instruments display as FUEL QTY. The Left (L) and Right (R) displays show fuel quantity in the respective wing in 50 lb increments, ranging from 0 lb to 15,000 lb. The Total (T) display shows the combined fuel quantity of both wings in 50 lb increments, ranging from 0 lb to 30,000 lb. The FUEL system page on CAS displays the identical information. Under normal conditions, EICAS fuel quantity data is displayed white in color. If a low fuel condition in one wing is detected, the respective quantity display digits change to amber. If a low fuel condition in both wings is detected, the entire FUEL QTY display changes to amber. If a DAU fails, the associated side's display digits are replaced with amber dashes.

(3) Standby Fuel Quantity Indicator:

Located on the copilot's flight panel, the standby fuel quantity indicator contains three 4 digit displays: LEFT, RIGHT and TOTAL. Values shown on each display are in pounds to be multiplied by ten (LBS × 10). Above the LEFT and RIGHT displays are associated amber LOW FUEL warning lights which illuminate when approximately 650 pounds (295 kg) of usable fuel remains in the respective hopper.

To the right of the TOTAL display is a three-position, momentary-action TEST switch, used to test both the standby fuel quantity indication system and / or the EICAS fuel quantity indication system. Switch positions are "D" (for standby fuel quantity indicator display testing only), off (the spring-loaded center position) and "T" (for testing the entire fuel quantity indication system). Holding the TEST switch in "D" results in the standby fuel quantity indicator's three displays showing all eights (8888); EICAS is not tested in this mode. Holding the TEST switch in "T" results in the standby fuel quantity indicator and EICAS FUEL QTY display showing 7000 pounds in

each wing and 14,000 pounds total. (If the No. 21 probe compensator is not covered, the EICAS FUEL QTY display will show 9000 pounds in each wing and 18,000 pounds total.) The EICAS FUEL QTY display will also be amber. During the test, the amber LOW FUEL warning lights on the standby fuel quantity indicator will flash for approximately 30 seconds and then illuminate steady. In addition, the amber L-R FUEL LEVEL LOW caution messages will be displayed on CAS. With the test completed, releasing the TEST switch returns the displays to indicating actual fuel quantity and extinguishes the amber LOW FUEL warning lights.

A switchlight, labeled FUEL QTY IND, is installed immediately to the right of standby fuel quantity indicator. When depressed, standby fuel quantity indicator display is inhibited. When extended, normal display is available. The FUEL QTY IND switchlight legend is blue in color and is always illuminated.

Power for the standby fuel quantity indication system is provided by the Essential DC bus. It remains functional all the way down to, and including, Emergency Battery (E-BATT) operation.

B. Fuel Low Level Warning System:

The fuel low level warning system (Figure 11) consists of a temperature-sensitive thermistor in each hopper, a dual-circuit control unit and two amber LOW FUEL lights on the standby fuel quantity indicator. The system receives power from the Essential DC bus.

When the hoppers are full, fuel covers the thermistor and the control units maintains its two warning relays in the energized position. No warnings are displayed in this state.

If a hopper’s fuel level drops below approximately 650 pounds (295 kg), it exposes the thermistor to air. Because there is always a difference between fuel temperature and air (or fuel / air vapor) temperature, resistance changes in the thermistor. The control units senses this change in resistance and de-energizes the respective warning relay. When the relay is de-energized, it provides a signal to the fuel quantity signal conditioner.

The signal conditioner, in turn, flashes the respective amber LOW FUEL light on the standby fuel quantity indicator for 30 seconds to alert the flight crew, then reverts to steady illumination. The signal conditioner also provides a signal to EICAS to display the appropriate cautions. An amber L-R FUEL LEVEL LOW message is displayed on CAS; the respective FUEL QTY display changes to amber. If a low fuel condition in both wings is detected, the entire FUEL QTY display changes to amber.

C. Fuel Temperature Indication System:

A resistance-type temperature bulb in the left fuel hopper provides a signal to DAU No. 1. DAU No. 1 then sends the signal to EICAS for display on the FUEL system page as fuel tank temperature (FUEL TANK TEMP). Fuel tank temperature is displayed in 1° C resolution within a normal operating range of -40° C to 54° C.

Depending on fuel temperature, the EICAS displays the temperature reading with either white or red digits, as listed in the **Limitations** portion of this section.

3. Controls and Indications:

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
FUEL LOW LEVEL	PO	B-1	Emergency Battery Bus
L FUEL QTY	PO	C-1	Essential DC Bus
R FUEL QTY	PO	D-1	Essential DC Bus

B. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
L-R FUEL LEVEL LOW	Fuel level in hopper is below approximately 650 lb (295 kg).

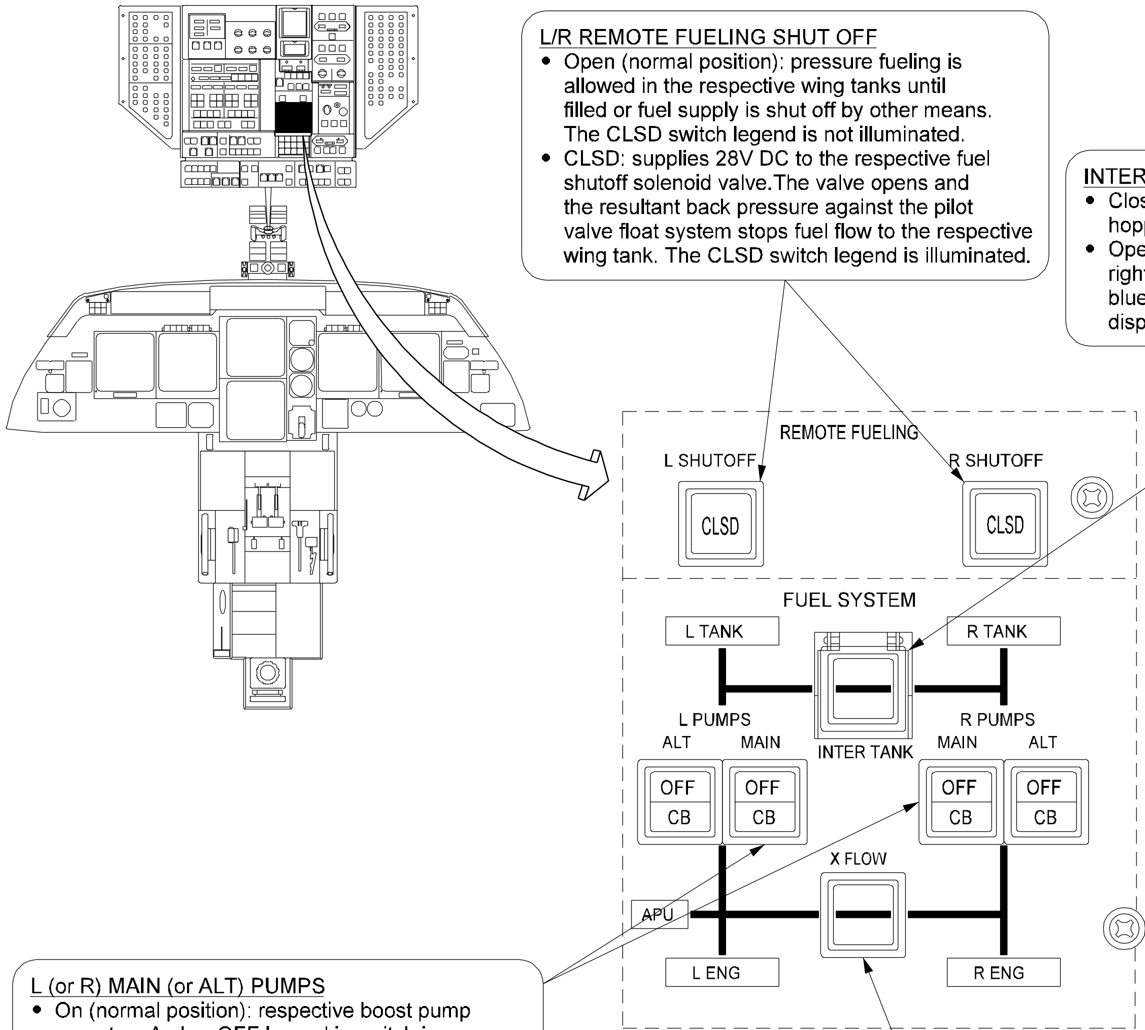
Annunciation:	Cause or Meaning:
LOW FUEL light(s) (amber) illuminated on standby fuel quantity indicator.	Fuel level in respective hopper is below approximately 650 lb (295 kg).

4. Limitations:

A. Flight Manual Limitations:

Fuel Tank Temperature (FUEL TANK TEMP):

- 54° C and above: red digits
- -40° C to 54° C and above: white digits
- Less than -40° C: red digits



L/R REMOTE FUELING SHUT OFF
- Open (normal position): pressure fueling is allowed in the respective wing tanks until filled or fuel supply is shut off by other means. The CLSD switch legend is not illuminated.
- CLSD: supplies 28V DC to the respective fuel shutoff solenoid valve. The valve opens and the resultant back pressure against the pilot valve float system stops fuel flow to the respective wing tank. The CLSD switch legend is illuminated.

INTER TANK
- Closed (normal position): intertank valve (in right hopper) is closed. Bar in switch is extinguished.
- Open: intertank valve opens, combining left and right hoppers. Bar in switch is illuminated and a blue FUEL INT TNK OPEN advisory message is displayed on CAS.

L (or R) MAIN (or ALT) PUMPS
- On (normal position): respective boost pump operates. Amber OFF legend in switch is extinguished.
- OFF: respective boost pump is shut off. Amber OFF legend in switch is illuminated.
- CB: respective boost pump's power circuit breaker is open. Amber CB legend in switch is illuminated.

X FLOW
- Closed (normal position): crossflow valve (in left hopper) is closed. Bar in switch is extinguished.
- Open: crossflow valve opens, allowing fuel from fuel pump manifold pressure chamber on one side to flow to the chamber on the other side, enabling both engines to be operated from one tank system. Bar in switch is illuminated and a blue FUEL X FLOW OPEN advisory message is displayed on CAS.

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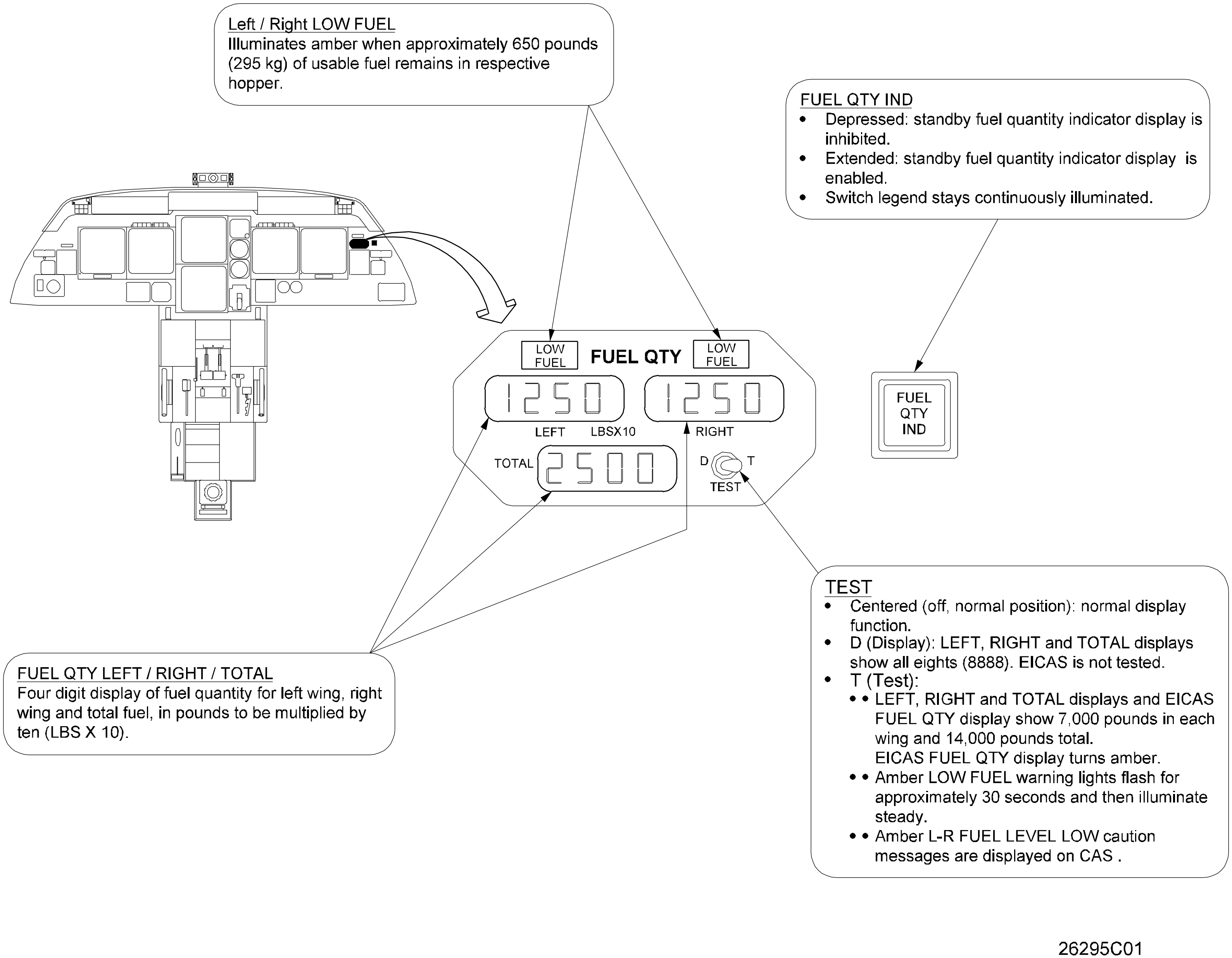
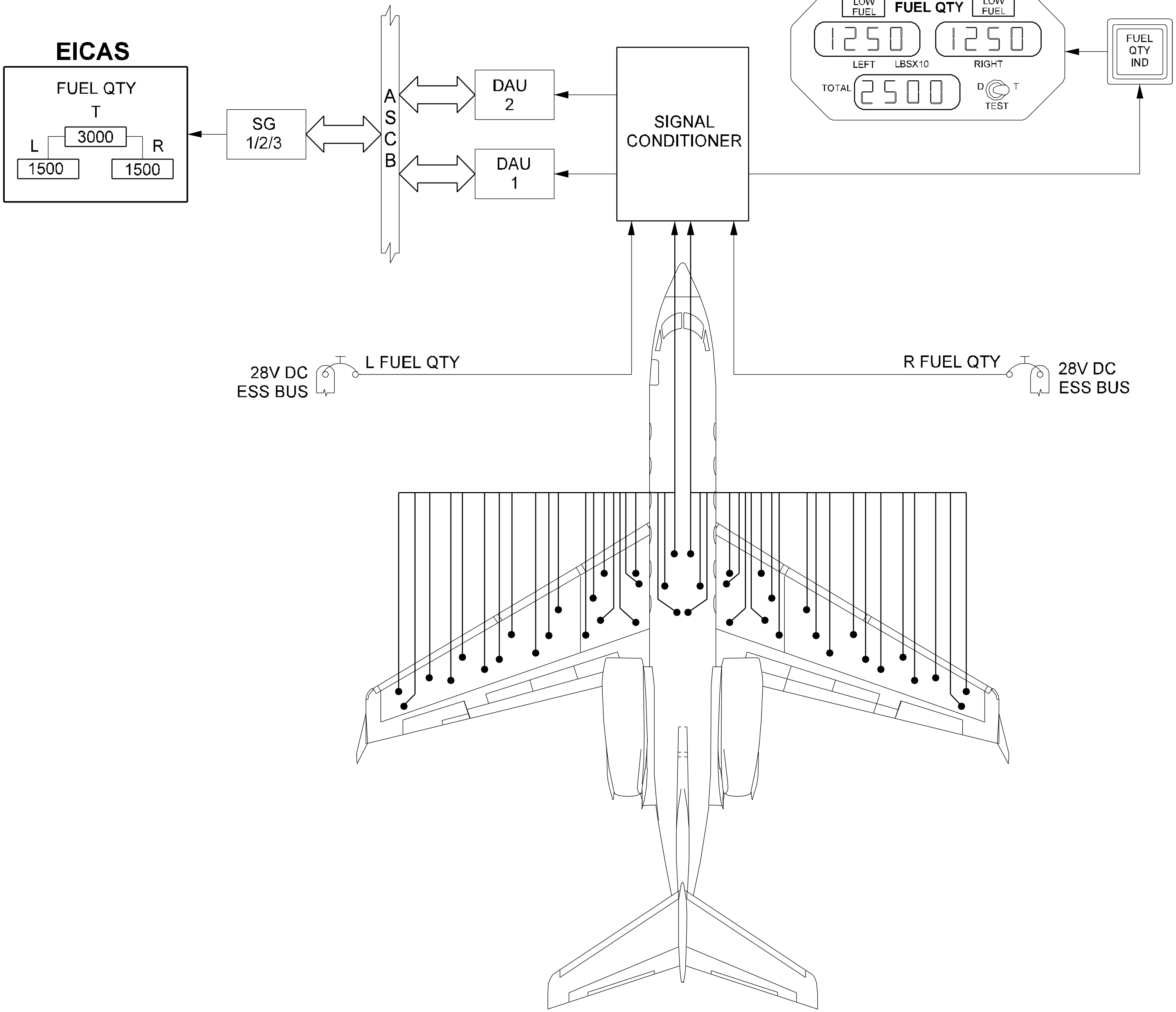
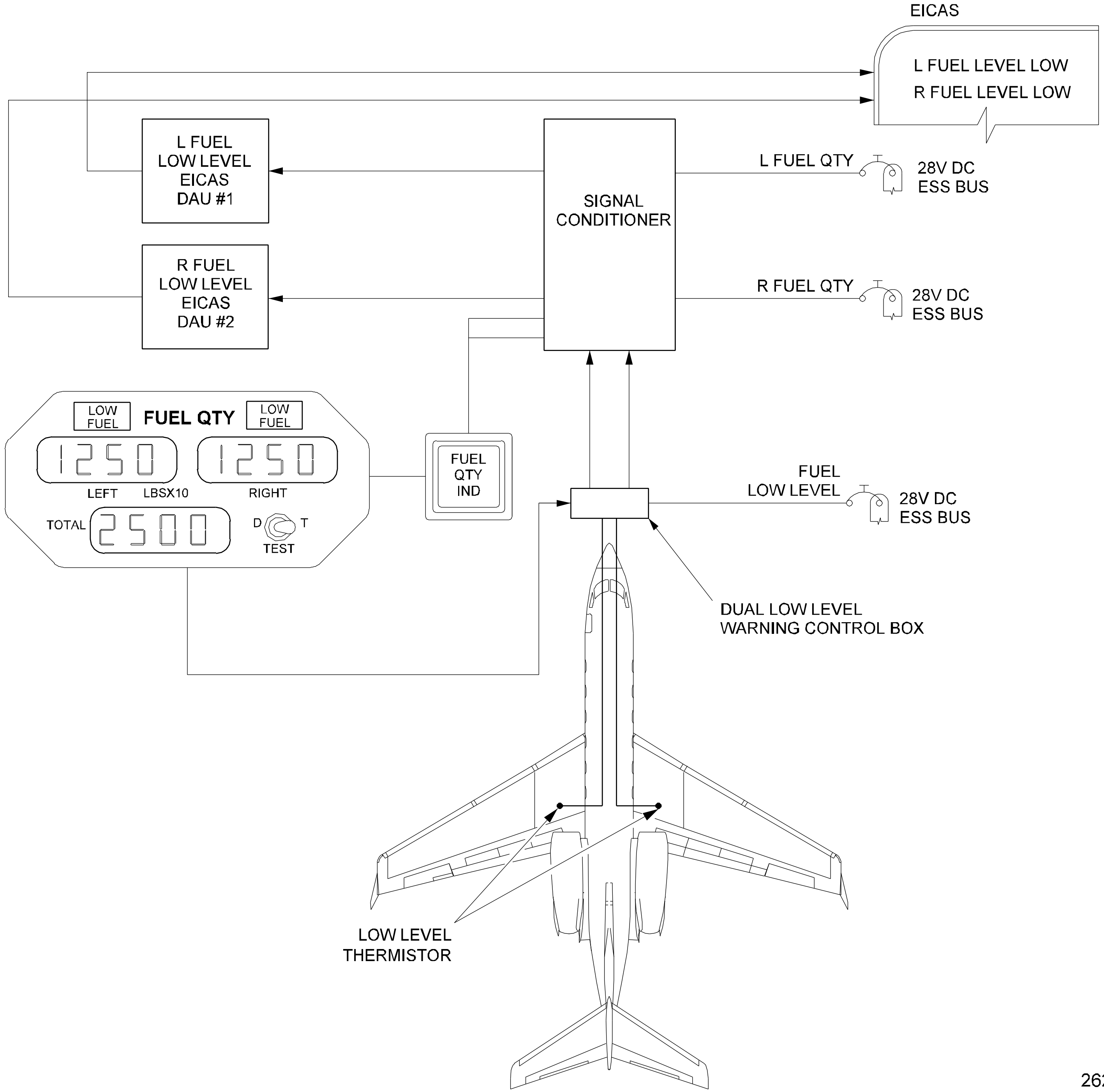
Fuel System Controls and Indications
Figure 8



Standby Fuel Quantity System Controls and Indications
Figure 9



Fuel Quantity System
Simplified Block Diagram
Figure 10



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Fuel Low Level System
Simplified Block Diagram
Figure 11

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2A-29-10: General

The hydraulic system for the Gulfstream IV (Figure 1) provides a means to store hydraulic fluid and deliver hydraulic fluid under pressure to using systems. Two engine-driven main hydraulic systems are installed in the aircraft. They are the Combined system (left engine) and the Flight system (right engine). System design is such that a single hydraulic system failure will not result in loss of the entire hydraulic system.

In addition to the main hydraulic systems, two auxiliary hydraulic systems are incorporated to provide operational redundancy and to perform various utility functions, in some cases without the need for operating engines or external power sources. The two auxiliary hydraulic systems are the hydraulically-driven Utility system and the electrically-driven Auxiliary system.

Each primary flight control actuator (aileron, elevator and rudder) and each spoiler actuator are termed “dual-tandem actuators”, in that both Combined system and Flight system power is used through independent load paths, yet summed for output. Thus, if a loss of either hydraulic system input to actuator did occur, the remaining hydraulic system continues to allow actuator function.

The system is designed for use with Type IV phosphate ester-based hydraulic fluid (Hy-Jet IV, Hy-Jet IV-A, Skydrol LD-4 and Skydrol 500B-4).

The hydraulic system is divided into 2 subsystems:

- 2A-29-20, Main Hydraulic Systems
- 2A-29-30, Auxiliary Hydraulic Systems

A master table of component availability by hydraulic system is shown on the following page.

Table 1. Master Table of Component Availability

SYSTEM/COMPONENT	HYDRAULIC SYSTEM			
	COMB	FLT	UTIL	AUX
Elevators	X	X		
Stall Barrier	X		X	
Ailerons	X	X		
Speed Brakes	X	X		
Flight Spoilers	X	X		
Ground Spoilers	X	X [(1)]		
Rudder	X	X		
Yaw Damper		X		
Left Thrust Reverser	X			
Right Thrust Reverser		X		
Utility Pump Motor		X		
Wing Flaps	X		X	X
Landing Gear and Doors	X		X	X [(2)]
Nosewheel Steering	X		X	
Brakes	X		X	X
STBY ELEC PWR SYS Motor	X			
Parking Brake Pressure				X
Main Entrance Door				X
Cargo Door [(3)]				X

(1) If servo pressure signal from COMB, UTIL or AUX is present.
(2) Ground use only, through ground service valve.
(3) Aircraft with Aircraft Service Change (ASC) 213 or 354 incorporated.

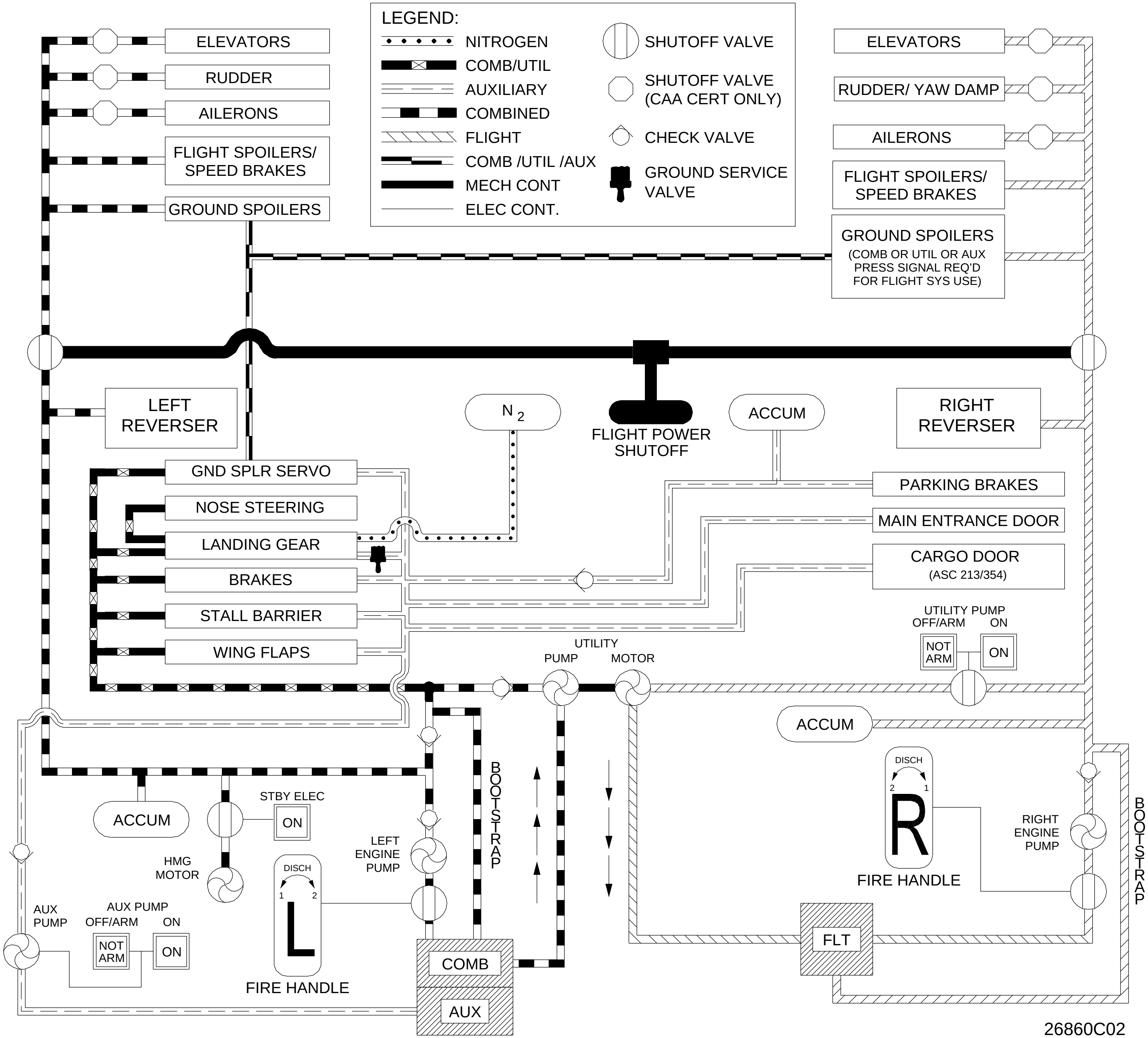

Figure 1. Hydraulic Schematic Block Diagram

2A-29-20: Main Hydraulic Systems

1. General Description:

The main hydraulic systems provide the primary means to store hydraulic fluid and deliver hydraulic fluid under pressure to the using systems. Each of the main hydraulic systems (Combined system and Flight system) is supplied pressurized fluid by a dedicated engine-driven pump. The left engine-driven pump provides pressurized fluid at 3000 ±50 psi (static) to the Combined system and its using components. The right engine-driven pump provides pressurized fluid at 3000 ±50 psi (static) to the Flight system and its using components. The systems' fluids do not intermix. Each reservoir is pressurized by its own hydraulic system at a 50:1 ratio. With hydraulic system pressure at 3000 psi, the respective reservoir pressurizes to 60 psi.

A. Combined Hydraulic System:

The Combined system (Figure 2) performs the majority of hydraulic functions for the aircraft, thus it has the largest fluid quantity of the two systems. Its pump is located on the left engine, and its reservoir and manifold are located on the left side of the aft equipment (tail) compartment.

Total system capacity is approximately 15.5 gallons (58.6 liters). This includes the reservoir, lines and all system components. Pressurized fluid is provided to the following systems, units and components:

- Ailerons
- Speed Brakes
- Flight Spoilers
- Ground Spoilers
- Rudder
- Elevator
- Stall Barrier
- Landing Gear and Gear Doors
- Left Thrust Reverser
- Flaps
- Wheel Brakes
- Nosewheel Steering
- Standby Electrical Power System Motor

B. Flight Hydraulic System:

Functionally and structurally similar, the Flight system (Figure 3) compliments the Combined system by sharing the workload, yet remaining independent and isolated for specific functions. Its pump is located on the right engine, and its reservoir and manifold are located on right side of the tail compartment.

Total system capacity is approximately 4.5 gallons (17 liters). This includes the reservoir, lines and all system components. Pressurized fluid is provided to the following systems, units and components:

- Ailerons
- Speed Brakes

- Flight Spoilers
- Ground Spoilers
- Rudder
- Elevator
- Yaw Damper
- Right Thrust Reverser
- Utility Pump Motor

C. Fluid Replenishing System:

Hydraulic system fluid replenishing can be accomplished by using either the fill connection on the ground test panel (underside of the aircraft, just forward of the tail compartment door) or an onboard replenishing system (inside the tail compartment).

2. Description of Subsystems, Units and Components:

A. Combined System and Flight System Reservoirs:

(See Figure 4 and Figure 5.)

(1) Combined System Reservoir Particulars:

The Combined system reservoir contains two chambers separated by a bulkhead; one chamber supplies the Combined and Utility systems, and the other supplies the Auxiliary system. Total capacity of the reservoir is 5.5 gallons (20.8 liters) with a Combined system working volume of 3.75 gallons (14.2 liters) and an Auxiliary system working volume of 1.75 gallons (6.6 liters). Both chambers are full under normal operating conditions. An opening at the top of the reservoir bulkhead allows fluid flow from the Combined chamber to the Auxiliary chamber. Filling of both chambers can be accomplished using either the fill connection on the ground test panel (underside of the aircraft, just forward of the tail compartment door) or a replenishing system (inside the tail compartment).

Four suction ports are included on the reservoir: engine pump, utility pump, auxiliary pump, and ground test rig. Four return ports are also incorporated: auxiliary, brake, auxiliary brake, and combined system. Other ports include one each for the bleed line, pressure relief valve, engine pump bypass, and utility pump bypass.

(2) Flight System Reservoir Particulars:

The Flight system reservoir differs from the Combined system reservoir in that it is a smaller, single chamber reservoir. Total capacity of the reservoir is 1.25 gallons (4.7 liters) with working volume of 0.86 gallons (3.25 liters). Filling is accomplished using the same methods as the Combined system reservoir.

Five ports are included on the Flight system reservoir: suction, return, drain, bleed and relief valve.

(3) Reservoir Pressure Relief Valves:

A pressure relief valve is installed on each reservoir to prevent overpressurization. If reservoir pressure exceeds 100 psi, the valve opens to vent excess pressure overboard through a vent line. The valve closes at 75 psi.

(4) Reservoir Thermal Switches:

A thermal switch is installed on each reservoir to monitor system fluid temperature. If fluid temperature reaches 220 ±5°F (104 ±3°C), the thermal switch closes causing a blue CMB HYD HOT or FLT HYD HOT advisory message to be displayed on the Crew Alerting System (CAS). As fluid temperature drops to 175 ±10°F (79 ±6°C), the thermal switch opens, clearing the CAS message.

NOTE:

Automatic operation of the utility pump is not possible with a FLT HYD HOT advisory message displayed.

(5) Reservoir Manual Bleeder Valves:

A manual bleeder valve installed on each reservoir provides a means to relieve trapped air pressure acquired during servicing. Pressing and holding the BLEED button on the valve opens the valve, venting excess air pressure overboard through a vent line.

(6) Reservoir Fluid Quantity Transmitters:

A fluid quantity transmitter installed on each reservoir measures fluid quantity by using the reservoir piston's movement to actuate a spring-loaded tape reel mechanism. When supplied with 28V DC from the Essential DC bus, the quantity transmitter supplies an output voltage proportional to fluid quantity to the Data Acquisition Units (DAUs) for display on the HYDRAULICS system page.

(7) Reservoir Sight Gauges:

A visual sight gauge on the aft end of each reservoir piston provides an indication of reservoir fluid quantity by denoting the FULL and REFILL fluid levels. No electrical power is necessary to use the visual sight gauge, but the system must pressurized to 3000 psi to obtain an accurate reading. The proper fluid level for the reservoir is the FULL indication on the sight gauge.

(8) Reservoir Vacuum Relief Check Valve (Combined System Only):

The Combined system's reservoir vacuum relief valve opens at a set negative pressure to allow ambient air into the reservoir. This prevents auxiliary hydraulic pump cavitation.

(9) Reservoir Piston Scuppers:

A piston scupper is installed on each reservoir to catch any fluid leakage from the reservoir piston end. This leakage is then vented overboard.

B. System Shutoff Valves:

System shutoff valves (Figure 6) are installed in the tail compartment to control fluid flow from each reservoir to its respective engine-driven hydraulic pump. They are normally open, electrically operated, motor-driven, gate-type valves.

Pulling the associated FIRE handle supplies 28V DC from the Essential DC bus to the valve motor. The motor then drives the valve closed. Pushing in the associated FIRE handle opens the valve.

Valve position signals are supplied to the Fault Warning Computer (FWC) for use on the Engine Instruments display, ENGINE START system page and HYDRAULICS system page. When the valve is in transit, amber dashes are shown in the associated system pressure display window. When the valve is closed, amber crosshatching is shown in the window.

Valve position is also shown on the valve itself by a red pointer flag/lever pointing to either OPEN or CLOSED. The flag/lever may be manually moved to the desired position. However, if the valve is moved to the CLOSED position, the associated HYD S/O circuit breaker must also be pulled to prevent the valve from being driven to its normally open position when power is applied to the aircraft.

C. Manually Operated Shutoff Valves:

A manually operated shutoff valve controls hydraulic fluid flow from the Combined and Flight hydraulic systems to the flight control actuators. Pulling the FLIGHT POWER SHUT OFF handle on the cockpit center pedestal (Figure 7) closes the shutoff valve through the use of a controlex cable. For detailed information on this system, see Section 2A-27-00: Flight Controls.

D. Engine Driven Hydraulic Pumps:

A positive displacement, constant delivery hydraulic pump is installed on each engine's right-hand high pressure gearbox (Combined on left engine, Flight on right engine). The pumps draw fluid from their respective reservoir and deliver the fluid under pressure to the using systems.

A small quantity of hydraulic fluid lubricates and cools the pump. This fluid is then returned to the reservoir through the bypass port.

During normal operations, fluid flows through the heat exchanger and engine pump bypass to the reservoir. At low ambient temperatures, when fluid is viscous, a relief valve opens at 100 psi to route engine pump bypass flow through the main return filter to the reservoir.

E. Hydraulic Heat Exchangers:

Combined and Flight hydraulic system fluid is circulated though dedicated hydraulic heat exchangers installed in the wing fuel hoppers (Combined in left hopper, Flight in right hopper). Functioning as a radiator, each heat exchanger allows the warmer hydraulic fluid to dissipate heat into the surrounding cooler fuel. This has an added benefit in that raising the fuel temperature within the hopper helps to alleviate cold-soak conditions.

F. Check Valves, Snubbers and Pressure Transmitters:

Pressurized fluid from each system's engine driven pump passes through a check valve installed to prevent reverse flow to the pump. A snubber installed downstream of the check valve at the pressure transmitter's inlet port damps any pressure surges in order to prevent erratic pressure indications.

Each system's pressure transmitter receives 28V DC power from the Essential DC bus. It then provides a variable output signal, proportional to fluid pressure, to the respective DAU. The DAU in turn prompts these signals for display on the Engine Instruments display, ENGINE START system page and HYDRAULICS system page.

G. Pressure Switches:

Each hydraulic system has a pressure switch installed downstream of the pressure transmitter to monitor system pressure.

As Combined system pressure builds to 1500 psi, its pressure switch opens, causing the amber CMB HYD FAIL message to be removed from CAS. In addition, the system's low pressure relay is de-energized. Conversely, as pressure falls to 800 psi, the pressure switch closes, causing the amber CMB HYD FAIL message to be displayed on CAS and the system's low pressure relay to be energized. (See Section 2A-29-30, Auxiliary Hydraulic Systems, for a description of the low pressure relay.)

As Flight system pressure builds to 1500 psi, its pressure switch opens, causing the amber FLT HYD FAIL message to be removed from CAS. As pressure falls to 800 psi, the pressure switch closes, causing the amber FLT HYD FAIL message to be displayed on CAS.

NOTE:

Automatic operation of the utility pump is not possible with Combined system pressure greater than 800 psi or with Flight system pressure less than 2000 psi.

H. System Filter Manifolds:

(See Figure 8 and Figure 9.)

The Combined and Flight hydraulic systems each have a dedicated filter manifold located in the tail compartment (Combined on left side, Flight on right side). Each manifold consists of three filters: main pressure, main return and engine pump bypass.

Differential Pressure Indicators (DPIs) for each filter provide a visual means of noting filter bypass by causing a red indicator on the DPI to “pop up” when the pressure differential between the filter inlet and outlet exceeds a preset value. The DPI indicator may be reset for monitoring or troubleshooting as required.

As fluid flows from the hydraulic pump, it passes through the main pressure filter. This filter is a “non-bypass” filter, meaning there is no alternate path to bypass the filter element; fluid will continue to flow through the element, though at reduced rates.

The main return and engine pump bypass filters are bypass type filters in that a relief valve is incorporated. If the filter element begins to clog, the pressure differential between the filter inlet and outlet opens a relief valve, allowing the fluid to bypass the filter element.

I. System Accumulators:

Each system contains an accumulator installed between the main pressure filter and the using systems to damp any pressure surges caused by the associated engine driven hydraulic pump or the using system. Each accumulator is a 50 cubic inch (0.8 liter) capacity cylinder charged with nitrogen to 1000 psi at 70° F (21.1°C). Filler valves and pressure gauges for the accumulators are installed on the ground test panel (Figure 10).

Each accumulator cylinder contains a free moving piston that divides the cylinder into two chambers. One chamber contains the nitrogen charge

while the other contains pressurized fluid from the hydraulic system. As pressurized fluid acts against the piston, it compresses the nitrogen charge, providing shock damping by absorbing pressure transients. Should pressure within the accumulator exceed 3850 psi, a relief valve will open to vent excess pressure.

J. Pressure Relief Valves:

The Combined system and Flight system each contain a pressure relief valve to prevent system overpressurization. The Combined system relief valve is installed in the left main wheel well while the Flight system relief valve is installed in the tail compartment.

If system pressure exceeds 3850 psi, the relief valve opens to route system pressure back to the respective reservoir through the main return filter. The using systems are still supplied with pressurized fluid. When system pressure falls to 3200 psi, the valve closes.

K. Ground Test Panel:

The ground test panel (Figure 10) is located on the underside of the aircraft, just forward of the tail compartment door. It contains a reservoir fill, pressure and suction quick disconnect group for both the Combined and Flight systems. It also contains an accumulator air filler valve, accumulator pressure gauge and reservoir quantity gauge for each hydraulic system.

L. Replenishing System:

The Combined system and Flight system reservoirs can be replenished through use of a hydraulic replenisher panel (Figure 11) located in the tail compartment. The reservoir to be serviced is selected manually with a control valve, with actual replenishment taking place through use of a hand pump. Reservoir quantity can then be viewed on the hydraulic quantity indicator (with hydraulic power applied). Procedures for Combined system and Flight system hydraulic reservoir servicing can be found in Chapter 9: Handling and Servicing Procedures.

3. Controls and Indications:

NOTE:

A description of the Engine Instruments display, ENGINE START system page and HYDRAULICS system page can be found in Section 5: Engine Instruments and Crew Alerting System (EICAS), of Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV.

A. Circuit Breakers:

The main hydraulic system is protected by the following circuit breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
COMB HYD PRESS	CPO	A-3	ESS 28 VDC Bus
COMB HYD QTY	CPO	A-4	ESS 28 VDC Bus
L HYD S/O	CPO	A-5	ESS 28 VDC Bus
FLT HYD QTY	CPO	B-4	ESS 28 VDC Bus

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
R HYD S/O	CPO	B-5	ESS 28 VDC Bus
FLT HYD PRESS	CPO	C-4	ESS 28 VDC Bus
FLIGHT HYD CONT	CPO	C-5	L MAIN 28 VDC Bus

B. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
CHECK HYD QUANTITY	Combined and/or Flight hydraulic system quantity is low.
CMB HYD FAIL	Combined hydraulic system pressure low.
FLT HYD FAIL	Flight hydraulic system pressure low.

C. Advisory (Blue) Messages and Annunciations:

CAS Message:	Cause or Meaning:
CMB HYD HOT	Combined hydraulic system reservoir fluid temperature has reached 220 ±5°F (104 ±3°C).
FLT HYD HOT	Flight hydraulic system reservoir fluid temperature has reached 220 ±5°F (104 ±3°C).

4. Limitations:

A. Flight Manual Limitations:

(1) Hydraulic Fluids:

The following fire-resistant Type IV hydraulic fluids are approved for use:

- Hy-Jet IV
- Hy-Jet IV-A
- Skydrol LD-4
- Skydrol 500B-4

B. System Notes:

(1) Although pulling the FIRE handle inhibits hydraulic fluid flow from the reservoir to its respective engine-driven hydraulic pump, it also closes the fuel shutoff valve, shuts off the alternator and disables the thrust reverser, effectively shutting down the associated engine. The flight crew should not pull the FIRE handles except in circumstances when available guidance indicates doing so.

(2) Hydraulic fluids conforming to MIL-H-5606 and MIL-H-83282 are used in the landing gear shock struts of Gulfstream IV aircraft. Hydraulic fluids conforming to MIL-H-5606 and MIL-H-83282 are red in color and are **NEVER, UNDER ANY CIRCUMSTANCES**, to be mixed with the Type IV Phosphate Ester-based hydraulic fluids used in the hydraulic systems of Gulfstream IV aircraft. Type IV Phosphate Ester-based hydraulic fluids are purple in color.

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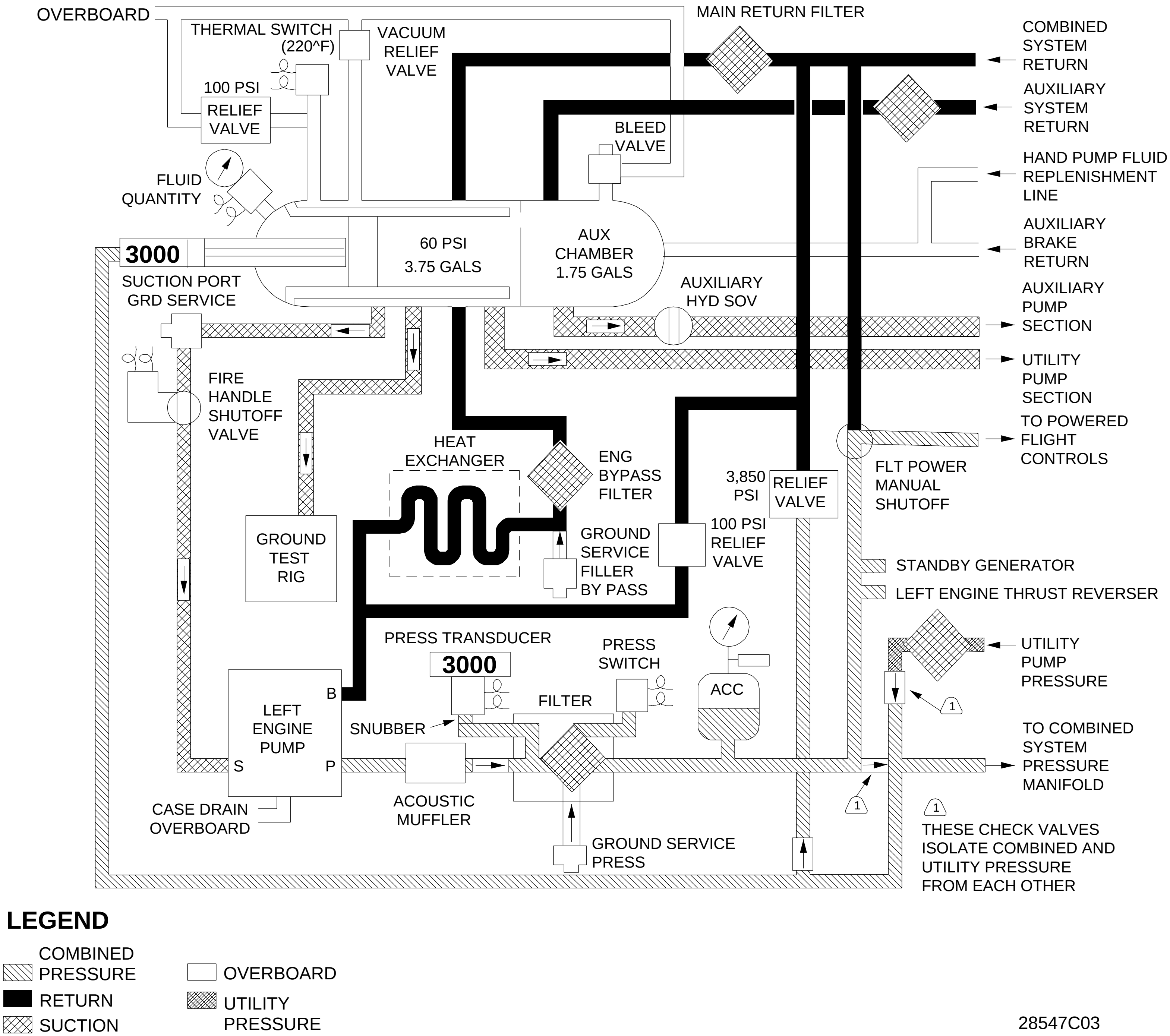


Figure 2. Combined Hydraulic System Simplified Block Diagram

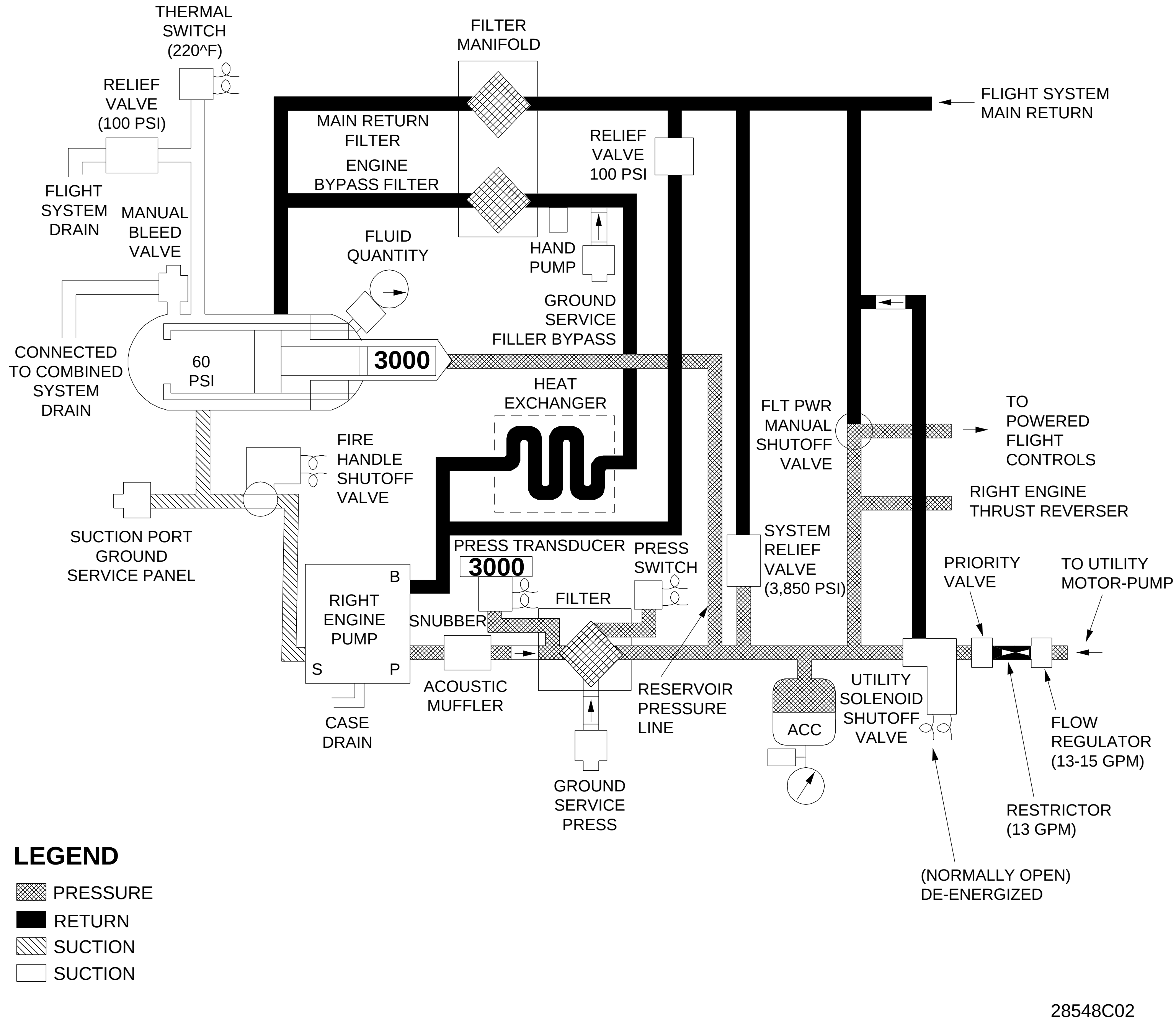


Figure 3. Flight Hydraulic System Simplified Block Diagram

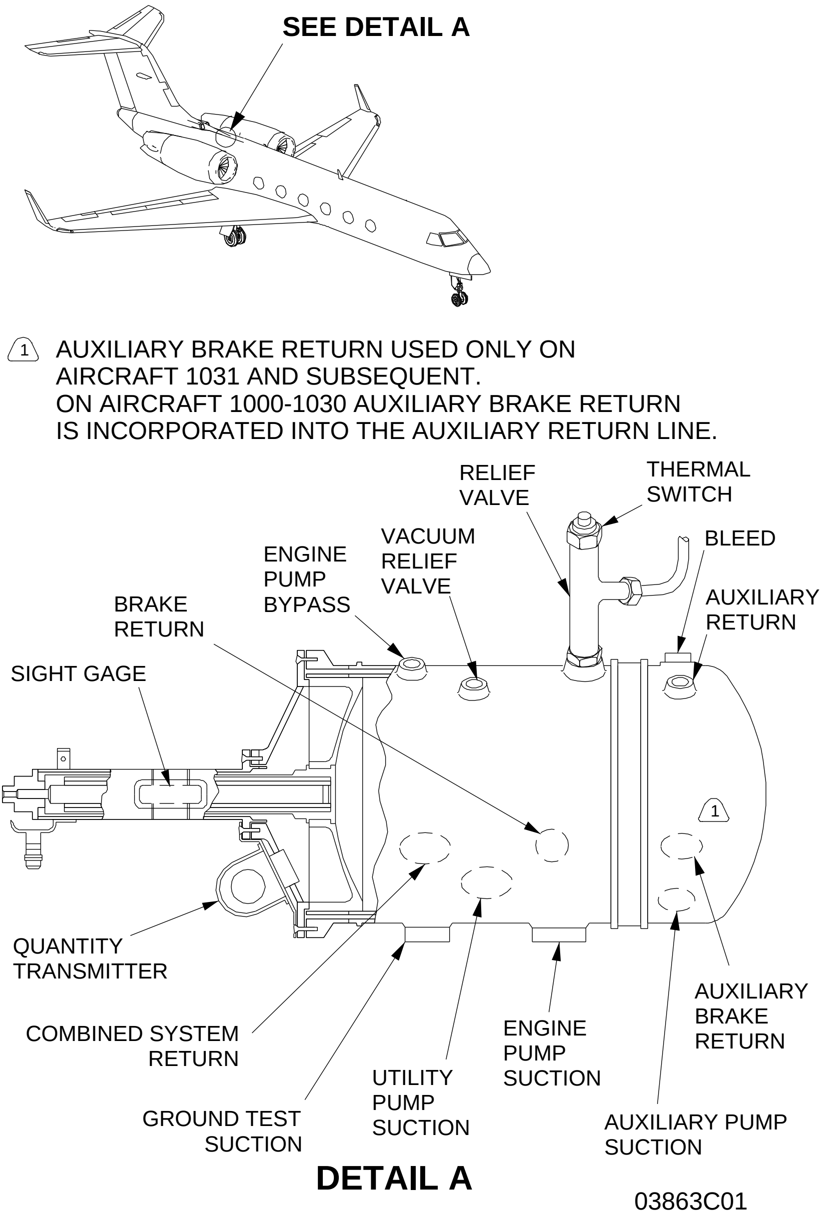


Figure 4. Combined Hydraulic System Reservoir

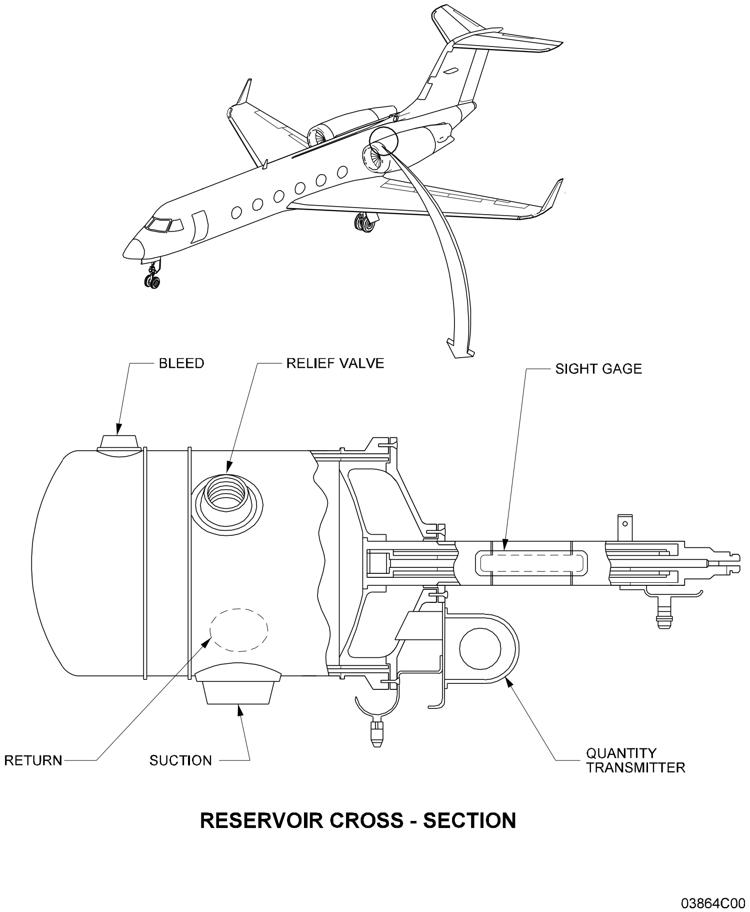


Figure 5. Flight Hydraulic System Reservoir

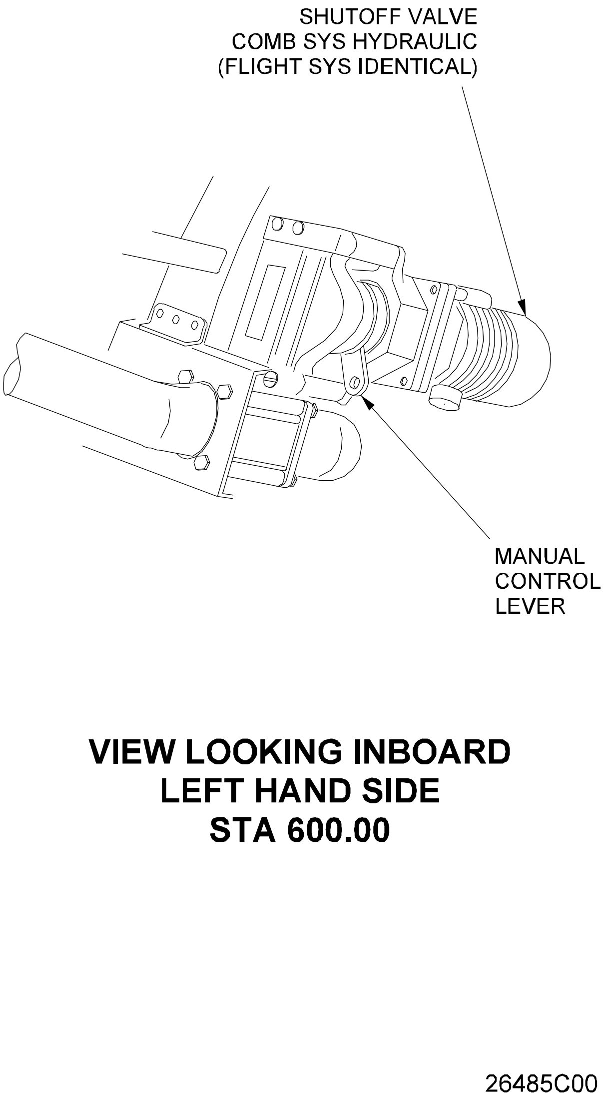


Figure 6. System Shutoff Valve

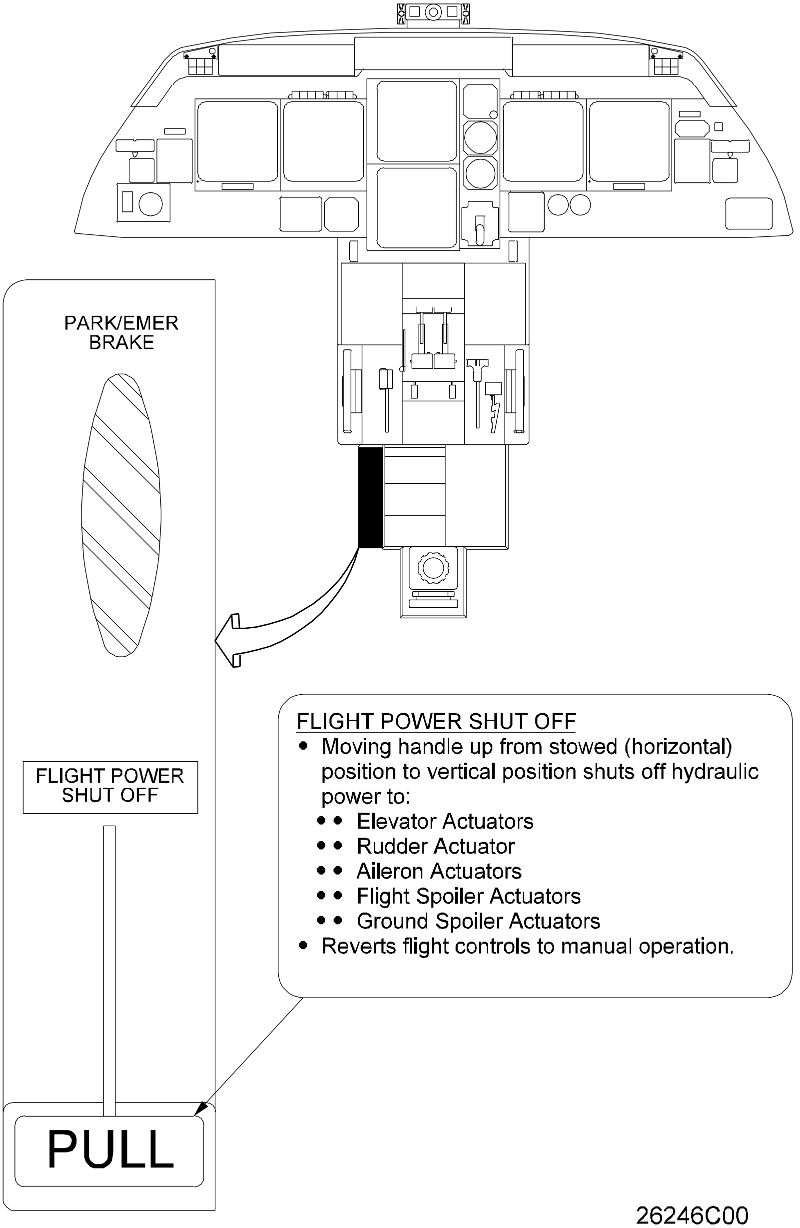


Figure 7. FLIGHT POWER SHUT OFF Valve

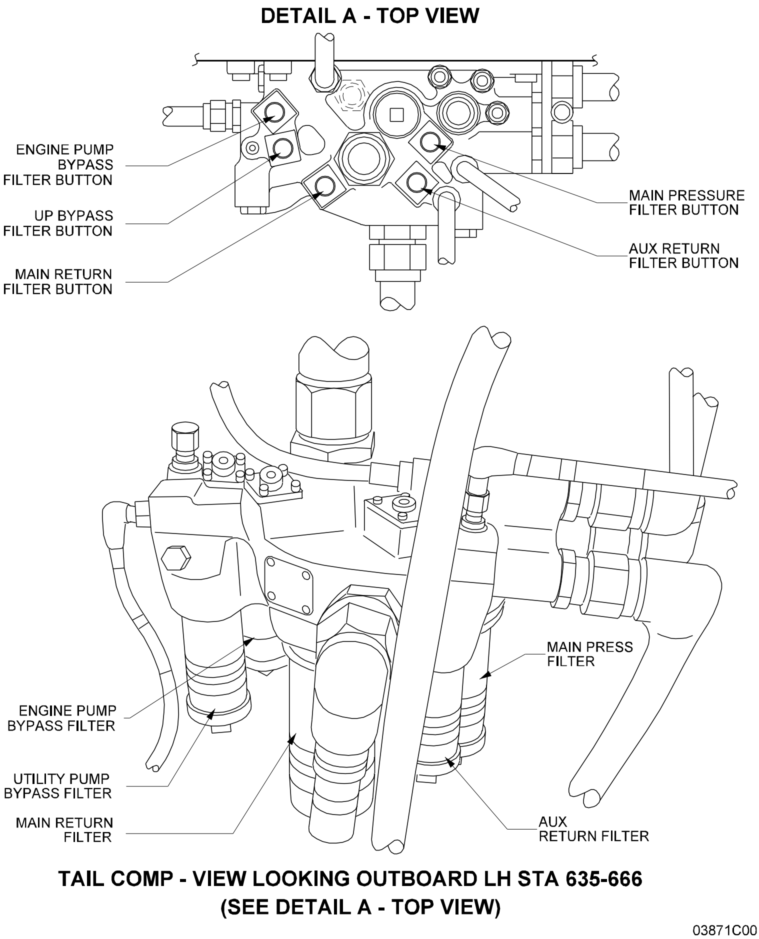
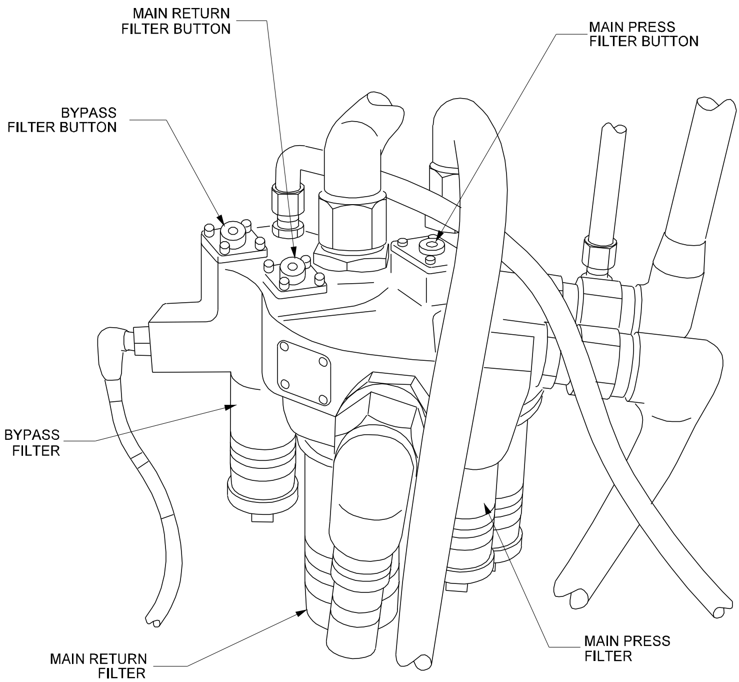


Figure 8. Combined Hydraulic System Filter Manifold



TAIL COMP - VIEW LOOKING OUTBOARD RH STA. 650-668

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Figure 9. Flight Hydraulic System Filter Manifold

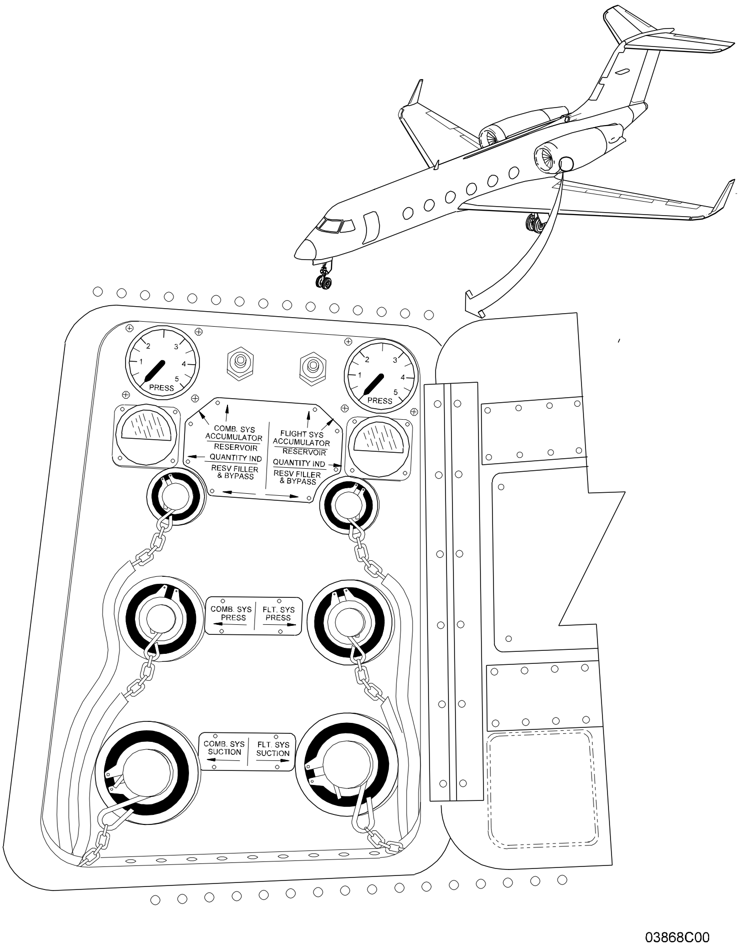

Figure 10. Ground Test Panel

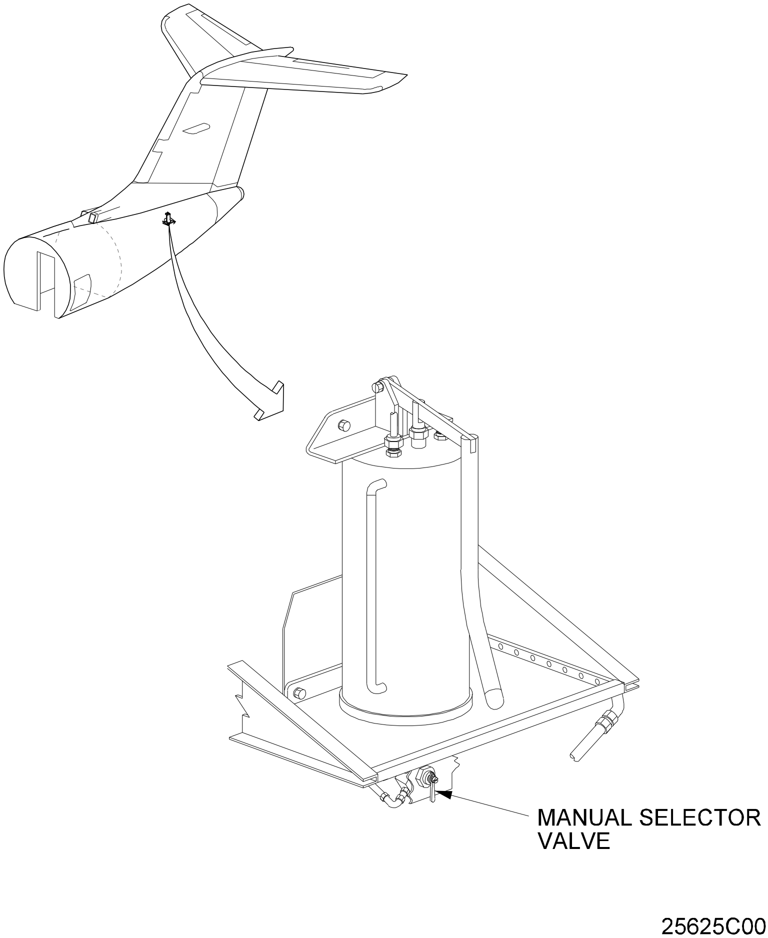

Figure 11. Onboard Hydraulic Replenisher

2A-29-30: Auxiliary Hydraulic Systems

In addition to the two main hydraulic systems, two auxiliary hydraulic systems are incorporated to provide operational redundancy and to perform various utility functions, in some cases without the need for operating engines or external power sources. The two auxiliary hydraulic systems are the electrically-driven Auxiliary system and the hydraulically-driven Utility system. When accordingly configured, both systems are capable of automatic operation when certain conditions are satisfied. In addition, they are capable of manual operation through use of associated control switches.

The Auxiliary and Utility hydraulic systems are described in the following subsections within this section:

- 2A-29-31: Auxiliary Hydraulic System
- 2A-29-32: Utility Hydraulic System

2A-29-31: Auxiliary Hydraulic System

1. General Description:

The Auxiliary hydraulic system (Figure 12) receives pressure from an electrically driven pump which draws fluid from the Auxiliary chamber of the Combined hydraulic system reservoir. The pump provides 3000 psi for the operation of the following systems:

- Wing Flaps
- Ground Spoiler Control Pressure
- Brakes
- Main Entrance Door (see below)
- Parking/Emergency Brakes (see below)
- Landing Gear and Landing Gear Doors (see below)
- Cargo Door (see below)

The Auxiliary hydraulic system functions independently as the only source of hydraulic power for closing the main entrance door and charging the parking/ emergency brakes accumulator. In addition, the Auxiliary hydraulic system may be used for ground operation of the landing gear and landing gear doors for maintenance checks, through the use of a ground service valve.

On aircraft with Aircraft Service Change (ASC) 213 or ASC 354 incorporated, the Auxiliary hydraulic system also serves as the only source of hydraulic power for opening and closing the cargo door.

2. Description of Subsystems, Units and Components:

The Auxiliary hydraulic system consists of the following units and components:

A. Auxiliary Reservoir Chamber:

The Combined system reservoir contains two chambers separated by a bulkhead; one chamber supplies the Combined and Utility systems, and the other supplies the Auxiliary system. Total capacity of the reservoir is 5.5 gallons (20.8 liters) with an Auxiliary system working volume of 1.75 gallons (6.6 liters). Both chambers are full under normal operating conditions. An opening at the top of the reservoir bulkhead allows fluid flow from the Combined chamber to the Auxiliary chamber. Filling of both chambers can be accomplished using either the fill connection on the ground test panel (underside of the aircraft, just forward of the tail compartment door) or a replenishing system (inside the tail compartment).

B. Shutoff Valve:

An electrically operated solenoid shutoff valve controls fluid flow to the Auxiliary hydraulic pump. Controlled by the same circuits that operate the Auxiliary hydraulic pump power relay, the shutoff valve receives 28V DC power from the Essential DC bus through the AUX HYD PUMP circuit breaker. When the Auxiliary hydraulic pump receives power, the shutoff valve opens to allow hydraulic fluid from the Auxiliary chamber to the Auxiliary pump.

C. Auxiliary Hydraulic Pump:

The electrically driven Auxiliary hydraulic pump, commonly referred to as the “Aux pump”, is installed on the aft side of the left main wheel well. It consists of a fan-cooled 28V DC motor that drives a variable-delivery, axial-piston type pump.

The Aux pump motor receives 28V DC from the Essential DC bus (via the Battery Tie bus) through the Aux pump power relay. The power relay, in turn, receives power from the Essential DC bus through the AUX HYD PUMP circuit breaker.

When activated, the Aux pump motor initially draws a maximum of 750 amps, decreasing to 66-70 amps as the motor reaches operating speed. If the motor continuously draws more then 80 amps, the Aux pump overload sensor trips the AUX HYD PUMP circuit breaker. This de-energizes the Aux pump power relay and the Aux pump stops.

A thermal switch is installed on the Aux pump case to monitor pump temperature. If pump temperature reaches 300 ±10°F (149 ±6°C), the thermal switch closes causing an amber AUXILIARY HYD HOT caution message to be displayed on the Crew Alerting System (CAS).

D. Pressure Snubber and Transmitter:

A pressure snubber is installed upstream of the pressure transmitter to damp any pressure surges in order to prevent erratic pressure indications. The pressure transmitter receives 28V DC power from the Essential DC bus through the AUX HYD PRESS circuit breaker. It then provides a variable output signal, proportional to fluid pressure, to the Data Acquisition Unit (DAU). The DAU, in turn, prompts these signals for display on the Engine Instruments display, ENGINE START system page and HYDRAULICS system page.

E. System Filters:

The Auxiliary hydraulic system main and return filters are bypass type filters in that a relief valve is incorporated. If either filter element begins to clog, the pressure differential between the filter inlet and outlet opens a relief valve, allowing the fluid to bypass the filter element. Both filters also include a check valve that prevents back pressure from reaching the Aux pump discharge port.

Differential Pressure Indicators (DPIs) for each filter provide a visual means of noting filter bypass by causing a red indicator on the DPI to “pop up” when the pressure differential between the filter inlet and outlet exceeds a preset value. The DPI indicator may be reset for monitoring or troubleshooting as required.

F. Relief Valve:

The Auxiliary hydraulic system contains a pressure relief valve to prevent system overpressurization. The relief valve is installed in the left main wheel well. If system pressure exceeds 3850 psi, the relief valve opens to route system pressure back to the Combined system reservoir Auxiliary chamber. When system pressure falls to 3500 psi, the valve closes.

3. Controls and Indications:

A. System Control:

(1) AUX PUMP OFF/ARM and ON/OFF Switches (Figure 18):

The Aux pump is normally operated with the AUX PUMP ON/OFF switch located in the HYDRAULIC CONTROL section of the cockpit overhead panel. Selection to ON starts the pump, deslection stops the pump.

The AUX PUMP OFF/ARM switch provides a means of automatically starting the Aux pump. With the OFF/ARM switch selected to ARM, the Aux pump will start automatically if:

- Either nutcracker is in the GROUND mode
- Combined and Utility hydraulic system pressures are less than 1500 psi
- Any toe brake pedal is depressed more than 10°

With the OFF/ARM switch selected to OFF, automatic operation is inhibited.

(2) DOOR Switch Controls (Figure 13 and Figure 14):

As stated previously, the Aux pump is the only source of hydraulic power for closing the main entrance door. Provided the DOOR SAFETY SWITCH is not selected to SAFE, the Aux pump will start automatically when either the INSIDE DOOR CONT SWITCH (pilot’s circuit breaker panel) or OUTSIDE DOOR SWITCH (forward left fuselage) are selected to CLOSE. It will continue running until the door is closed and locked, when it will shut off automatically and return the appropriate DOOR switch to neutral.

(3) Ground Service Valve Controls (Figure 15):

A ground service valve is located inside an access door just aft of the right nose landing gear door. With power applied to the Essential 28V DC bus, movement of the spring-loaded valve handle permits fluid flow to the landing gear and landing gear doors. At the same time, the moving handle contacts an Aux pump control switch, activating the Aux pump. Releasing the handle stops the Aux pump and returns the valve to its normal position.

(4) Cargo Door Switch Controls:

On aircraft with ASC 213 or ASC 354 incorporated, the Auxiliary hydraulic system also serves as the only source of hydraulic power for opening and closing the cargo door. Unlike main entrance door operation, however, cargo door operation requires that the Aux pump be manually selected on and off using the cockpit switches when opening and closing the cargo door using its OPEN/CLOSE switch. For a detailed description of the cargo door, see Section 2A-52-00, Doors.

B. Auxiliary Hydraulic Pump Controls – Preflight Check:

During preflight, automatic operation of the Aux pump should be tested as follows:

(1) Verify that both Combined and Utility hydraulic system pressures are below 1500 psi.

(2) Select the AUX PUMP OFF/ARM switch to ARM (NOT ARM switch legend extinguished).

(3) Depress any brake pedal.

(4) Verify the Aux pump starts and remains on after releasing the brake pedal.

(5) Select the AUX PUMP OFF/ARM switch to OFF (NOT ARM switch legend illuminated).

C. Circuit Breakers:

The Auxiliary hydraulic system is protected by the following circuit breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
AUX HYD PRESS	CPO	D-4	ESS 28 VDC Bus
AUX HYD PUMP	CPO	D-5	ESS 28 VDC Bus

D. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
AUXILIARY HYD HOT	Auxiliary hydraulic pump case temperature above 300°F (149°C).

E. Other Indications:

A description of the Engine Instruments display, ENGINE START system page and HYDRAULICS system page can be found in Section 5: Engine Instruments and Crew Alerting System (EICAS), of Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV.

4. Limitations:

A. Flight Manual Limitations:

There are no flight manual limitations for the Auxiliary hydraulic system at the time of this revision.

B. System Notes:

Because of the high amperage required for Aux pump operation, consider using external electrical power during periods requiring prolonged use of the pump.

2A-29-32: Utility Hydraulic System

1. General:

The Utility hydraulic system (Figure 16) provides operational redundancy to the Combined hydraulic system. In the event of Combined system failure or Combined system pressure falling to less than 800 psi (such as a windmilling engine), a hydraulic motor powered by Flight hydraulic system pressure operates a hydraulic pump installed in the Combined hydraulic system. The pump, in turn, pressurizes Combined system fluid to 3000 psi for use by the following systems:

- Wing Flaps
- Landing Gear
- Brakes
- Stall Barrier
- Nosewheel Steering

The Utility system also supplies a servo pressure signal to the ground spoiler system, thus enabling operation of the ground spoilers using Flight system pressure, provided all other ground spoilers operational conditions are satisfied.

The Utility system will operate automatically if all of the following conditions are satisfied:

- Combined system hydraulic reservoir fluid quantity is greater than one gallon
- Flight hydraulic system pressure is greater than 2000 psi
- Flight hydraulic system fluid temperature is below 220 ±5°F (104 ±3°C)
- UTILITY PUMP OFF/ARM switch is selected to ARM
- UTILITY PUMP ON/OFF switch is selected to OFF

2. Description of Subsystems, Units and Components:

The Utility hydraulic system consists of the following units and components:

A. System Shutoff Valve:

An electrically operated solenoid shutoff valve controls Flight hydraulic system fluid flow to the Utility hydraulic motor/pump. Normally, with the Combined and Flight hydraulic systems operating at 3000 psi, the solenoid energizes to the closed position to prevent motor/pump operation.

The UTILITY PUMP OFF/ARM and ON/OFF switches (cockpit overhead panel) control the shutoff valve. With the OFF/ARM switch in the ARM position (NOT ARM legend not illuminated) and the ON/OFF switch in the OFF position (ON legend not illuminated), the solenoid valve energizes and de-energizes as conditions are satisfied. Depressing the ON/OFF switch (ON legend illuminated) de-energizes the shutoff valve. The valve then opens, allowing Flight hydraulic system pressure to the motor/pump regardless of the OFF/ARM switch position.

B. Priority Valve:

A priority valve is installed downstream of the shutoff valve to control fluid flow to the motor/pump. Depending on Flight hydraulic system pressure, the priority valve opens and closes to prevent loss of Flight hydraulic system pressure to critical systems such as the flight controls.

When Flight hydraulic system pressure at the valve inlet is above 2000 psi, the priority valve opens and Flight hydraulic system pressure flows to the motor/pump. If demands on the system increase such that pressure at the valve inlet drops below 1730 psi, the valve closes, inhibiting flow to the motor/pump, preventing pressure starvation to critical systems. When Flight hydraulic system pressure rises to 2000 psi, the priority valve reopens.

C. Flow Regulating Valve:

A flow regulating valve installed between the priority valve and the motor/pump limits the Flight hydraulic system flow rate to the motor/pump to 13 Gallons Per Minute (GPM). This prevents the motor/pump from using the entire 22.5 GPM provided by the Flight hydraulic system. The lower volume allowance also assists the motor/pump in reaching its maximum operating speed.

D. Utility Hydraulic Motor/Pump:

The Utility hydraulic motor/pump consists of a fixed-displacement, hydraulically-driven motor that drives a fixed-displacement hydraulic motor by means of an interconnect shaft. The motor/pump assembly is installed in the right side of the tail compartment.

A 1.4 GPM restrictor connects the pump pressure port to the main return line. The restrictor allows a small amount of fluid to circulate within the pump to assist in damping pressure surges, reduce pump noise and reduce starting torque.

E. Pressure Snubber and Transmitter:

A pressure snubber is installed upstream of the pressure transmitter to damp any pressure surges in order to prevent erratic pressure indications. The pressure transmitter receives 28V DC power from the Essential DC bus through the UTILITY HYD PRESS circuit breaker. It then provides a variable output signal, proportional to fluid pressure, to the DAU. The DAU, in turn, prompts these signals for display on the Engine Instruments display, ENGINE START system page and HYDRAULICS system page.

F. System Filters:

The Utility hydraulic system main pressure filter is a “non-bypass” filter, meaning there is no alternate path to bypass the filter element. Fluid will continue to flow through the element at increasingly reduced rates until ceasing completely.

The Utility pump bypass filter is a bypass type filter in that a relief valve is incorporated. If either filter element begins to clog, the pressure differential between the filter inlet and outlet opens a relief valve, allowing the fluid to bypass the filter element.

DPIs for each filter provide a visual means of noting filter bypass by causing a red indicator on the DPI to “pop up” when the pressure differential between the filter inlet and outlet exceeds a preset value. The DPI indicator may be reset for monitoring or troubleshooting as required.

G. Hydraulic Pump Control Circuit:

(See Figure 17).

(1) General:

An electrically-operated hydraulic pump control circuit is incorporated to ensure fail-safe pressure output of the Combined, Flight and Utility hydraulic system pumps during various scenarios. It also provides visual indications (in the form of CAS messages) should the Combined or Flight hydraulic systems overheat or fail. The circuit is composed of the Utility hydraulic system solenoid shutoff valve, the Combined and Flight hydraulic system pressure switches, the Combined and Flight hydraulic system thermal switches and various relays that control the sequence of circuit operation. All scenarios described in the following paragraphs assume proper configuration of the system.

(2) Engine Start Scenario:

As the right engine is started and Flight hydraulic system pressure increases above 1500 psi, the Flight hydraulic system pressure switch opens, causing the amber FLT HYD FAIL message to be

removed from CAS. Since the left engine has not yet been started, Combined hydraulic system pressure is zero. The amber CMB HYD FAIL message is displayed on CAS and the Combined hydraulic system pressure switch is closed. With the pressure switch closed, the hold relay is de-energized, thus no electrical power can be routed to close the Utility hydraulic system solenoid shutoff valve. Since the shutoff valve is open, the Utility pump operates, supplying 3000 psi of hydraulic pressure to its using systems and components. At this point, hydraulic pressure configuration is: Combined = 0, Flight = 3000 and Utility = 3000.

As the left engine is started (with the right engine running) and Combined hydraulic system pressure increases above 1500 psi, the Combined hydraulic system pressure switch opens, causing the amber CMB HYD FAIL message to be removed from CAS. With the pressure switch open, power is routed through both the hold relay and the UTILITY PUMP OFF/ARM switch and delivered to the Utility hydraulic system solenoid shutoff valve. When the shutoff valve closes, Utility pump operation stops. At this point, hydraulic pressure configuration is: Combined = 3000, Flight = 3000 and Utility = 0. This is the normal ground and flight configuration.

(3) Left Engine Failure On Takeoff Scenario:

Should an engine fail during takeoff, immediate landing gear retraction is essential. Assuming that the left engine has failed and is windmilling, the Combined hydraulic system would most likely be unable to provide landing gear retraction pressure, as the windmilling engine would only provide pressure fluctuating in the 800 psi range. In this range, the Combined hydraulic system pressure switch would cycle open and closed, resulting in intermittent operation of the Utility pump. To prevent intermittent Utility pump operation, the Combined hydraulic system pressure switch is also used to energize a hold relay as system pressure drops below 800 psi. With the landing gear control handle in the UP position and any landing gear door not closed, the hold relay remains energized, overriding any pressure switch cycling. Since the relay is held, no electrical power can be routed to close the Utility hydraulic system solenoid shutoff valve, thus the Utility pump operates, supplying 3000 psi of hydraulic pressure to the landing gear system. At this point, hydraulic pressure configuration is: Combined = 0, Flight = 3000 and Utility = 3000.

With all landing gear doors closed and Combined hydraulic system pressure greater than 1500 psi, the hold relay is de-energized, allowing the Utility hydraulic system solenoid shutoff valve to be energized closed. When the shutoff valve closes, Utility pump operation stops and pressure drops to zero. Should Combined hydraulic system pressure remain below 800 psi, however, the Utility hydraulic system solenoid shutoff valve remains de-energized (open) and Utility hydraulic system pressure remains at 3000 psi.

(4) Combined Hydraulic System Fluctuation Scenario:

If the landing gear doors are closed or the landing gear control handle is in the DOWN position, and Combined hydraulic system

pressure fluctuates in a range of greater than 1500 psi to below 800 psi, the Utility hydraulic system will also cycle on and off. Utility hydraulic system will be 3000 psi when Combined hydraulic system pressure is below 800 psi and 0 psi when Combined hydraulic system pressure is above 1500 psi.

(5) Combined Hydraulic System Inflight Failure Scenario:

If the Combined hydraulic system fails during flight, the pressure switch closes as Combined system pressure falls below 800 psi. This causes the amber CMB HYD FAIL message to be displayed on CAS.

With the pressure switch closed, the hold relay is de-energized, thus no electrical power can be routed to close the Utility hydraulic system solenoid shutoff valve. Since the shutoff valve is open, the Utility pump operates, supplying 3000 psi of hydraulic pressure to its using systems and components. At this point, hydraulic pressure configuration is: Combined = 0, Flight = 3000 and Utility = 3000.

(6) Flight Hydraulic System Inflight Failure Scenario:

If the Flight hydraulic system fails during flight, the pressure switch closes as Flight system pressure falls below 800 psi. This causes the amber FLT HYD FAIL message to be displayed on CAS.

Since the Utility hydraulic system solenoid shutoff valve receives its power from the Combined hydraulic system pressure switch, the valve remains closed and the Utility hydraulic pump does not operate. All components that normally receive hydraulic pressure from the Flight hydraulic system remain operative through the use of Combined hydraulic system pressure, with the exception of the right thrust reverser and the Utility hydraulic system motor.

At this point, hydraulic pressure configuration is: Combined = 3000, Flight = 0 and Utility = 0.

(7) Flight Hydraulic System Overheat Scenario:

If the Flight hydraulic system reservoir fluid temperature reaches 220 ±5°F (104 ±3°C), the thermal switch on the reservoir closes causing a blue FLT HYD HOT advisory message to be displayed on CAS. (The same logic applies to the Combined hydraulic system.) In the case of the Flight hydraulic system, however, in addition to causing the FLT HYD HOT message to be displayed, power is also routed to energize and hold the Utility hydraulic system solenoid shutoff valve closed. This prevents Utility hydraulic pump operation. This feature may be overridden by the flight crew, however, provided Flight hydraulic system is above 2000 psi, by selection of the UTILITY PUMP ON/OFF switch to ON.

If Flight hydraulic system reservoir fluid temperature drops to 175 ±10°F (79 ±6°C), the thermal switch opens, clearing the CAS message.

3. Controls and Indications:

(See Figure 18.)

A. Circuit Breakers:

The Utility hydraulic system is protected by the following circuit breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
UTILITY HYD PRESS	CPO	B-3	ESS 28 VDC Bus
UTILITY HYD PUMP OFF	CPO	C-3	ESS 28 VDC Bus

B. Caution (Amber) Messages and Annunciations:

CAS Message:	Cause or Meaning:
UTILITY HYD OFF	Utility hydraulic pump has been selected off.

C. Other Indications:

A description of the Engine Instruments display, ENGINE START system page and HYDRAULICS system page can be found in Section 5: Engine Instruments and Crew Alerting System (EICAS), of Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV.

4. Limitations:

A. Flight Manual Limitations:

There are no flight manual limitations for the Utility hydraulic system at the time of this revision.

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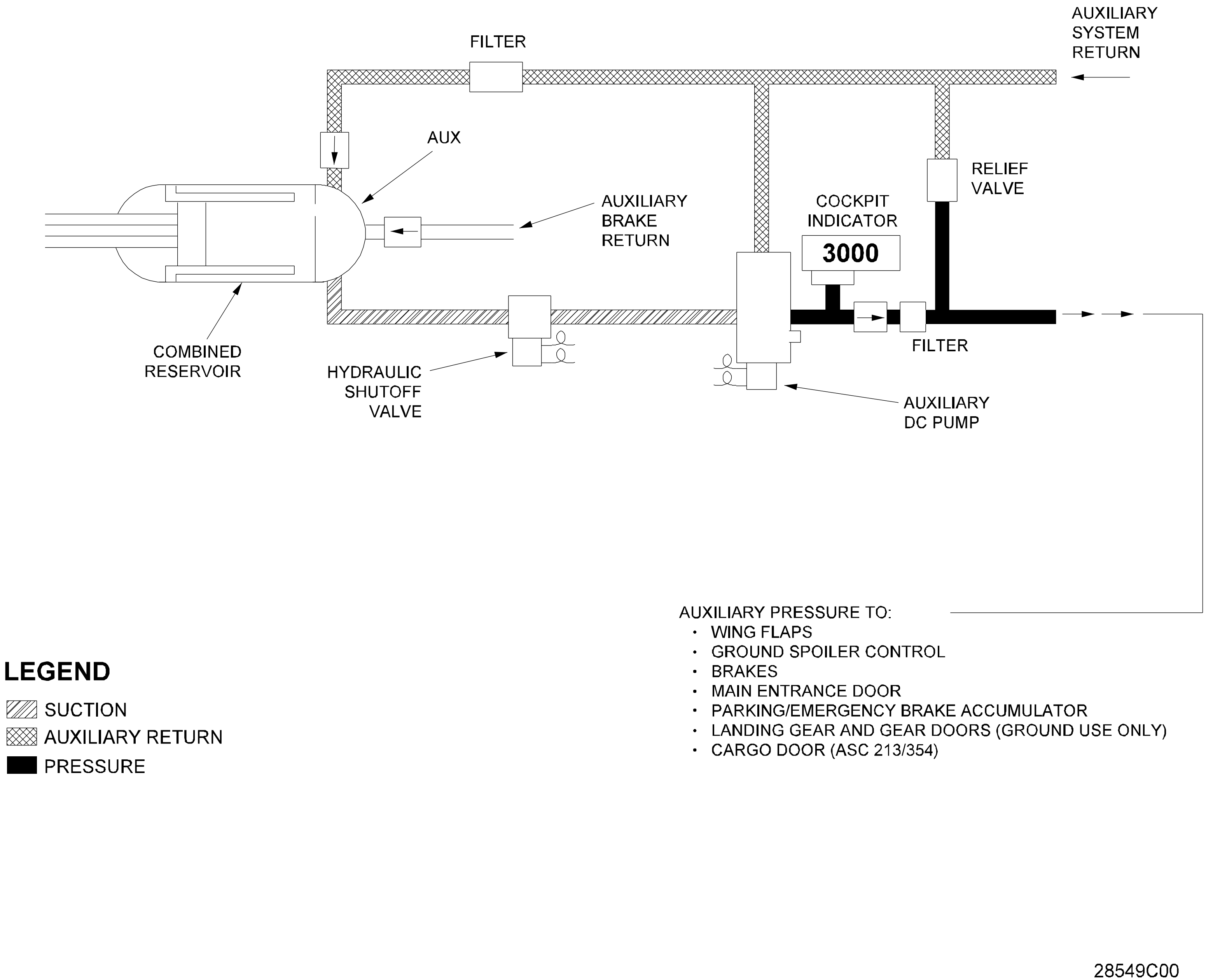

Figure 12. Auxiliary Hydraulic System Simplified Block Diagram

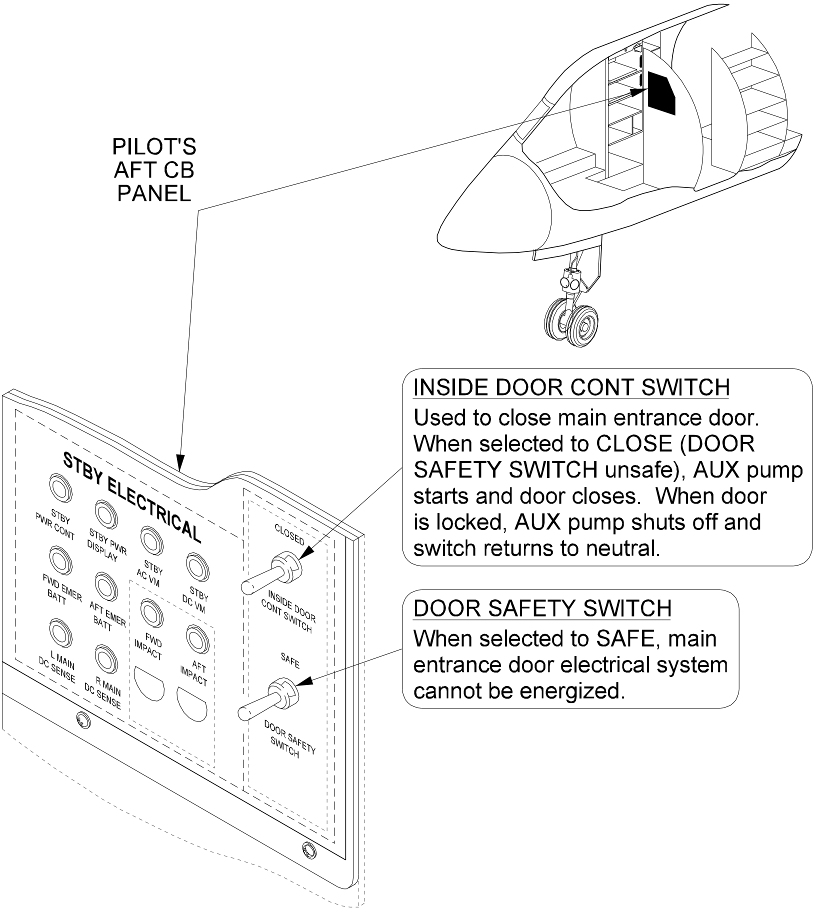

Figure 13. Main Entrance Door Interior Control Switches

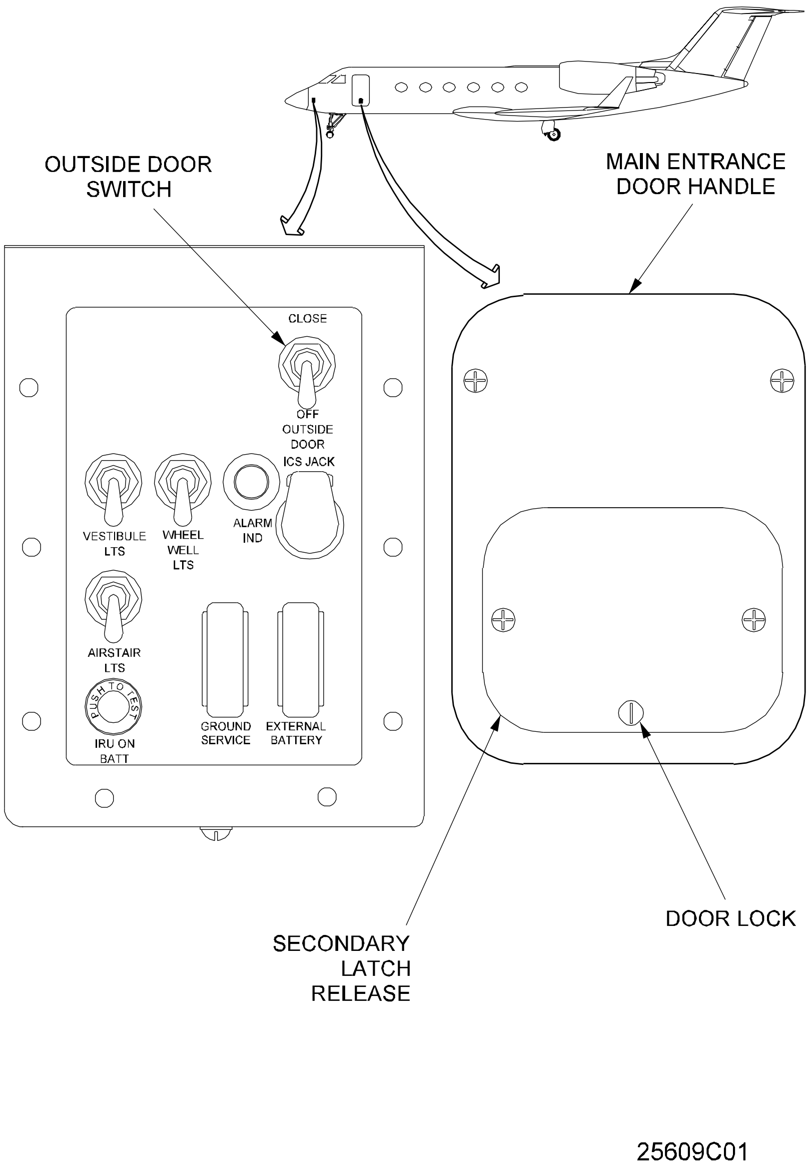

Figure 14. Main Entrance Door Exterior Control Switch

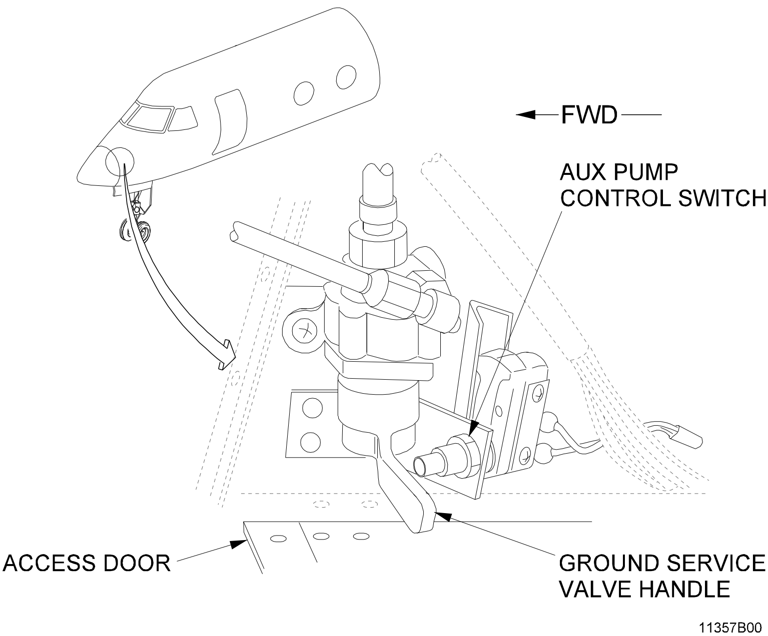

Figure 15. Ground Service Valve

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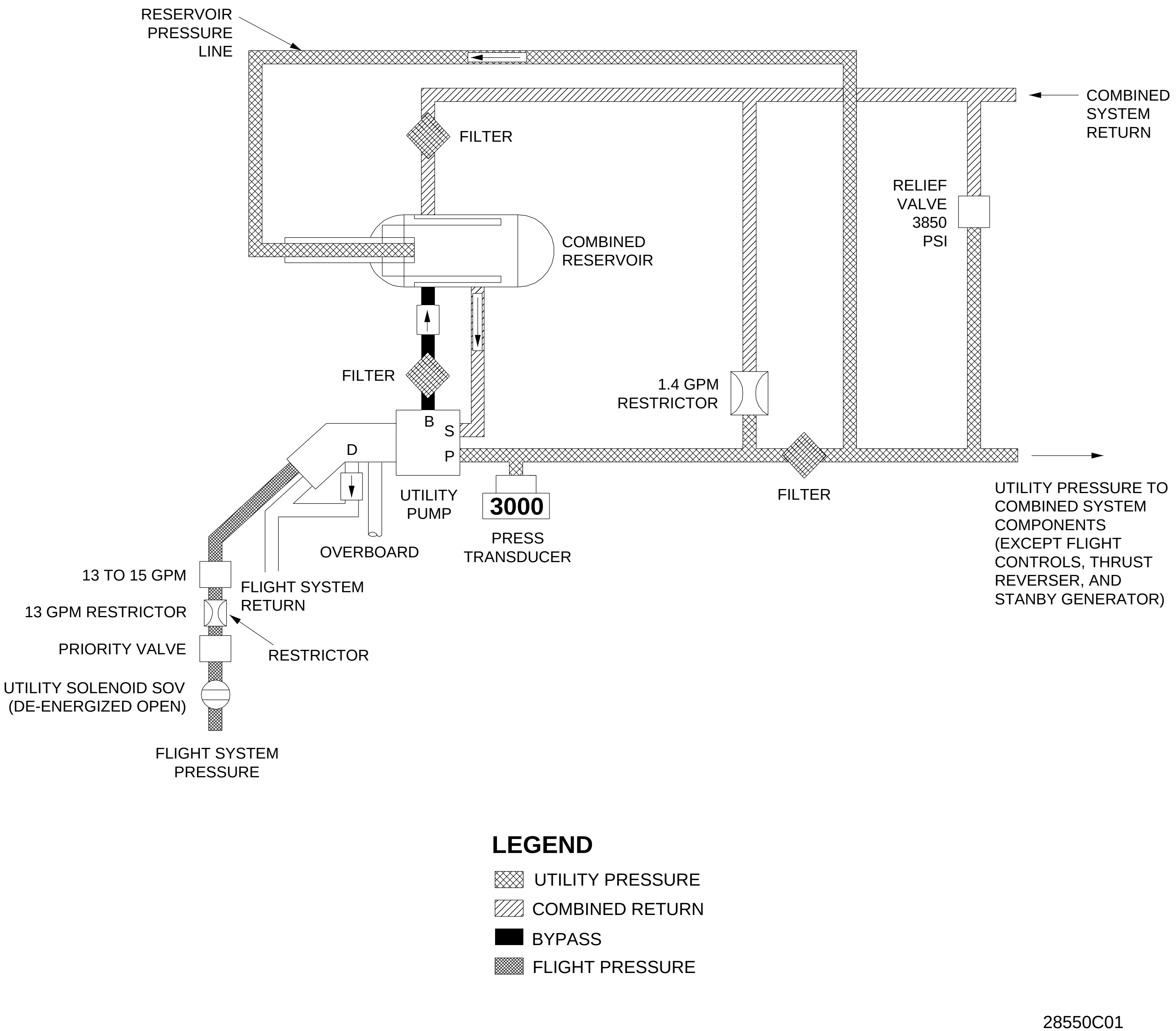

Figure 16. Utility Hydraulic System Simplified Block Diagram

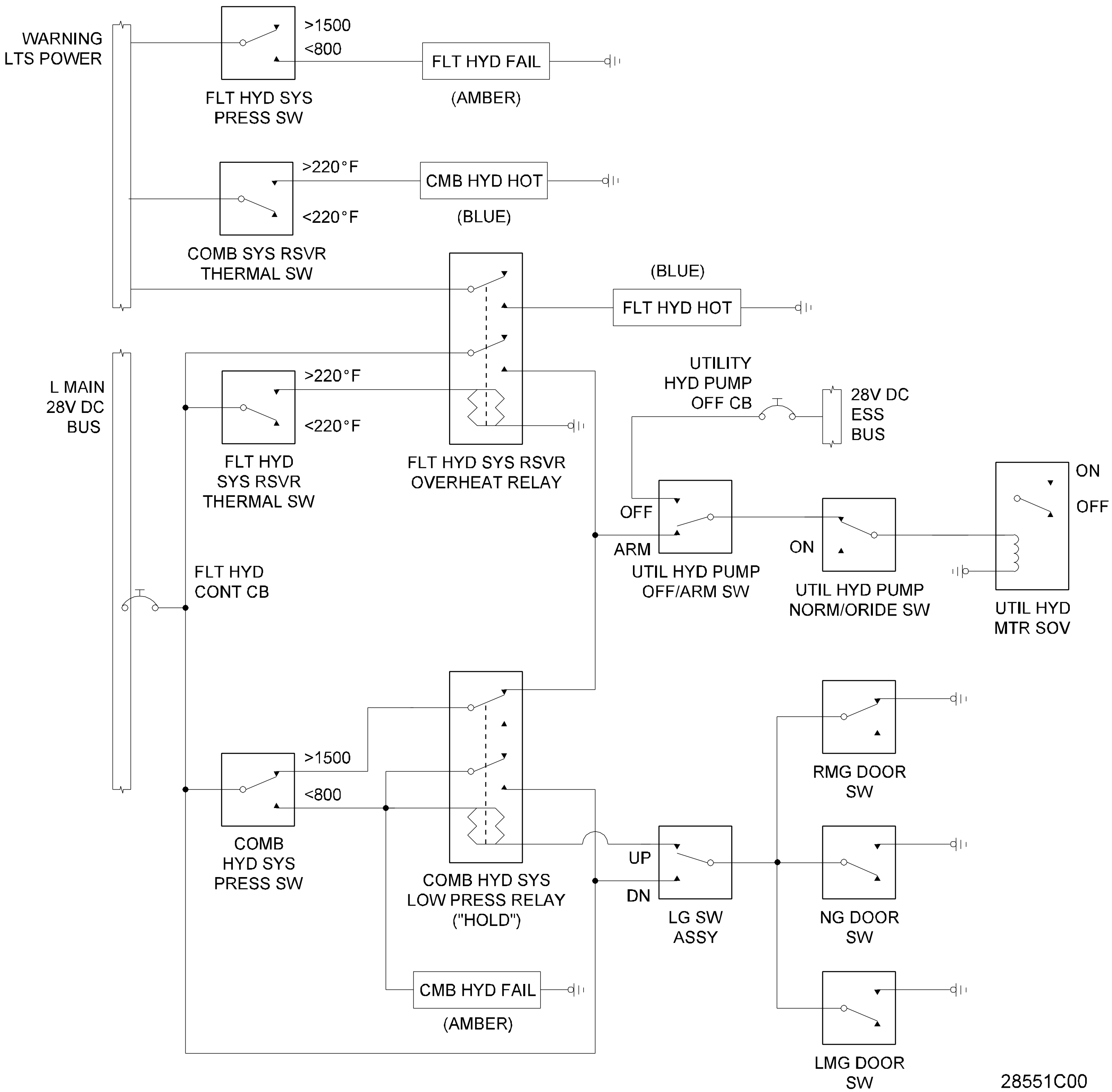

Figure 17. Utility Hydraulic
Pump Control Circuit

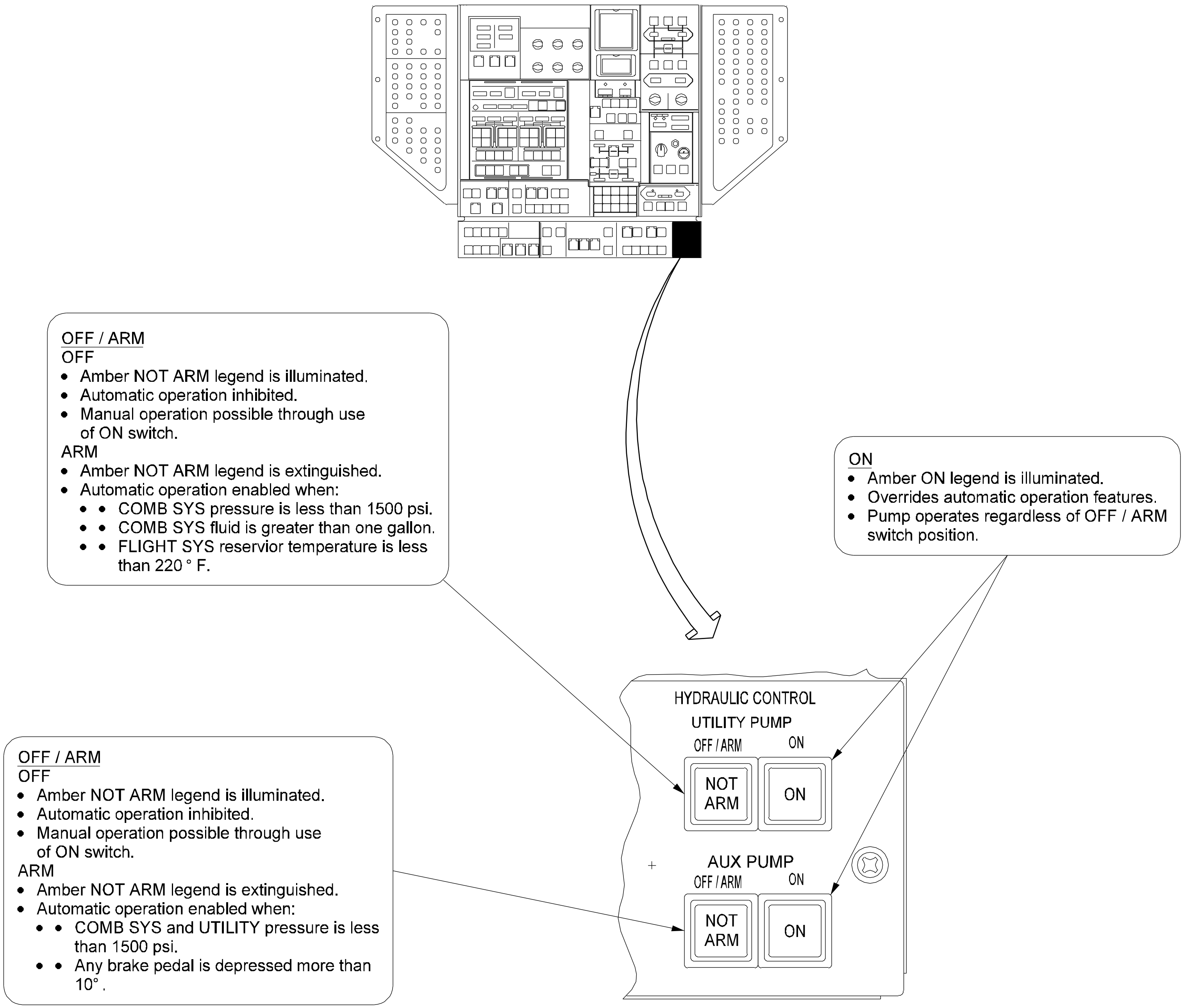

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Figure 18. UTILITY PUMP/AUX PUMP Control Switches

2A-30-10: General Description

The Gulfstream IV uses both pneumatically and electrically powered ice and rain protection systems to provide the flight crew with a means of removing and preventing ice accumulation on the following structures and components:

- Wing Leading Edges
- Engine Cowl Inlet Leading Edges
- Cockpit Windows
- Pitot Probes
- Angle of Attack Probes
- Total Air Temperature Probe

The remainder of the airplane (including the empennage) is designed in a manner that requires no anti-icing equipment.

The ice and rain protection system is divided into the following subsystems:

- 2A-30-20: Wing Anti-Ice System
- 2A-30-30: Cowl Anti-Ice System
- 2A-30-40: Windshield Ice and Rain Protection System
- 2A-30-50: Probe Anti-Ice System

Bleed air for the wing anti-ice system is supplied by the engine bleed air manifold. The engine bleed air manifold is supplied mid-stage air from each engine's 7th stage compressor section through the 7th stage check valve. Depending on conditions, high-stage air may also be supplied by each engine's 12th stage compressor through the 12th stage air control valve (often referred to as the HP valve).

Bleed air for the cowl anti-ice system is supplied directly from the engines' 7th and/or 12th stage compressor offtakes. Each engine also has a mechanical ice shedder on the Low Pressure (LP) compressor (fan) spinner.

Electrically powered heating elements warm the cockpit windows to prevent fogging and ice formation. Precipitation removal from the forward cockpit windows is provided by a pair of electrically operated windshield wipers.

Electrically powered heating elements also warm the pitot probes, Angle of Attack (AOA) probes and Total Air Temperature (TAT) probe.

An optional ice detector is available to provide the flight crew with visual warnings of accumulating ice. Installed on the right, left or both forward cheek panels and powered by the AC electrical system, the detector probe vibrates at approximately 40,000 Hertz (Hz) with no ice accumulated. At this frequency, only ice can adhere to the probe, thus false warnings are eliminated. If ice should begin to accumulate on the probe, the resonate frequency will begin to drop, eventually reaching 39,867 Hz. At this threshold, an amber ICE DETECTED caution message is prompted for display on the Crew Alerting System (CAS) and detector probe heat is energized. Detector probe heat then melts away any accumulated ice, allowing the frequency to again increase to 40,000 Hz, shutting off probe heat. If the probe is still free of ice sixty (60) seconds later, the ICE DETECTED CAS message is removed, indicating the airplane is clear of icing conditions.

There are also production provisions for an ice detector “ICE DET” test button on the TEST panel (copilot side of center console [SN 1457 and subs]).

2A-30-20: Wing Anti-Ice System

1. General Description:

The wing anti-ice system provides ice protection for each wing by warming the leading edge with engine bleed air from the bleed air manifold. It consists of identical, independently operating left and right sides, each side having a tap into the bleed air manifold. Although separate and independent, the left and right sides are joined by a crossover duct installed downstream of the wing anti-ice control valves. This duct allows one side of the system to provide anti-ice protection for both wing leading edges.

The following units and components together compose this system:

- Engine Bleed Air Manifold
- Anti-Ice Sensors
- Anti-Ice Controllers
- Anti-Ice Servo Air Pressure Regulator and Torque Motors
- Anti-Ice Control Valves
- System Ducts and Piccolo Tubes
- Normal and Overheat Thermal Switches

2. Description of Subsystems, Units and Components:

A. Engine Bleed Air Manifold:

The engine bleed air manifold is supplied mid-stage air from each engine's 7th stage compressor section through the 7th stage check valve. Depending on conditions, high-stage air may also be supplied by each engine's 12th stage compressor through the 12th stage air control valve (often referred to as the HP valve).

In addition to supplying air for wing anti-icing, air from the bleed air manifold is used for crossbleed starting and airplane services such as air conditioning and pressurization.

B. Anti-Ice Sensors:

A wing anti-ice sensor is installed on the front beam (spar) of each wing to sense the temperature in the leading edge plenum. Each sensor is a thermistor, thus its resistance varies proportionally to its temperature.

With the wing anti-ice system operating, each sensor sends its associated anti-ice controller a low voltage signal that is proportional to the sensed temperature.

C. Anti-Ice Controllers:

Each anti-ice controller receives power from the Essential 28 VDC bus. The controllers receive low voltage signals from its associated anti-ice sensor, then convert that signal into a high output signal to command its associated anti-ice servo air pressure regulator and torque motor.

D. Anti-Ice Servo Air Pressure Regulator and Torque Motors:

The anti-ice servo air pressure regulator and torque motor controls the pressure to its associated anti-ice control valve using an electrical signal received from its associated anti-ice controller. The electrical signal is then converted into a pneumatic signal used to position the anti-ice control valve proportionally.

E. Anti-Ice Control Valves:

The anti-ice control valve is spring loaded closed and pneumatically powered open. It is electrically controlled by its associated (L/R) WING switch in the ANTI ICE section of the cockpit overhead panel. With the WING switch selected ON, the ON legend in the switch illuminates and a blue (L/R) WING A/I advisory message is displayed on CAS. The control valve is allowed to open and begins to seek a position determined by the pressure downstream of its associated servo air pressure regulator and torque motor. The control valve regulates wing leading edge temperature by controlling bleed air flow to its associated system ducts and piccolo tube. It also regulates downstream pressure to 23 psi any time upstream pressure exceeds this value.

F. System Ducts and Piccolo Tubes:

During normal operation, the left and right wing anti-ice systems are isolated from each other (upstream of the crossover duct) by the isolation valve. The left engine bleed air system supplies the left wing anti-ice system and the right engine bleed air system supplies the right wing anti-ice system. This isolation eliminates the problem two-engine flow-sharing. Should one engine bleed air system fail, however, opening the isolation valve allows the operative engine bleed air system to supply air to both wing anti-ice systems.

A crossover duct is installed downstream of the anti-ice control valves to join the left and right wing system ducts. The ability to join the two systems adds an additional safety feature to the system in that, as long as one control valve is operating, both wings will receive anti-ice air.

From the anti-ice control valve, regulated engine bleed air flows to both wing leading edges through ducts installed beneath the cabin floor. Perforated tubes (called piccolo tubes because of their appearance) inside the wing leading edges distribute the bleed air along the length of the wing leading edge.

Bleed air leaving the piccolo tube heats the leading edge skin, preventing ice formation. The air then circulates aft into the leading edge plenum (where the wing anti-ice sensor is located), eventually exhausting overboard through the wing's center section exhaust port (airplanes Serial Number [SN] 1000 through 1320 not having Aircraft Service Change [ASC] 381). On airplanes SN 1000 through 1320 having ASC 381 and airplanes SN 1321 and subsequent, the air is exhausted into the main landing gear wheel well area to minimize the possibility of frozen brakes.

G. Normal and Overheat Thermal Switches:

Each wing has an inboard and outboard overheat thermal switch and a combined normal/overheat thermal switch approximately mid-wing. All switches receive power from the Essential 28 VDC bus.

(1) Normal Thermal Switch:

When wing temperature reaches 100 ±5° F (38 ±3° C), a normal thermal switch (one in each wing) closes. This illuminates the associated blue WING WARM annunciator on the cockpit overhead panel. The temperature in the wing plenum is modulated (sensor to controller to torque motor to control valve) so that it remains at approximately 130° F (55° C).

(2) Overheat Thermal Switch:

If a malfunction occurs with an increase in wing plenum temperature, any of the three parallel-wired overheat thermal switches will close when the temperature reaches 180 ±5° F (82 ±3° C). This causes an amber L-R WING HOT caution message will be displayed on CAS.

On airplanes SN 1000, 1034, 1129 and subsequent and airplanes SN 1001 through 1128 having ASC 232, should two minutes elapse without the anti-ice duct temperature reaching 100 ±5° F (38 ±3° C), an amber L-R WING TEMP LOW caution message will be displayed on CAS.

3. Controls and Indications:

(See Figure 1 and Figure 2.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
L WING ANTI-ICE	CP	J-14	ESS DC Bus
R WING ANTI-ICE	CP	K-14	ESS DC Bus

B. Caution (Amber) CAS Messages:

CAS Message:	Cause or Meaning:
L-R WING HOT	Wing anti-icing exhaust duct temperature is greater than 180° F (82° C).
L-R WING TEMP LOW **(1)**	Wing anti-ice exhaust duct temperature is still below 100° F (38° C) two (2) minutes after selection of the WING ANTI ICE switch(es) to ON (time delay).

NOTE(S):

(1) SN 1000, 1034, 1129 and subs; SN 1001-1128 having ASC 232.

C. Advisory (Blue) CAS Messages:

CAS Message:	Cause or Meaning:
L-R WING A/I	Wing anti-ice is ON.

D. Other Annunciations:

Annunciation:	Cause or Meaning:
LEFT/RIGHT WING WARM annunciator illuminated (cockpit overhead panel).	Wing anti-ice plenum in wing leading edge(s) has reached minimum operating temperature of 100° F (38° C).

NOTE:

A description of the Engine Instruments and Crew Alerting System (EICAS) can be found in Section 5 of Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV.

4. Limitations:

A. Flight Manual Limitations:

Operation of wing anti-icing is required if icing conditions are imminent, or immediately upon detection of ice formation on wings, winglets, or windshield edges.

B. Notes On Single Anti-Ice System Operation:

If only one wing anti-ice system (left side or right side) is operative (WING ANTI ICE switch selected ON), the operative side will have the associated blue WING A/I advisory message displayed on CAS; the inoperative side will not. This is because the power needed to generate the signal to display the message comes directly from the WING ANTI ICE switch.

Both wings will still receive wing anti-icing air due to the crossover duct. Normal temperature control for both wings will be carried out by the operative side's components. The WING WARM annunciator, however, will not illuminate for the inoperative side.

Overtemperature warning will still be active on both sides, because power for this function is not dependent upon WING ANTI ICE switch position. The amber WING TEMP LOW caution CAS message, however, cannot be displayed for the inoperative side (airplanes SN 1000, 1034, 1129 and subsequent and airplanes SN 1001 through 1128 having ASC 232).

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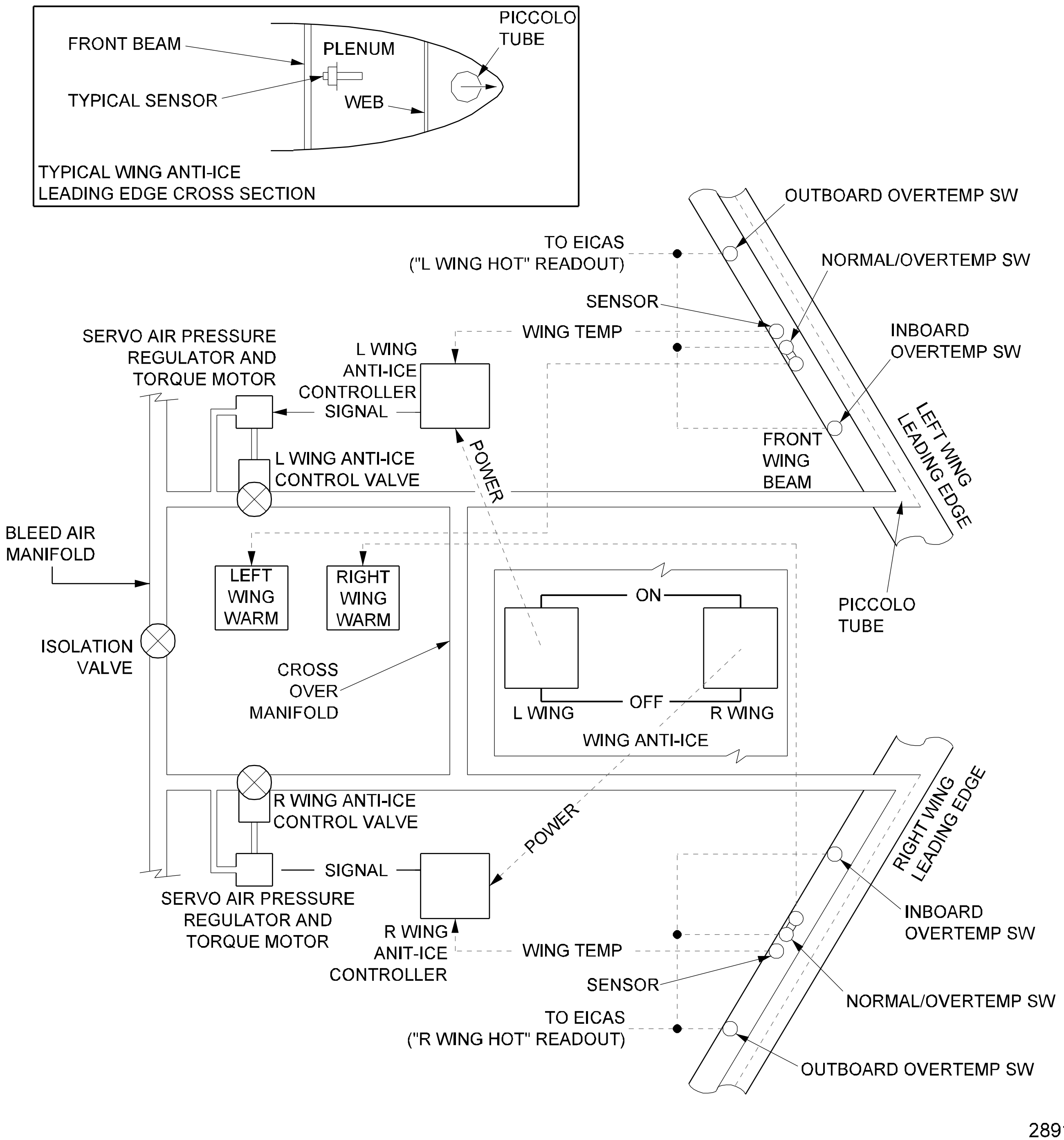
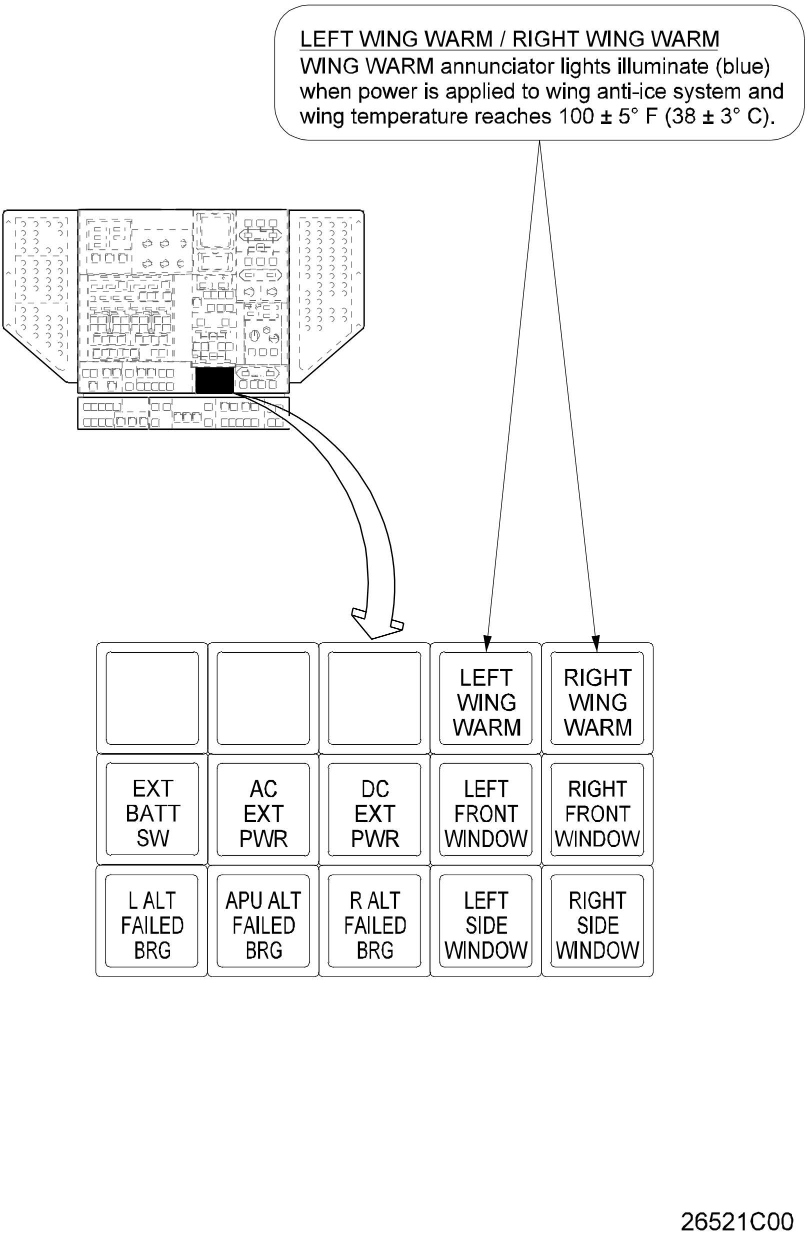


Wing Anti-Ice System
Simplified Block Diagram
Figure 1



WING WARM Annunciators
Figure 2

2A-30-30: Cowl Anti-Ice System

1. **General Description:**

The cowl anti-ice system prevents the formation of ice on the engine nose cowl. This is accomplished by circulating engine bleed air from the bleed air manifold through the leading edge of the engine inlet.

The design of the Tay 611-8 engine does not feature inlet guide vanes to the Low Pressure (LP) compressor (fan), therefore an active ice protection system is not required. Any ice accumulation on the fan spinner is dislodged before reaching significant proportions by a flexible rubber tip on the spinner. The tip spins in an eccentric manner, making any ice accumulation asymmetric in shape. The asymmetric properties of the ice in turn causes it to be shed early in its development by centrifugal force.

The following units and components together compose the cowl anti-ice system:

- Cowl Anti-Ice Shutoff and Pressure Regulating Valves
- Cowl Anti-Ice Pressure Transmitters / Pressure Switches
- Cowl Anti-Ice Thermal Switches
- Nose Cowl Ducts and Piccolo Tubes

2. **Description of Subsystems, Units and Components:**

A. **Cowl Anti-Ice Shutoff and Pressure Regulating Valves:**

The cowl anti-ice shutoff and pressure regulating valve (commonly referred to as the cowl anti-ice valve) is spring loaded closed and pneumatically powered open. It is electrically controlled by its associated (L/R) COWL switch in the ANTI ICE section of the cockpit overhead panel. With the COWL switch selected ON, the ON legend in the switch illuminates, a blue (L/R) COWL A/I ON advisory message is displayed on CAS and a green A/I icon appears above the LP turbine digital scale on the engine instruments display. The control valve is allowed to open and begins to seek a position determined by sensed pressure downstream of the valve. During straight and level unaccelerated flight, regulated cowl anti-ice pressures of 16 to 22 psi are considered normal.

B. **Cowl Anti-Ice Pressure Transmitters / Pressure Switches:**

Cowl anti-ice pressure is tapped at the cowl anti-ice valve and sampled by a pressure transmitter in the engine cowl. Receiving power from the Essential 28 VDC bus, the pressure transmitter in turn supplies a signal to the cowl anti-ice pressure indicator located in the ANTI ICE section of the cockpit overhead panel.

On airplanes SN 1000 through 1059 having ASC 51A and airplanes SN 1060 and subsequent, a pressure switch is installed in each engine cowl. If cowl anti-ice is selected ON and pressure drops below 10±1 psi, the pressure switch will cause an amber L-R COWL PRESS LOW caution message to be displayed on CAS. On airplanes SN 1000 through 1189 having ASC 243 and airplanes SN 1190 and subsequent, the trip point is 4±1 psi (after an approximately 15 second delay). When pressure increases above the trip point - which normally occurs when thrust is increased - the message automatically clears. If increasing thrust does not increase pressure above 4 psi, the flight crew should depart icing conditions.

C. Cowl Anti-Ice Thermal Switches:

A thermal switch is installed downstream of each engine's cowl anti-ice valve to monitor the temperature of bleed air entering the nose cowl ducts. If the bleed air temperature exceeds 662°F (350°C), the thermal switch will cause an amber L-R COWL A/I OVHT caution message to be displayed on CAS.

D. Nose Cowl Ducts and Piccolo Tubes:

From the cowl anti-ice valve, regulated engine bleed air flows through ducts to the nose cowl. Perforated tubes (called piccolo tubes because of their appearance) inside the nose cowl distribute the bleed air along the circumference of the nose cowl leading edge, heating the leading edge skin and preventing ice formation. The air is vented overboard through the engine air inlet.

3. Controls and Indications:

(See Figure 4 and Figure 5.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
L COWL ANTI-ICE	CP	J-13	ESS DC Bus
R COWL ANTI-ICE	CP	K-13	ESS DC Bus
L COWL A/I PRESS	CP	J-12	ESS DC Bus
R COWL A/I PRESS	CP	K-12	ESS DC Bus

B. Caution (Amber) CAS Messages:

CAS Message:	Cause or Meaning:
L-R COWL A/I OVHT	Engine cowl temperature is above 662°F (350°C).
L-R COWL PRESS LOW**(1)**	Cowl anti-ice pressure is less than 10±1 psi or 4±1 psi **(2)**.

NOTE(S):

(1) SN 1060 and subs; SN 1000-1059 having ASC 51A.

(2) SN 1190 and subs; SN 1000-1189 having ASC 243.

C. Advisory (Blue) CAS Messages:

CAS Message:	Cause or Meaning:
L-R COWL A/I ON	Cowl anti-ice is ON.

NOTE:

A description of the Engine Instruments and Crew Alerting System (EICAS) can be found in Section 5 of Honeywell's <u>SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV</u>.

4. Limitations:

A. Flight Manual Limitations:

(1) Requirements of Use:

Operation of the cowl anti-icing system is required for taxi and takeoff when Static Air Temperature (SAT) is below +8° C and visible moisture, precipitation, or wet runway are present. Engine operation at 85% LP for one (1) minute is recommended just prior to takeoff and at intervals of not more than sixty (60) minutes under these temperature and moisture conditions.

(2) Low Cowl Anti-Ice Pressure:

Depart icing conditions if COWL PRESSURE LOW message illuminates and cowl anti-ice pressure cannot be maintained at or above 4 psi.

NOTE:

The COWL PRESSURE LOW message illuminates at 4 psi on SN 1190 and Subs, and SN 1000 through 1189 with ASC 243. It illuminates at 10 psi on SN 1000 through 1189 without ASC 243. Cowl anti-ice is effective at 4 psi or greater.

(3) Use In Flight:

Use of cowl anti-icing system is required in flight as indicated in Figure 6: Temperature Range For Cowl Anti-Icing, when visible moisture or precipitation is present, or when signs of icing are observed. Ice accretion may be observed on wings or windshield edges.

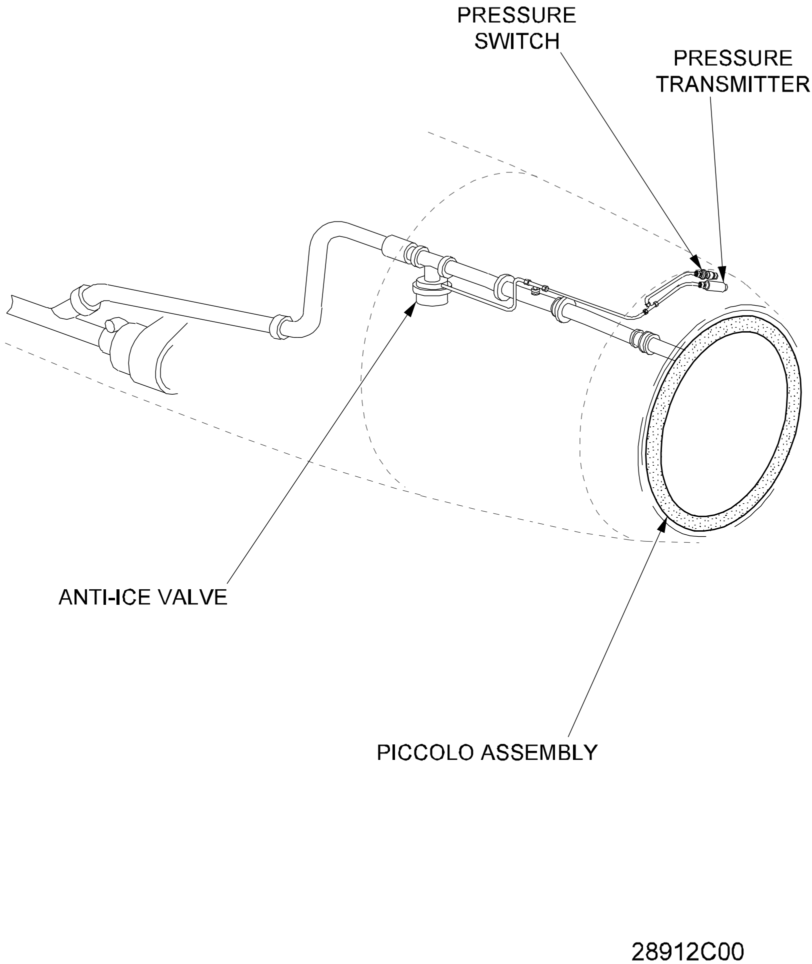
B. System Notes:

(1) Ice Formation During Idle Descents:

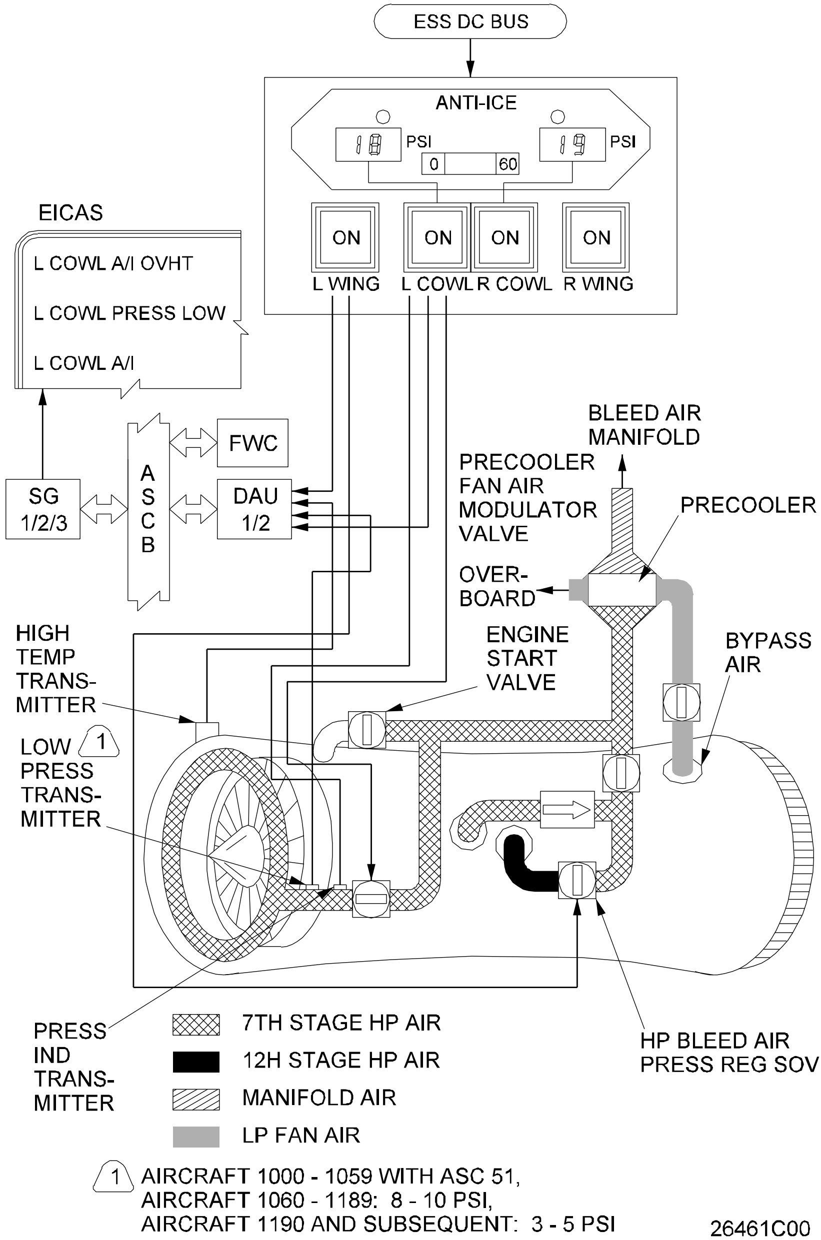
Although adequate bleed air for cowl anti-icing is available under all normal operating conditions, flight crews may note a limited amount of ice formation on the nose cowl during idle descents. When power is increased, some of this ice will be shed into the engine inlet. Shedding this limited amount of ice will not cause any adverse effects on engine operation.

(2) Icing and Engine Vibration:

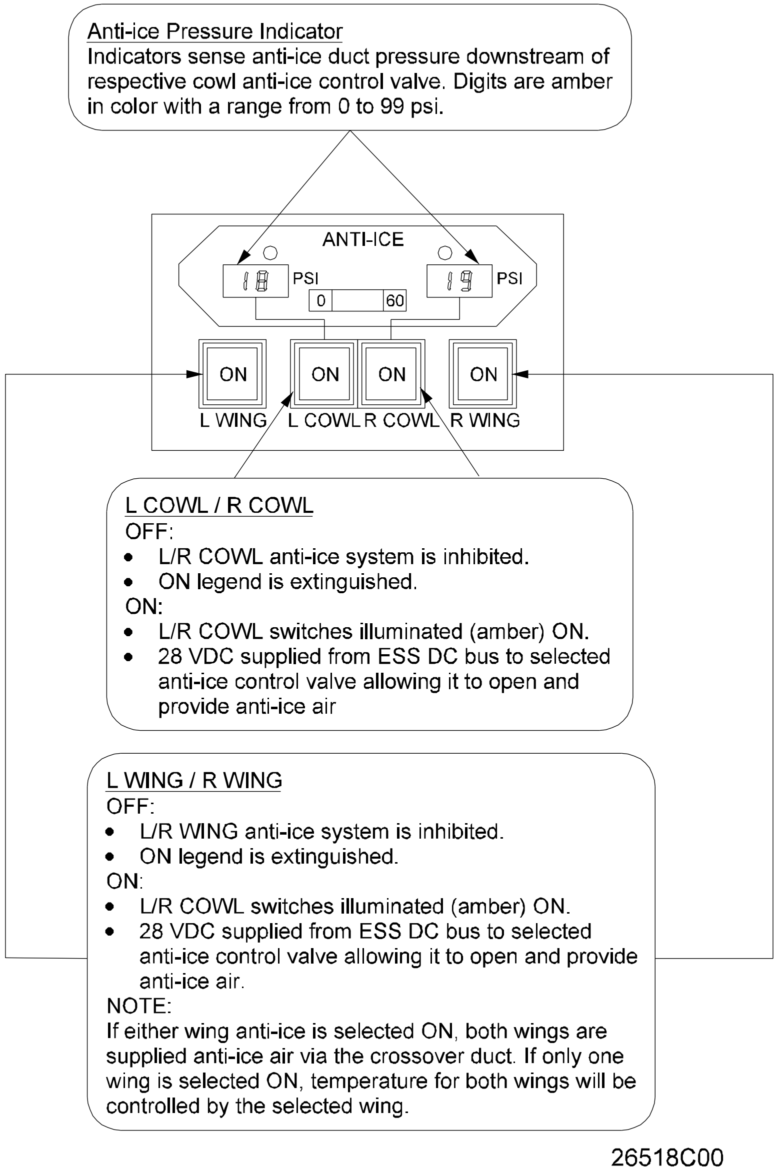
Increases in engine vibration level above 1.25 inches per second LP may develop in icing conditions. To assist in shedding ice, if sustained vibration is indicated and operational circumstances permit, retard one power lever at a time to IDLE for five (5) seconds and then restore to required thrust. Should the vibration still persist after doing so, momentarily adjust the thrust levels to 85% LP.



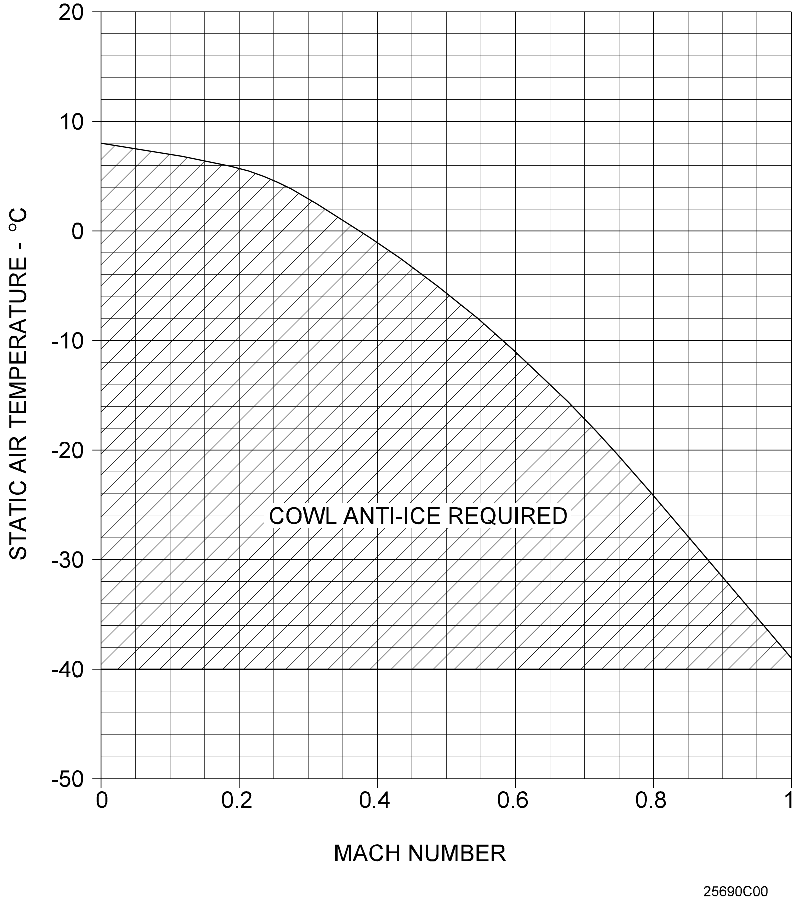
Cowl Anti-Ice System Layout
Figure 3



Cowl Anti-Ice System Simplified Block Diagram
Figure 4



Anti-Ice Control Panel
Figure 5



Temperature Range For Cowl Anti-Icing
Figure 6

2A-30-40: Windshield Ice and Rain Protection System

1. General Description:

The windshield ice and rain protection system provides deicing and defogging for the front windshields and side windows through the use of embedded electrical heating elements. Rain removal (front windshields only) is provided by a pair of electrically operated windshield wipers. The windshield ice and rain protection system is divided into the following subsystems:

- Windshield Heat System
- Windshield Wiper System

2. Description of Subsystems, Units and Components:

A. Windshield Heat System:

The windshield heat system automatically maintains front windshield and side window temperature for deicing and defogging. It is designed in such a manner that no single failure can cause total loss of windshield heat. It is not required for bird strike protection. The system consists of the following units and components:

- Heating Media
- Temperature Sensors
- Windshield Heat Controllers
- Windshield Heat Control Switch(es)
- Window Heat Advisory Lights

(1) Heating Media:

The left and right front windshields are constructed as laminates of three panes of glass and two interlayers. The outer pane is constructed of very thin but very strong glass. Should the glass be fractured, heating for that windshield will be lost. A conductive coating is applied on the inside surface of the outer pane. A bus bar at the top and bottom of each windshield distributes AC power from the associated windshield heat controller to the conductive coating. Two temperature sensors are also imbedded in each windshield.

The left and right side windows constructed as laminates of two acrylic layers with one interlayer and an Aircon heating mat sandwiched in between. A bus bar at the top and bottom of each window distributes AC power from the associated windshield heat controller to the heating mat. Two temperature sensors are also imbedded in each window.

(2) Temperature Sensors:

As stated above, two temperature sensors are imbedded in each windshield and window. They are normally referred to as the No. 1 and No. 2 sensors. Both sensors are connected to the windshield heat controller section serving that windshield/window. The windshield heat controllers use the No. 1 sensor for thermostatic control of the window. Should the No. 1 sensor circuit short or open, the windshield heat controllers switch to the No. 2 sensor with no interruption of service or notification to the flight crew. Should the No. 2 sensor then fail, the windshield heat controllers disable heat to that window and notify the flight crew by extinguishing the window heat advisory light.

On airplanes SN 1000 through 1155 having ASC 3 and ASC 275, SN 1156 through 1203 having ASC 275 and airplanes SN 1204 and subsequent, a temperature sensor test switch is incorporated. The switch is installed on the right-hand radio rack and is labeled W/S SENSOR TEST. With the windshield heat control switch(es) selected ON (OFF legend extinguished) and the TEST switch placed in either the SENSOR #1 or SENSOR #2 position, the associated sensor for each windshield and window is tested. Results of the test are obtained by observing the window heat

advisory lights. An illuminated advisory light confirms an operative sensor. If the advisory light does not illuminate, the light itself should be checked using the annunciator lights test switch. If the light tests satisfactorily, then a faulty sensor can be suspected and reported.

(3) Windshield Heat Controllers:

Two dual-unit heat controllers regulate AC power to the windshield and window heating media. The Left Front/Right Side (LF/RS) heat controller governs left front windshield and right side window heating; the Right Front/Left Side (RF/LS) heat controller governs right front windshield and left side window heating. Windshield/ window temperature is maintained at 90-100°F (32-38°C) by the controller. Sensor monitoring and switching is also performed by the heat controller.

Remote Control Circuit Breaker (RCCB) logic units are incorporated as connection and protection devices for each window. Should a window draw current in excess of a preset value, the RCCB will disconnect that window by causing the system circuit breaker to pop.

On airplanes SN 1183 and subsequent, the Electrical Load Warning System (ELWS) is capable of automatically shedding windshield heat, if ON, to protect APU generating capabilities. At or above 34,000 feet (pressure altitude sensed by the Air Data Computer), the ELWS processor will automatically shed windshield heat based on the bus being powered by the APU alternator, i.e., RF/LS heat controller if the APU alternator is powering the Left Main AC bus, LF/RS heat controller if the APU alternator is powering the Right Main AC bus. Once at or below 32,000 feet PA, windshield heat is automatically restored.

Power for control and heat is provided by Left and Right Main AC buses as shown in the table that follows. Power for indication is provided by the Essential 28 VDC bus.

Windshield/Window	Control Power	Operating Power	Voltage
Left Front	Right Main AC, ϕC	Right Main AC, ϕB Right Main AC, ϕC	208 VAC
Right Side	Left Main AC, ϕA	Right Main AC, ϕC	115 VAC
Right Front	Right Main AC, ϕC	Left Main AC, ϕA Left Main AC, ϕB	208 VAC
Left Side	Left Main AC, ϕA	Left Main AC, ϕA	115 VAC

(4) Windshield Heat Control Switch(es):

(See Figure 7.)

For airplanes SN 1000 through 1095 (except SN 1001) not having ASC 239: A single WSHLD switchlight (cockpit overhead panel, ANTI ICE HTR section) controls power to both windshield heat controllers through two heat control relays: the left front/right side relay and the right front/left side relay. Selection to ON energizes the relays and extinguishes the amber OFF legend in the switchlight. Once energized, the relays supply power to their associated heat controller.

For airplanes SN 1001, SN 1096 and subsequent, SN 1000 through 1095 having ASC 239, and CAA certified aircraft: Two WINDSHIELD switchlights, LF/RS and RF/LS (cockpit overhead panel, ANTI ICE HTR section), control power to their associated windshield heat controller through four heat control relays: left side, left front, right front and right side. Selection to ON energizes the associated relays, extinguishes the amber OFF legend in the switchlight and supplies power to the associated heat controller.

(5) Window Heat Advisory Lights:

(See Figure 8.)

With windshield heat selected ON and the heat controllers supplying power to the windshield/window heating media, the LEFT FRONT, LEFT SIDE, RIGHT FRONT and RIGHT SIDE WINDOW annunciators (cockpit overhead panel) illuminate green to signal that heat is being applied to the window. The annunciators extinguish when the associated window reaches its normal operating temperature, and continue to cycle between illuminated and extinguished as heating power is applied and removed from the windshield/window heating media.

Each annunciator capsule contains four bulbs. Receiving power from the Essential 28 VDC bus, they are tested and controlled using the airplane's annunciator lights dim and test function.

B. Windshield Wiper System:

A pair of electrical windshield wipers remove precipitation from the left and right windshields. Being independent of each other, the left windshield wiper system receives power from the Essential 28 VDC bus; the right from the Right Main 28 VDC bus. Each system consists of the following units and components:

- Windshield Wiper Control Unit
- WIPER and LO/HI Control Switch
- Windshield Wiper Motor/Arm/Blade Assembly

(1) Windshield Wiper Control Unit:

The left and right windshield wiper control units each contain two relays that control wiper motor operation and speed in conjunction with the LO/HI control switch. A noise filter is incorporated to reduce any radio interference generated by wiper motor operation.

(2) WIPER and LO/HI Control Switch:

(See Figure 9.)

Each windshield wiper's operation and speed is controlled by a dedicated WIPER switch and associated LO/HI switch (cockpit overhead panel, WINDSHIELD WIPER section). The WIPER switch controls 28 VDC power directly to one side of the associated wiper motor and indirectly through the associated LO/HI switch and control unit relays to the motor. Selection of the WIPER switch to ON causes the associated wiper to operate at the speed determined by the associated LO/HI switch position. The ON legend illuminates blue.

With a wiper selected ON, selection of the WIPER switch a second

time extinguishes the ON legend in the switch and reverses the wiper motor, moving the wiper arm/blade assembly to the park position. Once in the park position, power is removed from the motor circuit.

The LO/HI switch controls a pair of relays within the associated control unit that manipulate the associated motor circuitry to achieve the desired wiper speed. When high speed is selected, the HI legend in the switch is illuminated blue.

(3) Windshield Wiper Motor/Arm/Blade Assembly:

The left and right windshield wiper motor/arm/blade assemblies are mechanically identical, but mirror images of each other and for the most part are not interchangeable. The left and right wiper blades, however, are identical.

When the wiper reaches the park position, a park switch in the motor breaks the motor power circuit and energizes an electrically actuated friction-type brake to prevent further motor movement. Should the wiper motor experience an overload, an internal thermo-protector temporarily de-energizes the motor. When the motor has sufficiently cooled, the thermo-protector automatically resets.

3. Controls and Indications:

(See Figure 7 through Figure 9.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
L FRONT PWR **(1)**	CP	J-8	R MAIN AC Bus, ϕB
R SIDE PWR **(2)**	CP	K-10	R MAIN AC Bus, ϕC
L SIDE PWR **(3)**	CP	J-10	L MAIN AC Bus, ϕA
R FRONT PWR **(4)**	CP	L-8	L MAIN AC Bus, ϕB
L WSHLD WIPER	CP	J-11	ESS DC Bus
R WSHLD WIPER	CP	J-12	R MAIN DC Bus

NOTE(S):

(1) SN 1000-1095, excluding SN 1001 & 1034. For SN 1001, 1034 and SN 1096 & subs, CB is labeled L FRONT WSHLD.

(2) SN 1000-1071, excluding SN 1001 & 1034. For SN 1001, 1034 and SN 1072 & subs, CB is labeled R SIDE WSHLD.

(3) SN 1000-1071, excluding SN 1001 & 1034. For SN 1001, 1034 and SN 1072 & subs, CB is labeled L SIDE WSHLD.

(4) SN 1000-1095, excluding SN 1001 & 1034. For SN 1001, 1034 and SN 1096 & subs, CB is labeled R FRONT WSHLD.

4. Limitations:

A. Flight Manual Limitations:

There are no Flight Manual limitations for the windshield ice and rain protection system at the time of this revision.

B. System Notes:

(1) Care of Acrylic Surface of Cockpit Side Windows:

Exercise caution to guard against scratching the acrylic surface of the cockpit side windows when handling items normally adjacent to the windows, such as sunscreens and oxygen masks.

(2) Cabin Window Defogging:

Cabin window defogging is accomplished by passively drying the cabin air moving in and out of the space between the inner and outer panes of each cabin window. Pressure variances within the cabin resulting from normal airplane operation force cabin air through two desiccant dehydrators when entering and exiting the window enclosure.

(3) Windshield Wiper Operation:

The windshield wipers are effective at airspeeds of up to 200 KCAS. The windshield wipers should never be used when windshields are dry.

(4) Windshield Heat Operation:

On airplanes SN 1000 through 1273 not having ASC 126, electrical power transients may cause the windshield heat controller to latch in the full power OFF or full power ON modes. To prevent windshield damage after electrical power transients, select windshield heat OFF for two (2) seconds, then back ON. This resets windshield heat controller logic. On airplanes SN 1274 and subsequent and SN 1000 through 1273 having ASC 126, a one second time delay relay is installed to ensure the windshield heat controller has adequate time to reset itself, thus cycling power OFF and ON is not necessary.

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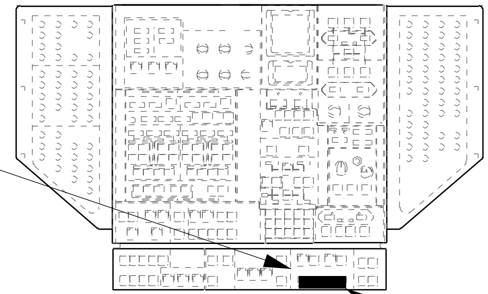
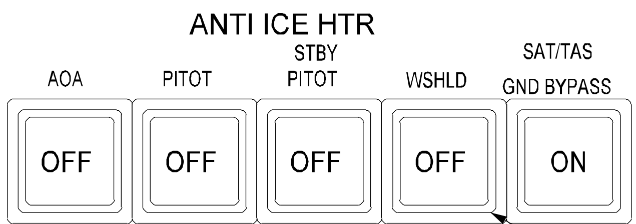
SEE DETAIL A



SEE DETAIL B

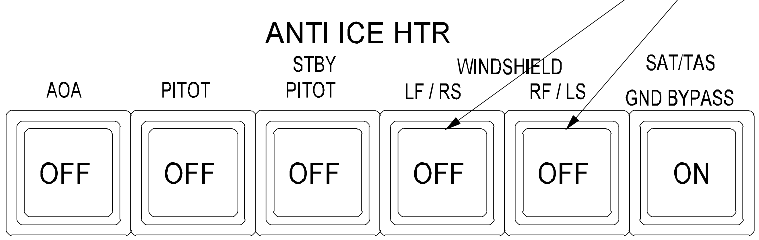


DETAIL A

Aircraft SN 1000 thru 1095 except 1001 without ASC 239.

WSHLD / WINDSHIELD
ON:
- Amber OFF legend is extinguished.
- Windshield heat operates as determined by the windshield heat control unit.

OFF:
- Amber OFF legend is illuminated.
- Windshield heat is inhibited.



DETAIL B

Aircraft SN 1000 thru 1095 with ASC 239 and 1001, CAA certified aircraft and 1096 and sub.

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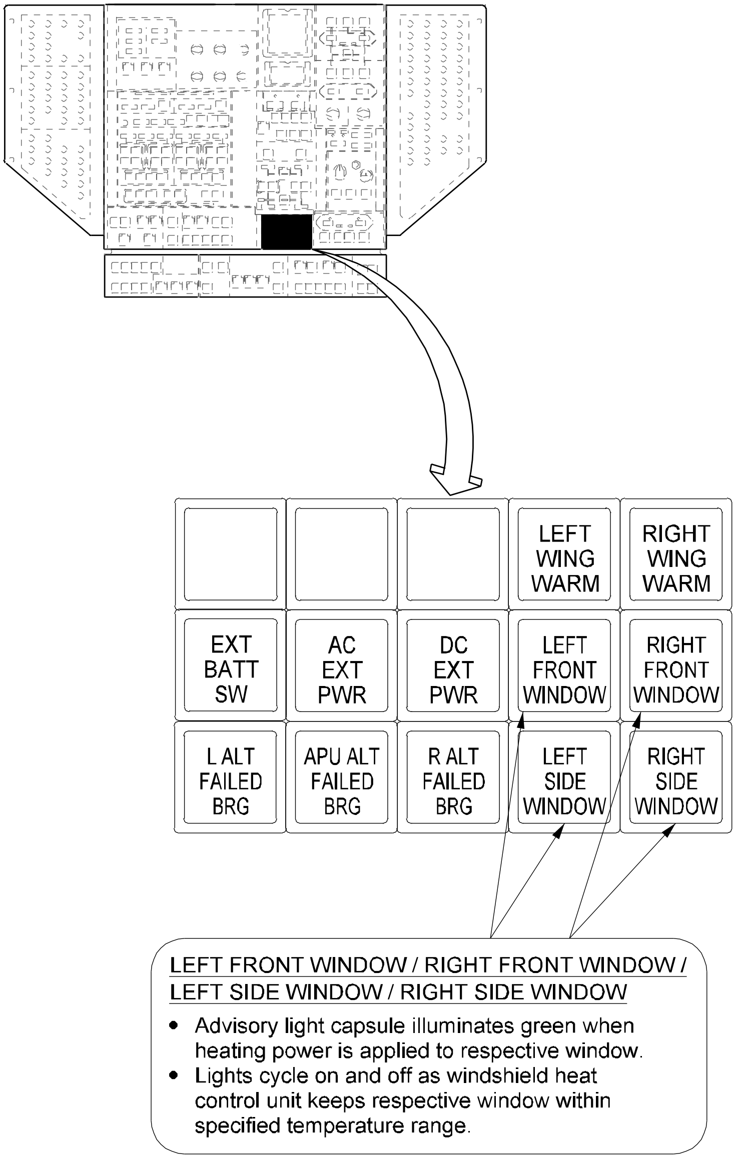
Windshield Heat Controls and Indications
Figure 7



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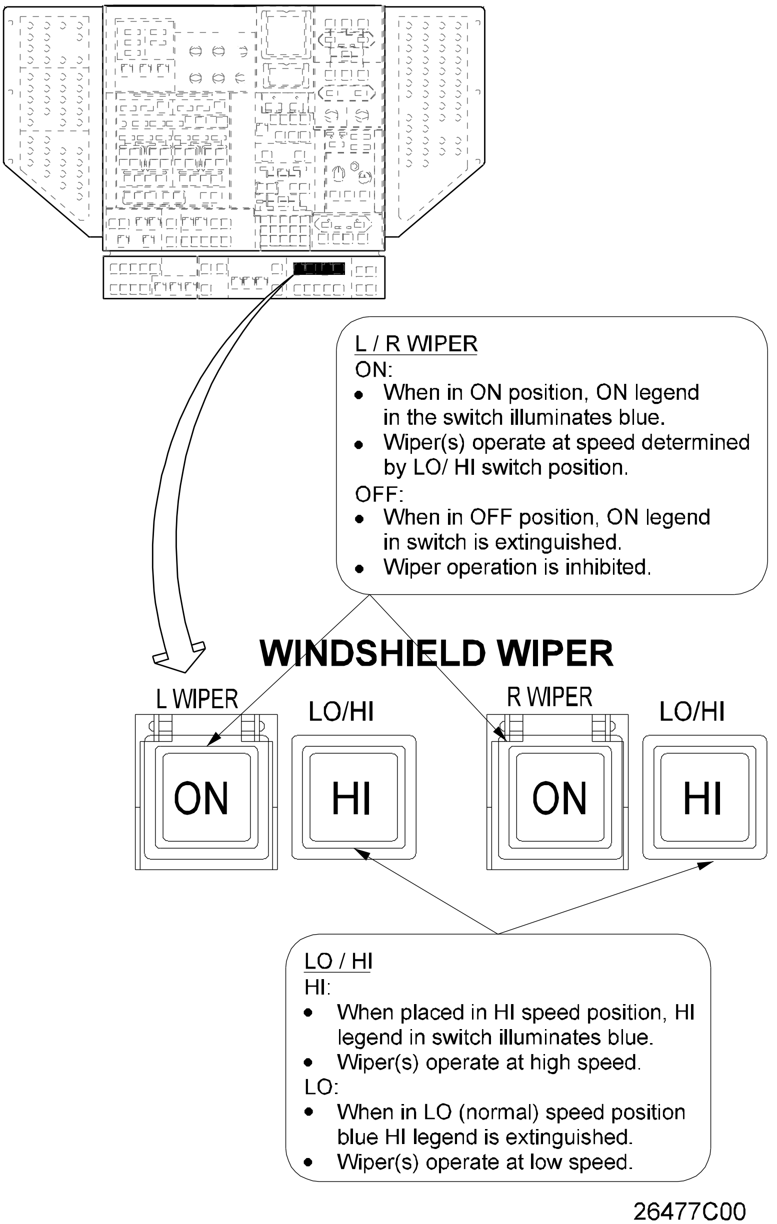
Window Heat Advisory Lights
Figure 8



Windshield Wiper System Controls and Indications
Figure 9

2A-30-50: Probe Anti-Ice System

1. General Description:

The probe anti-ice system uses electrically powered heating elements to protect the pitot probes, Angle of Attack (AOA) probes and Total Air Temperature (TAT) probe from ice accumulation.

2. Description of Subsystems, Units, and Components:

A. Pitot Probe Heat System:

The pitot probes are located on the nose of the airplane, just forward of the windshield. A third, standby pitot tube is located on the left forward fuselage.

The pitot probe heat system uses the PITOT and STBY PITOT switches (cockpit overhead panel, ANTI ICE section) and power relays to control AC electrical power to its resistance-type heating elements. Associated current sensors monitor current flow through the heating elements, cautioning the flight crew about out of range conditions through L-R PITOT HT FAIL and STBY PITOT HT FAIL caution Crew Alerting System (CAS) messages.

In addition to the CAS messages, the OFF legend in the PITOT switch plays a role in notifying the flight crew of pitot probe heat failures. On airplanes SN 1000 through 1143 not having Aircraft Service Change (ASC) 187, the OFF legend will illuminate if either the PITOT switch is selected OFF or if a **single or dual** heating element failure occurs. On airplanes SN 1144 and subsequent and SN 1000 through 1143 having ASC 187, the OFF legend will illuminate if either the PITOT switch is selected OFF or if a **dual** heating element failure occurs.

Selection of the PITOT switch to ON supplies 28 VDC power to energize the left and right pitot probe heat relays. Once energized, the relays extinguish the OFF legend in the switch and route single-phase 115 VAC power to the relay's associated heating element. (See Controls and Indications for specific control and operating power sources.) When the current sensors sense normal current flow through the heating elements, the associated L-R PITOT HT FAIL caution CAS messages are cleared.

Selection of the STBY PITOT switch to ON supplies Essential 28 VDC bus power to energize the standby pitot probe heat relay. Once energized, the relay extinguishes the OFF legend in the switch and routes Essential 115 VAC bus power to the heating element. When the current sensor senses normal current flow through the heating element, the STBY PITOT HT FAIL caution CAS message is cleared.

Airplanes having ASC 10 have a pitot heat inhibit relay installed that energizes when the outside battery switch is on the ON position. When energized, the relay opens the circuit between the PITOT and STBY PITOT switches and their associated probe heat relays.

B. Angle of Attack (AOA) Probe Heat System:

The AOA probes are located on either side of the forward fuselage.

The AOA probe heat system uses both a case heater and probe heater controlled by the AOA switch (cockpit overhead panel, ANTI ICE section) through two probe heat relays. Two thermal switches are also incorporated: one for the case to maintain a moisture-free environment within the case and one for the probe to prevent overheating.

Selection of the AOA switch to ON supplies Right Main and Essential 28 VDC bus power through the PITOT HT circuit breakers to the No. 1 and No. 2 probe heat relays. Once energized, the relay routes Right Main and Essential 28 VDC bus power to the case and probe heating elements. The OFF legend in the switch is then extinguished and the AOA HEAT 1-2 FAIL caution CAS message is cleared.

The case heater thermal switch cycles the case heating element on and off to maintain a case temperature of 90-125° F (32-52° C). By cycling heater power, condensation formation within the case and its resultant effect on probe operation are avoided.

The probe thermal switch prevents overheating by inhibiting heating element power at 280° F (138° C). When probe temperature drops below the trip point, power to the heating element is restored.

C. Total Air Temperature (TAT) Probe Heat System:

The TAT probe is located on the lower right side of the forward fuselage below the AOA probe.

The TAT probe heat system is a shared function of the pitot probe heat system. When in flight, selection of the PITOT switch to ON supplies Right Main 28 VDC bus power through the TAT probe heat relay through nutcracker relay No. 8. Once energized, the TAT probe heat relay routes single-phase Right Main 115 VAC bus power to the TAT probe heating element. When the current sensor senses normal current flow through the heating element, the TAT HT FAIL caution CAS message is cleared.

When the airplane is on the ground, selection of the SAT/TAS GND BYPASS switch (cockpit overhead panel, ANTI ICE section) to ON (ON legend illuminated) bypasses the nutcracker relay to allow TAT probe heating.

To ensure accurate calibration readings, air is introduced through the TAT probe when the airplane is on the ground. Through this process, known as aspiration, bleed air is supplied to the probe from a shutoff valve that opens when the nutcracker shifts to the ground mode. This shutoff valve, known as the “total temp valve”, receives its power from the Right Main 28 VDC bus.

3. Controls and Indications:

(See Figure 10.)

A. Power Sources:

Probe:	Switch:	Control Power:	Operating Power:
Left Pitot	PITOT	ESS DC Bus	ESS AC Bus ϕA
Right Pitot	PITOT	R MAIN DC Bus	R MAIN AC Bus ϕA
TAT	PITOT	R MAIN DC Bus	R MAIN AC Bus ϕB
Standby Pitot	STBY PITOT	ESS DC Bus	R MAIN AC Bus ϕA
Left AOA	AOA	ESS DC Bus	ESS DC Bus
Right AOA	AOA	R MAIN DC Bus	R MAIN DC Bus

B. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
AOA PRB HTR #1	CP	L-14	ESS DC Bus
AOA PRB HTR #2	CP	M-14	R MAIN DC Bus
STBY PITOT HT CONT	CP	L-13	ESS DC Bus
STBY PITOT HT PWR	CP	M-13	R MAIN AC Bus ϕB
L PITOT HT CONT	CP	L-12	ESS DC Bus
R PITOT HT CONT	CP	M-12	R MAIN DC Bus
L PITOT HT PWR	CP	L-11	ESS AC ϕA
R PITOT HT PWR	CP	M-11	R MAIN AC Bus ϕA
TOTAL TEMP PROBE HTR	CP	L-10	R MAIN AC Bus ϕB
TOTAL TEMP VALVE	CP	M-10	R MAIN DC Bus

C. Caution (Amber) CAS Messages:

CAS Message:	Cause or Meaning:
L-R PITOT HT FAIL	Pitot tube heater elements not energized.
STBY PITOT HT FAIL	Standby pitot heater element not energized.
AOA HEAT 1-2 FAIL	Angle-of-attack probe heater has failed.
TAT PROBE HT FAIL	Total Air Temperature probe heater has failed.

4. Limitations:

A. Flight Manual Limitations:

There are no Flight Manual limitations for probe anti-ice system at the time of this revision.

B. System Notes:

(1) Loss of AOA Probe Heat In Flight:

If an AOA probe heater fails in flight and the associated CB is of no assistance, it is recommended that icing conditions be avoided, if possible. If flight into icing conditions is unavoidable, it is recommended that the stall barrier computer with the associated failed AOA probe heater be disabled through its associated CB and the operative system be monitored. Full stall barrier protection will still be provided by the operative system.

(2) Loss of Pitot Probe Heat In Flight:

If a pitot probe heater fails in flight, it is recommended that the flight crew be alert for erroneous indications from the airspeed indicator, air data system computer, airspeed warning sensor and Mach trim compensation.

(3) Loss of TAT Probe Heat In Flight:

If a TAT probe heater fails in flight and the associated CB is of no assistance, the flight crew should be aware that TAT inputs to the DADCs may be inaccurate which, in turn, may affect DADC outputs to the FMS. It is therefore recommended that the flight crew be alert for inaccurate readouts from the FMS.

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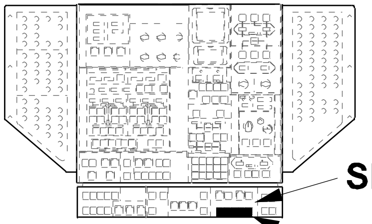
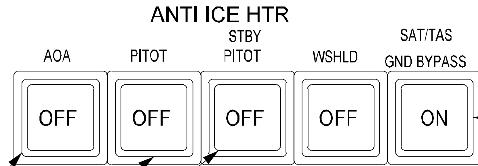




DETAIL A
Aircraft SN 1000 thru 1095 except 1001 without ASC 239

AOA
OFF:
- Amber OFF legend is illuminated.
- Heating power to probe and probe case is inhibited.

ON:
- Amber OFF legend is extinguished.
- Power is provided to left probe and probe case by Essential 28 VDC bus.
- Power is provided to right probe and probe case by Right Main 28 VDC bus.

STBY PITOT
OFF:
- Amber OFF legend is illuminated.
- Heating power to probe heating element is inhibited.

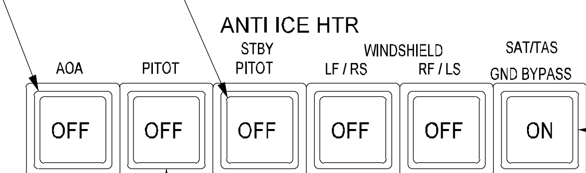
ON:
- Amber OFF legend is extinguished.
- Standby pitot heat power is provided by Right Main 115 VAC bus Ø B.

SAT/TAS GND BYPASS
OFF:
- Amber ON legend is extinguished.
- Power is provided through pitot heat system (while in flight only with pitot heat selected).

ON:
- Amber ON legend is illuminated.
- The nutcracker switch is bypassed and heating power is provided to the TAT probe.

PITOT
OFF:
- Amber OFF legend is illuminated.
- Heating power to probe heating element is inhibited.

ON:
- Amber OFF legend is extinguished.
- Left pitot heat power is provided by Essential 115 VAC bus $ A.
- Right pitot heat power is provided by Right Main 115 VAC bus $ A.



DETAIL B
Aircraft SN 1000 thru 1095 with ASC 239 and 1001, CAA certified aircraft and 1096 and sub.

26542C01

ANTI-ICE HTR Control Panel
Figure 10

2A-31-10: General

The Gulfstream IV indicating / recording system provides the flight crew with positive control and visible operational indications of integrated aircraft systems. It also provides independent, accurate time displays necessary for any timed flight maneuvers. Visible and aural annunciations are presented to the flight crew in the form of messages and tones categorized as warning, caution and advisory. The system also receives, processes, records and displays data from associated aircraft systems.

The indicating / recording system is divided into the following subsystems:

- 2A-31-20, Cockpit Clocks System
- 2A-31-30, Flight Data Recording System
- 2A-31-40, Central Warning System
- 2A-31-50, Central Display System

2A-31-20: Cockpit Clocks System

1. General Description:

The cockpit clocks provide accurate time displays necessary for any timed flight maneuvers. This includes emergency turn coordination when operating on standby instruments.

2. Description of Subsystems, Units, and Components:

There are two cockpit clocks: #1 and #2 (commonly referred to as pilot's and copilot's), located on the left and right side of the flight panel, respectively. They are capable of displaying the following types of time:

- Greenwich Mean Time (GMT)
- Local Time (LT)
- Flight Time (FT)
- Elapsed Time (ET)

Each clock has a two inch digital display with a microprocessor-controlled chronometer and incandescent lighting. Conventional time is displayed in hours and minutes. Chronometer time is displayed in minutes and seconds. Both types of time are displayed in decimal form.

Each clock continues to receive power in all electrical power configurations down to and including the Emergency 28 VDC bus. Internal batteries are also installed to ensure operation in the event of a total loss of aircraft power. When fully charged, the internal batteries can power the clocks for up to 30 days.

3. Controls and Indications:

(See Figure 1.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
CLOCK #1	CP	H-14	EMER DC Bus 1A
CLOCK #2	CP	I-14	EMER DC Bus 2C

4. Limitations:

There are no Flight Manual limitations established for cockpit clocks system at the time of this revision.

TO SET GMT/LT/ET:

1. Depress SELECT key until desired time mode is displayed.
2. Depress SELECT and CONTROL keys simultaneously until digit in left-most column begins to flash.
3. Depress CONTROL key until desired digit is displayed in selected column.
4. Depress SELECT key to move to the next column. Repeat steps 3 & 4 as necessary.
5. Depress SELECT key after right-most digit column is set to return to normal function.

TO RESET FT:

1. Depress SELECT key until FT mode is displayed.
2. Depress and hold CONTROL key until FT displays 99:59, then release CONTROL key.

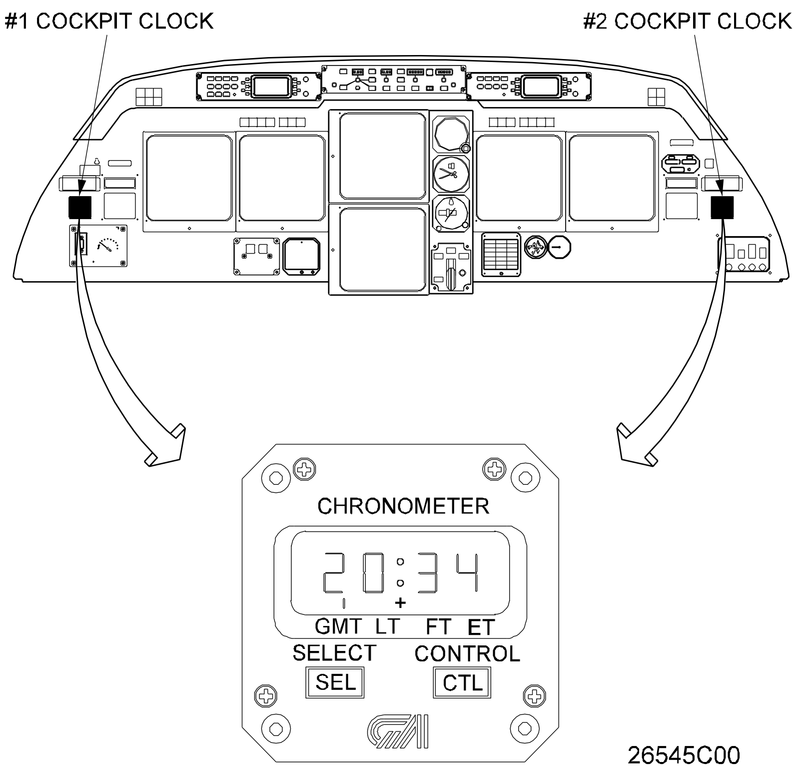


Figure 1. Cockpit Clocks (Typical)

2A-31-30: Flight Data Recording System

The Gulfstream IV flight data recording system provides a means to receive, process and store airplane operational information. The stored information is then protected so that it may later be retrieved for historical, investigative or decision-making purposes.

The flight data recording system is described in the following subsections within this section:

- 2A-31-31: Flight Data Recorder System (airplanes Serial Number [SN] 1183 and subsequent)
- 2A-31-32: 'G' Monitor System (airplanes SN 1034, 1156 and subsequent and airplanes SN 1000 through 1155 with Aircraft Service Change [ASC] 118)

2A-31-31: Flight Data Recorder System

1. **General Description:**

   On airplanes SN 1183 and subsequent, a solid-state digital flight data recorder system is incorporated to receive, process and record parameters relating to airplane operation and store the data in a protected environment.

   The Avionics Standard Communications Bus (ASCB) is the primary source of airplane flight data for the flight data recorder system. Parameter data generated by airplane systems or transducers (sensors) is transmitted to the recorder system in digital ARINC 429 or analog/discrete signal format. The data is converted to a common digital format for recording. All data acquisition and format conversions are accomplished within the flight data acquisition unit, which then transmits reformatted data to the flight data recorder, where it is stored until erased.

   The flight data recorder system consists of the following units and components:

   - Flight Data Acquisition Unit
   - Flight Data Recorder
   - Underwater Locating Beacon
   - Flight Data Recorder Impact Switch
   - Flight Data Recorder Maintenance Ground Override Switch
   - Flight Data Entry Panel
   - Flight Data Recorder Switch Panel

2. **Description of Subsystems, Units and Components:**

   A. **Flight Data Acquisition Unit:**

      The flight data acquisition unit, located in the right hand radio rack, accepts flight data information sent from the sensors listed below. It then processes the data and sends it to the flight data recorder for recording. Sensors providing input to the flight data acquisition unit are the:

      - Triaxial accelerometer
      - Left and right spoiler position sensors
      - Left and right aileron position sensors
      - Elevator position sensor
      - Elevator trim tab position sensor
      - Rudder position sensor

- Flaps position sensor

There are three fault indicator lights on the front panel: CAUTION, DFDR FAULT and FDAU FAULT. (See Figure 2.) The CAUTION indicator light is not functional for the GIV flight data recorder system. The DFDR FAULT indicator light is triggered by a malfunction in the flight data recorder circuitry. The FDAU FAULT indicator light is triggered by the flight data acquisition unit built-in test circuitry detecting a fault in the flight data acquisition unit. If either DFDR FAULT or FDAU FAULT indicator light comes on, it will be necessary to replace the flight data acquisition unit.

B. Flight Data Recorder:

The flight data recorder (Figure 3) records and stores digitized ASCB data. Data can be retrieved to assist in reconstruction of events relevant to an incident. The flight data recorder continuously records and retains data for a minimum of the previous 25 hours of operation. With electrical power removed, recorded data is retained for a minimum of two years. The flight data recorder is located in the aft equipment (tail) compartment. It is international orange in color for high visibility to aid in location.

ASC 423/423A is available for airplanes SN 1421 and subsequent requiring FAR Part 135 certification. This ASC upgrades the digital flight data recorder, allowing the recording of 57 parameters, in compliance with FAR Part 135.

ASC 428 is available for airplanes SN 1485 and subsequent requiring FAR Part 135 certification. This ASC upgrades the digital flight data recorder, allowing the recording of 88 parameters.

C. Underwater Locating Beacon:

An underwater locating beacon (Figure 3) is attached to the flight data recorder. When activated by contact with water, it transmits a traceable acoustic signal to aid in underwater location of the airplane and/or flight data recorder. The underwater locating beacon receives power from a self-contained battery.

D. Flight Data Recorder Impact Switch:

The flight data recorder impact switch (Figure 4) removes power from the flight data recorder upon impact. Removing power prevents pertinent data from being erased from the flight data recorder. The impact switch is mounted in the tail compartment's right side, just outboard of the flight data recorder. Activation of the impact switch also causes an annunciator light on the switch housing to illuminate, remaining illuminated until the impact switch is reset. Resetting the impact switch is accomplished through the use of a reset switch, also located on the impact switch housing.

E. Flight Data Recorder Maintenance Ground Override Switch:

The flight data recorder maintenance ground override switch (Figure 5) is located in the top of the right hand radio rack and is labeled FDR MAINT GRD OVRD.

During normal operations, the flight data recorder begins recording during engine start when sensed oil pressure rises to 10 psi or greater on either engine. Thus, there exists the possibility that recorded data could be overwritten during extended ground engine runs. If the aircraft is stationary and the Weight-On-Wheels (WOW) system is in the GROUND mode,

placing the FDR MAINT GRD OVRD switch in the OFF position allows maintenance personnel to inhibit flight data recording while aircraft engines are operating.

If the switch is inadvertently left in the OFF position, transfer of WOW to the AIR mode (such as during takeoff) automatically returns the switch to the AUTO position, enabling flight data recording. Manual positioning of the switch to the AUTO position is also possible.

F. Flight Data Entry Panel:

The flight data entry panel (Figure 6) is located in the copilot's side console. It is used by the flight crew to enter a four digit flight number or leg number by positioning its four FLIGHT NO. thumbwheels. Marking of events in the recorder's memory is conducted using the EVENT switch. Faults related to the flight data recorder and flight data acquisition unit are annunciated by the amber DFDR and FDAU lights, respectively.

G. Flight Data Recorder Switch Panel:

The flight data recorder switch panel (Figure 7) is located adjacent to the flight data entry panel. The panel has a FDR MANUAL ON/OFF toggle switch and a blue indicator light.

The FDR MANUAL switch is normally left in the OFF position. Selection to ON hard-selects the flight data recorder to ON, permitting recording prior to engine start-up (i.e., prior to acquiring engine oil pressure). Selection to ON will also return the radio rack FDR MAINT GND OVRD switch to AUTO, if the OVRD switch was selected to OFF. The FDR MANUAL switch is normally selected to OFF after engine start but, if left ON, will recycle automatically to OFF after the airplane has landed to prevent being inadvertently left ON. Whenever the FDR MANUAL switch is ON, the blue indicator light is illuminated.

3. Controls and Indications:

(See Figure 2 through Figure 7.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
FDR/FDAU	CP	C-7	ESS AC Bus
FDR CONT #1	CP	C-8	ESS DC Bus
FDR CONT #2	CP	C-9	BATT #1 Bus
POSN SNSRS	CP	C-10	#1 26 VAC XFMR Bus

B. Advisory (Blue) CAS Messages:

CAS Message:	Cause or Meaning:
FLIGHT REC FAIL	Flight data recorder has failed. It is normal for this message to be displayed until after an engine is started and 10 psi oil pressure is obtained.

C. Other Advisory Annunciations:

Annunciation:	Cause or Meaning:
ON annunciator illuminated adjacent to FDR MANUAL switch (copilot's side console).	FDR MANUAL switch is ON.
Flight data recorder impact switch annunciator light illuminated.	Flight data recorder impact switch has been activated.
FDAU annunciator illuminated on flight data entry panel.	Flight data acquisition unit fault or failure detected.
DFDR annunciator illuminated on flight data entry panel.	Flight data recorder fault or failure detected.
FDAU fault indicator illuminated on flight data acquisition unit.	Flight data acquisition unit fault or failure detected.
DFDR fault indicator illuminated on flight data acquisition unit.	Flight data recorder fault or failure detected.

4. Operation:

Other than setting the appropriate flight number or leg number on the flight data entry panel, operation of the flight data recorder system is automatic. On power up, the flight crew should expect a blue FLIGHT REC FAIL advisory message to be displayed on CAS and an amber DFDR annunciator illuminated on the flight data entry panel. The CAS message will be cleared and the annunciator will extinguish when oil pressure increases to 10 psi or greater in either engine. Selection of the FDR MANUAL switch to ON will also clear the CAS message and extinguish the annunciator.

5. Limitations:

There are no Flight Manual limitations established for the flight data recorder system at the time of this revision.

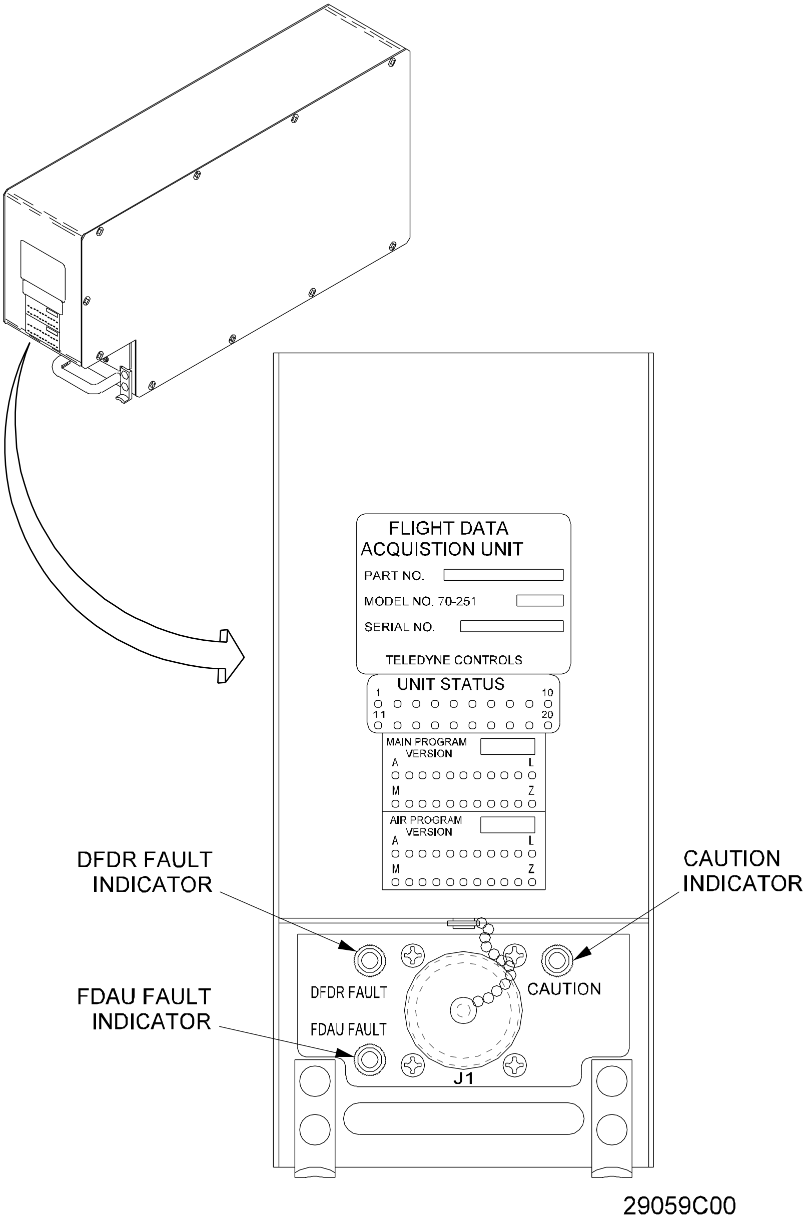


Figure 2. Flight Data Acquisition Unit Indicators

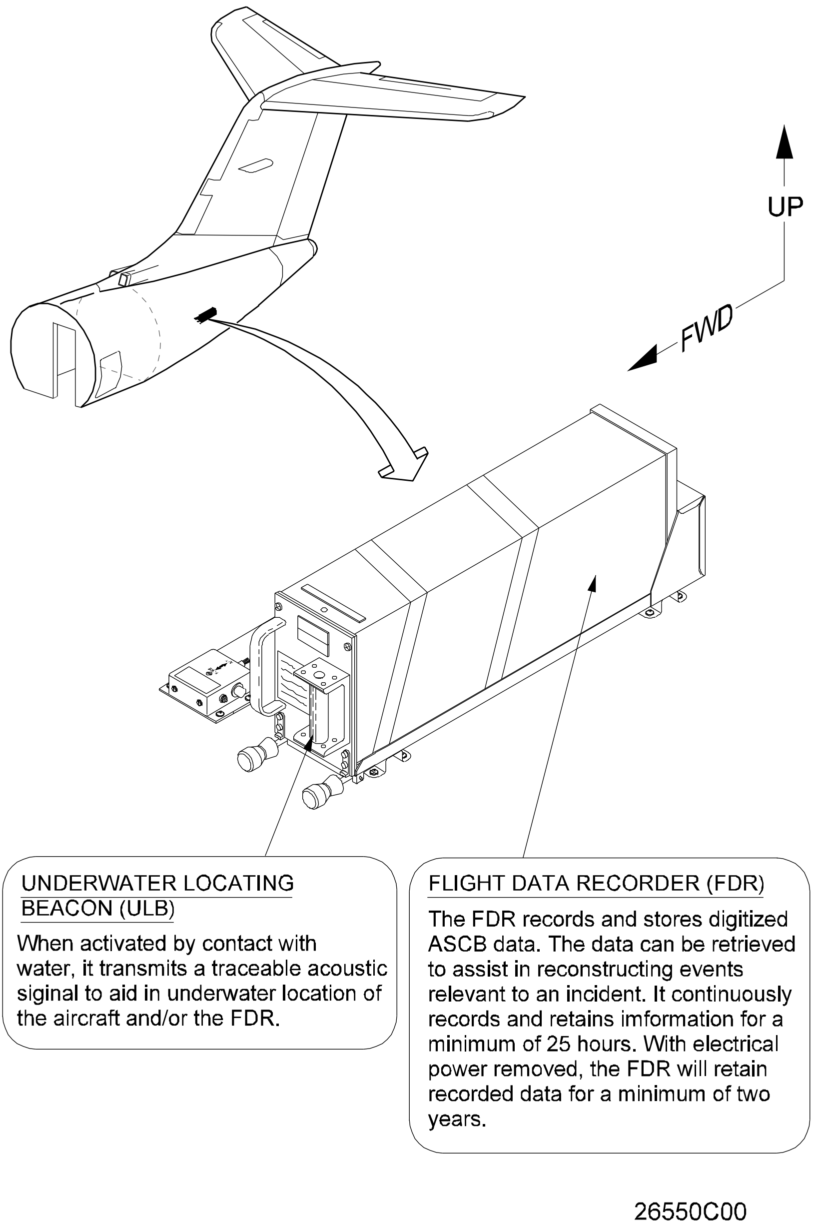


Figure 3. Flight Data Recorder With Underwater Locating Beacon

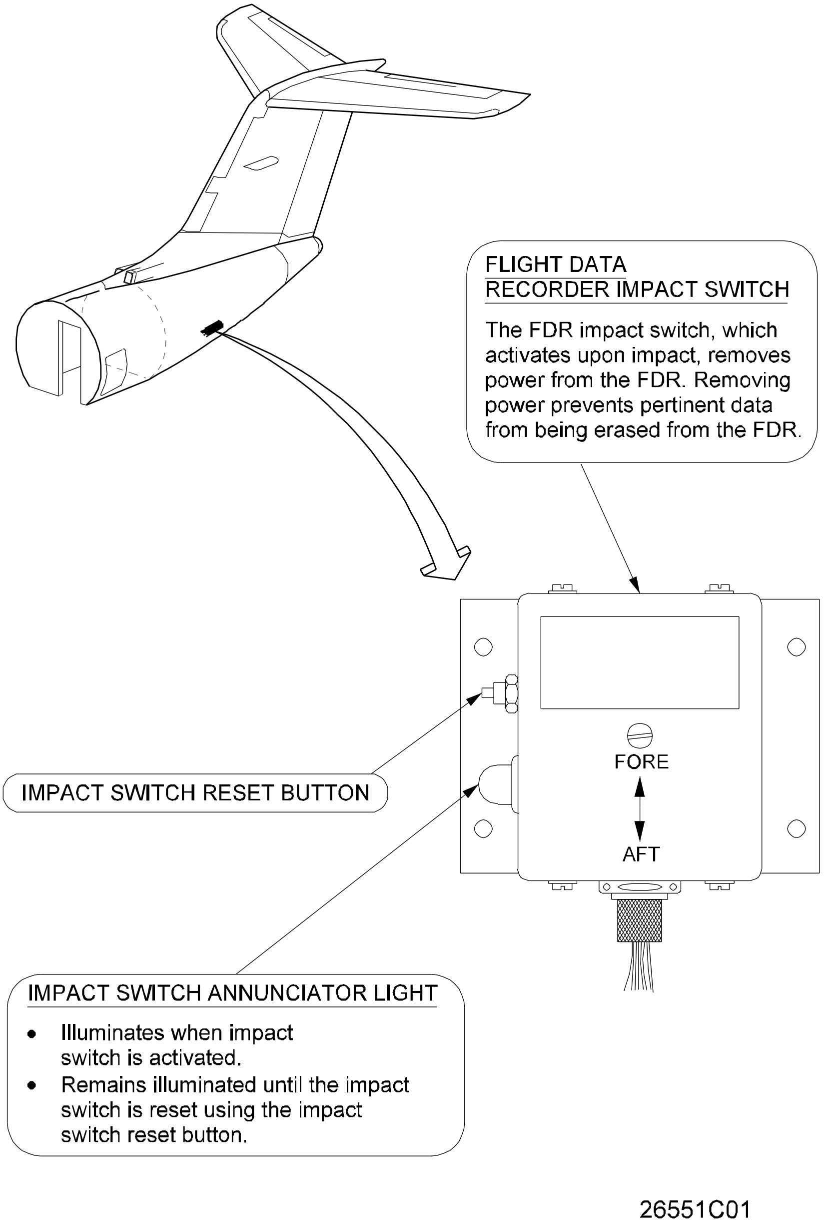


Figure 4. Flight Data Recorder Impact Switch

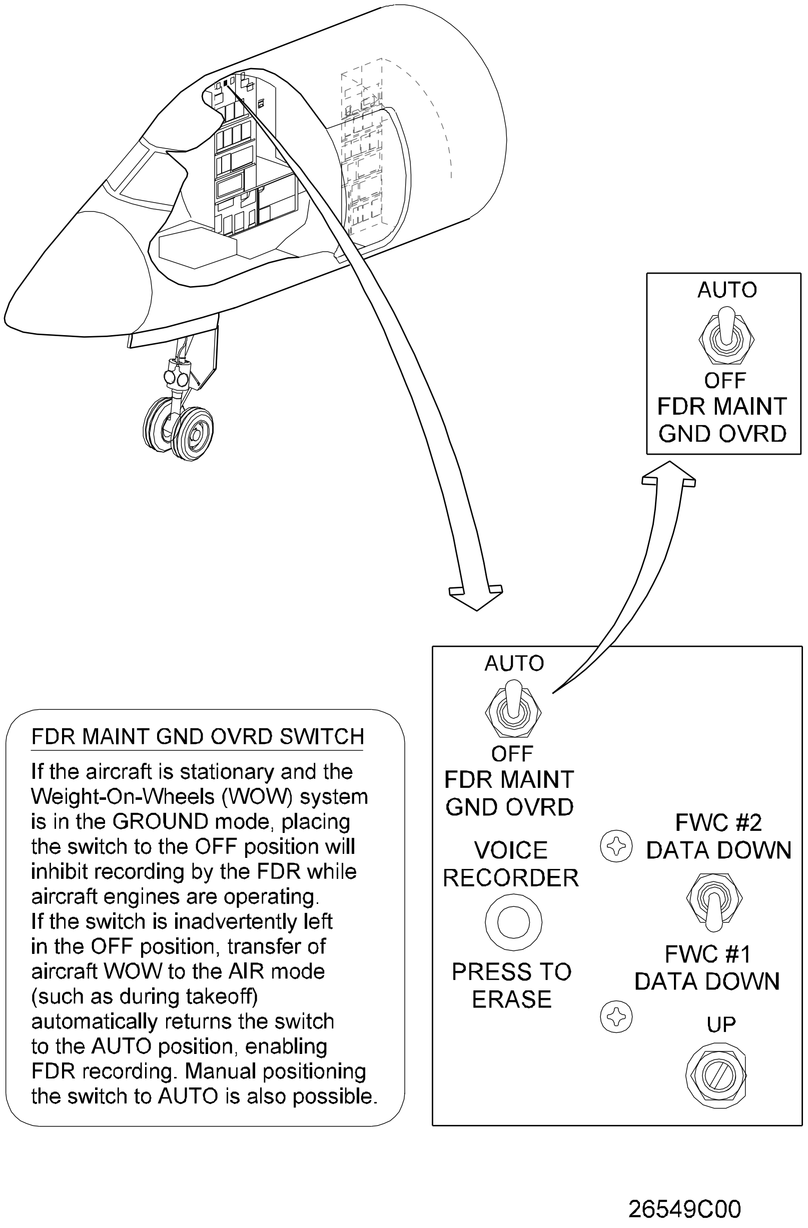


Figure 5. Flight Data Recorder Maintenance Ground Override Switch

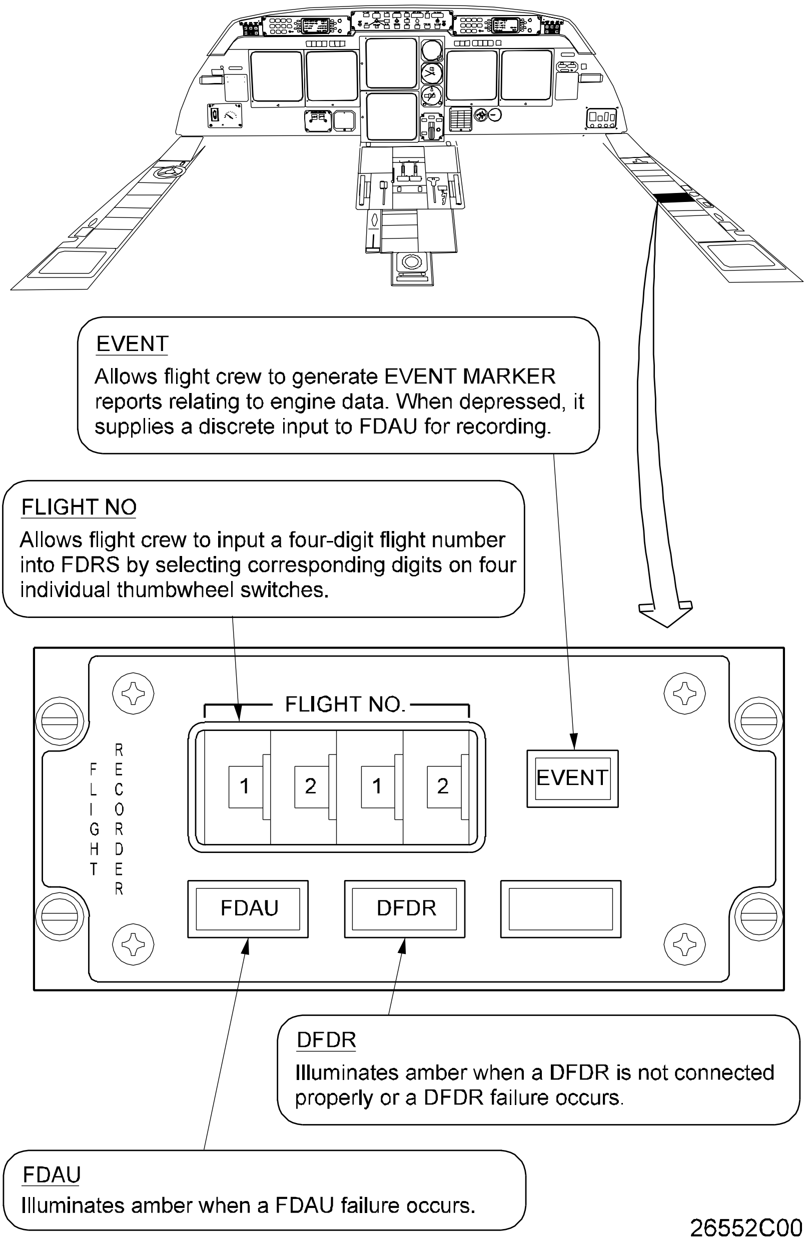


Figure 6. Flight Data Entry Panel

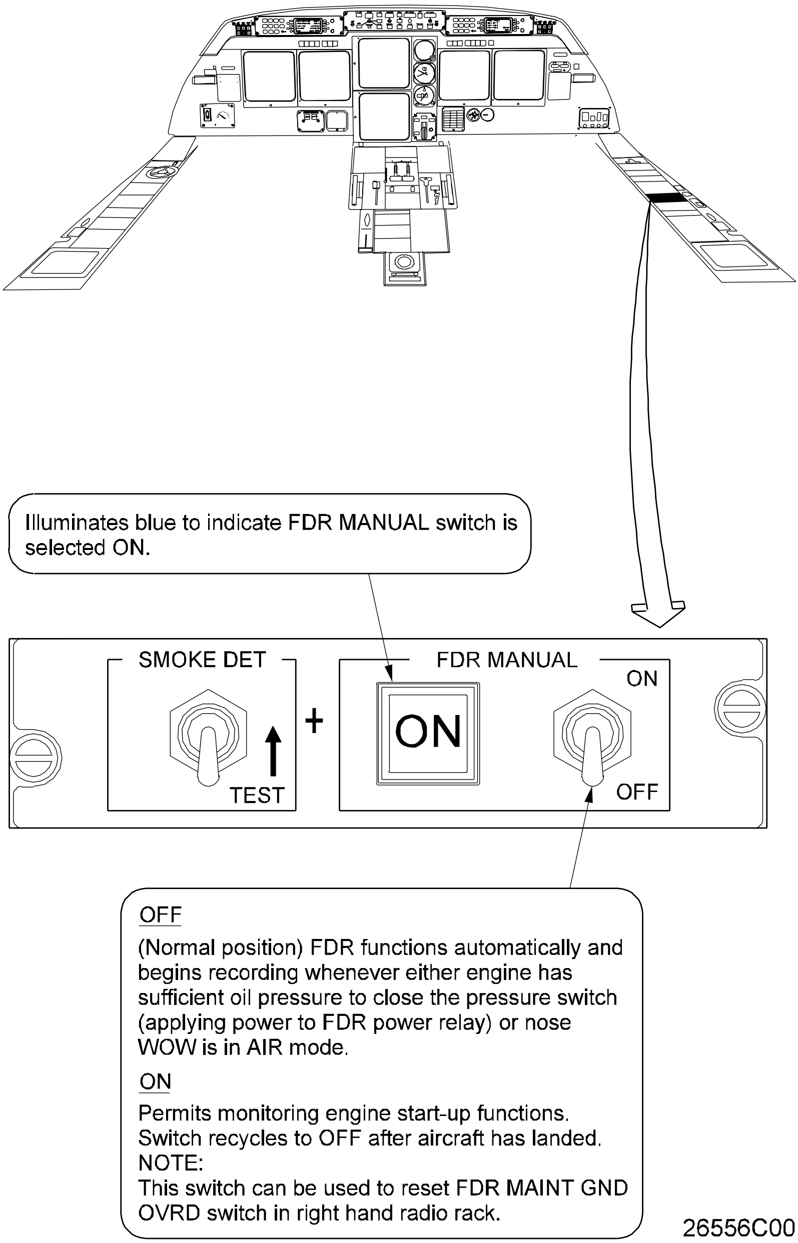


Figure 7. Flight Data Recorder Switch Panel

2A-31-32: 'G' Monitor System

1. General Description:

Airplanes SN 1034, 1156 and subsequent and airplanes SN 1000 through 1155 with ASC 118 incorporated have a gravity force ('G') monitor system to record the maximum vertical acceleration experienced by the airplane during landing. Data collected by this system is used to assist in determining the requirement for either overweight or hard landing inspections.

The monitoring system is activated during approach at 50 ft radio altitude with the landing gear extended. The system is active until the nose gear nutcracker signals weight-on-wheels. The maximum acceleration sensed, while the system is active, is recorded by the computer. Once deactivated by nutcracker logic, the system remains deactivated until the subsequent approach and landing, when activation criteria are again satisfied.

The recorded acceleration will be affected by normal occurrences experienced between main gear touchdown and nose gear nutcracker ground mode logic. These occurrences include but are not limited to:

- Ground spoiler deployment
- Rough, bumpy runways
- Travel through runway intersections
- Travel across arresting cable hardware housing

The above items can add 0.2 to 0.3 'G' to the touchdown acceleration and are considered valid loads on the airplane structure. Normal landing 'G' readings may vary from 1.3 to 1.6 'G'.

The established limit to determine the requirement for the hard landing inspection for airplane weights up to and including maximum landing weight is 2.0 'G' (2.3 'G' for airplanes SN 1214 and subsequent and airplanes SN 1000 thru 1213 with ASC 190). The limit to determine the requirement for an overweight landing inspection for airplane weights exceeding maximum landing weight decreases on a linear scale from 2.0 'G' to 1.6 'G' (2.3 'G' to 1.7 'G' for airplanes SN 1214 and subsequent and airplanes SN 1000 thru 1213 with ASC 190). The limit depends on actual airplane weight and assumes that extremely high side loads were not encountered during landing touchdown. If the actual 'G' recorded for the landing is within limits and no extremely high side loads were encountered at touchdown, no inspection is required. If extremely high side loads were encountered at touchdown, or the 'G' limit is exceeded, the flight crew shall log the event for maintenance action. See Figure 9 and Figure 10. For additional information, refer to Chapters 5 and 31 of the GIV Maintenance Manual.

2. Description of Subsystems, Units and Components:

The system is comprised of three major components:

- G-Meter Computer/Display
- Accelerometer
- G-Meter Fail Annunciator

A. G-Meter Computer/Display:

The G-meter computer (located in the right radio rack) retains the current landing plus the previous seven (7) landings in non-volatile memory. The recorded accelerations are read off the computer display.

The G-meter computer display provides the flight crew with a method to

determine actual landing force encountered by the aircraft during the previous eight touchdowns. It consists of a Light Emitting Diode (LED) display, a DISPLAY ON/OFF switch and a MODE switch. These components are shown and described in Figure 8.

B. Accelerometer:

The system accelerometer is installed in the center forward portion of the wheel well at Fuselage Station 456.7. The accelerometer is capable of measuring from -3.0 to +6.0 Gs.

C. G-Meter Fail Annunciator:

The G-meter fail annunciator is located in the right hand system monitor panel above the right hand radio rack (incidentally shown in Figure 12). The annunciator illuminates whenever the G-meter system loses power and/or the system detects a fault.

3. Controls and indications:

See Figure 8.

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
G-METER	CP	C-13	ESS DC Bus

4. Operation:

With the radio rack cover removed, the G-meter computer is seen mounted on the upper shelf of the right hand forward radio rack. When the DISPLAY SWITCH is placed to ON, all display digits will illuminate while the unit preforms a self test. After the self test is completed, the value for the most recent landing will be displayed.

The display readout is arranged as follows; the digit to the left is the landing number, 1 through 8, 1 being the most recent landing. The next 3 digits are the gravity force (“G”) readings from -3.00 to +6.00 Gs.

Each time the MODE switch is momentarily placed in the MEM position, the next landing’s value will be displayed in a last-in, first-out order, for the previous 8 landings.

The display can be reset by momentarily placing the MODE SWITCH in the RST position. This action involves only internal and front panel gravity force measurement as currently displayed and does not cause a reset of either the microcomputer or in any way modify information stored in the Electronically Erasable Programmable Read Only Memory (EEPROM).

5. Limitations:

A. Flight Manual Limitations:

There are no Flight Manual limitations for the ’G’ monitor system at the time of this revision.

B. System Notes:

Use Figure 9 or Figure 10 and readings from the G-meter computer to determine if hard landing or overweight landing inspections may be required.

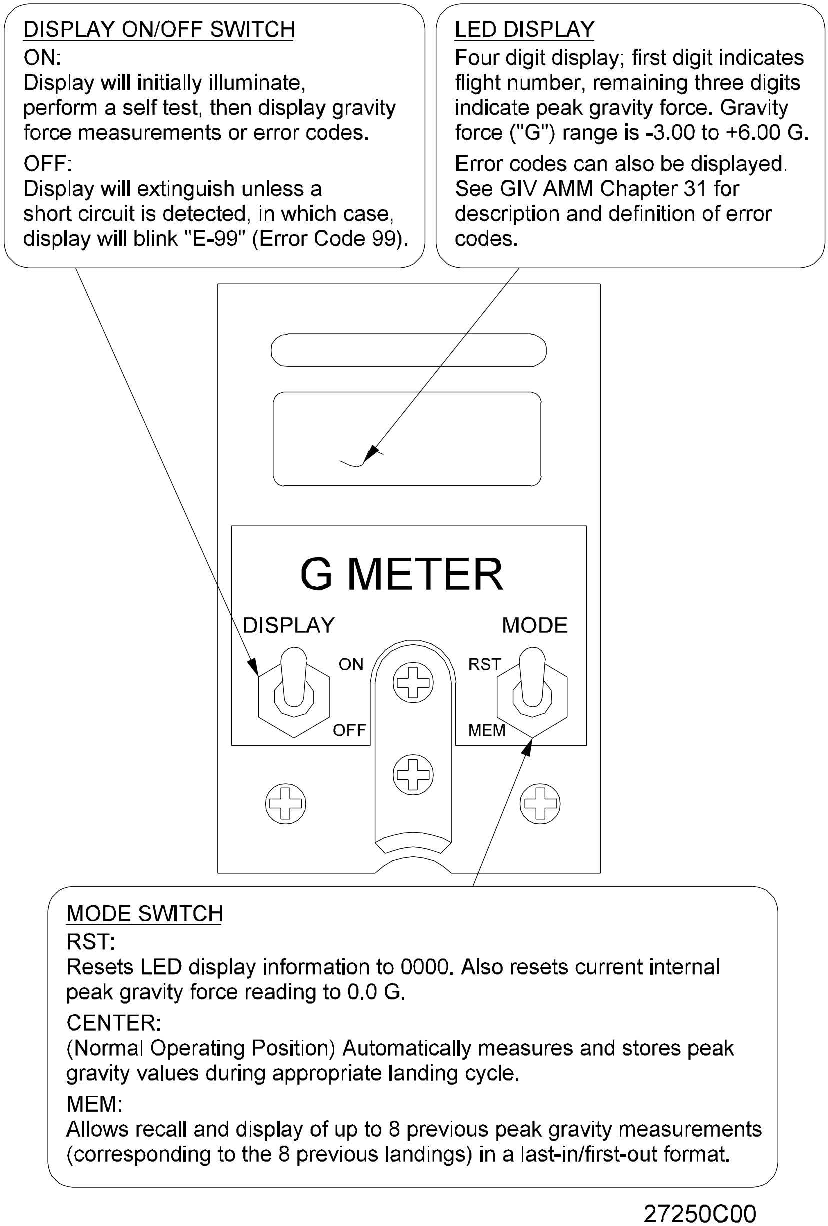


Figure 8. G-Meter Computer Display

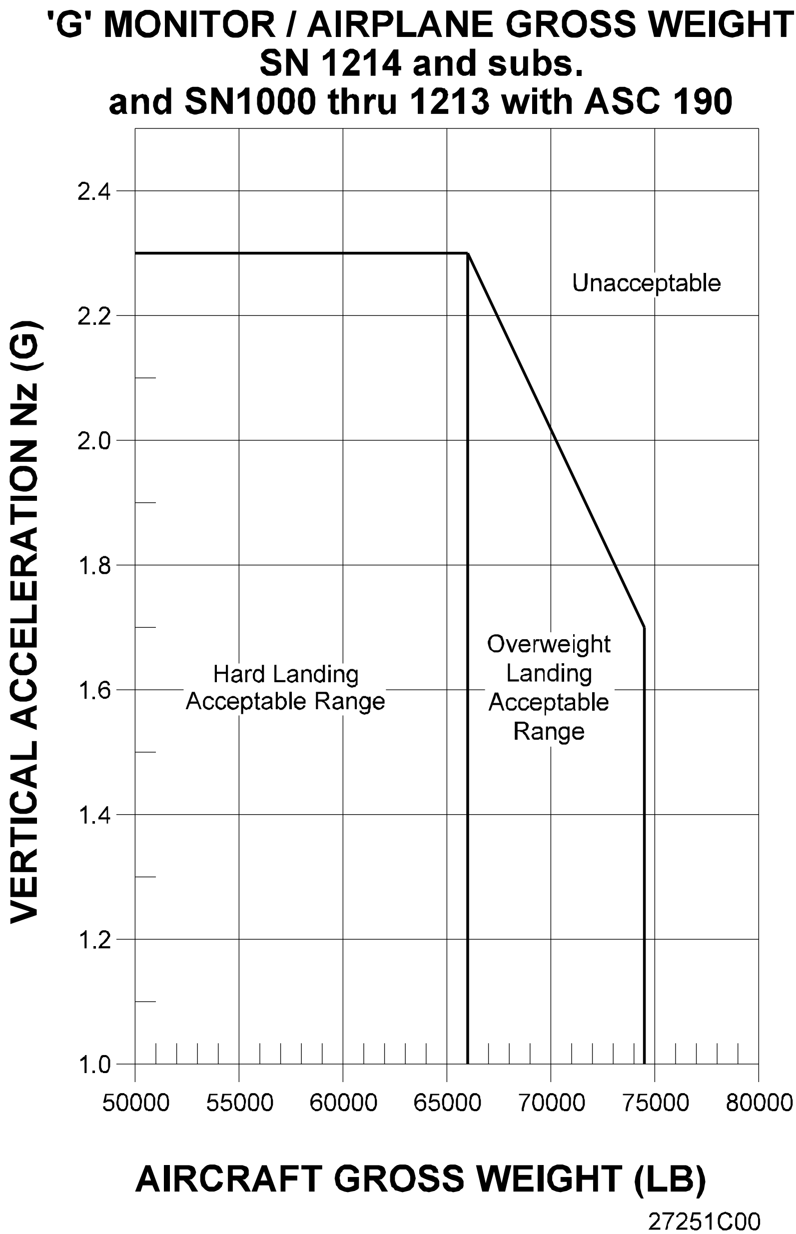


Figure 9. 'G' Monitor/Airplane Gross Weight: SN 1214 & Subs, SN 1000-1213 With ASC 190

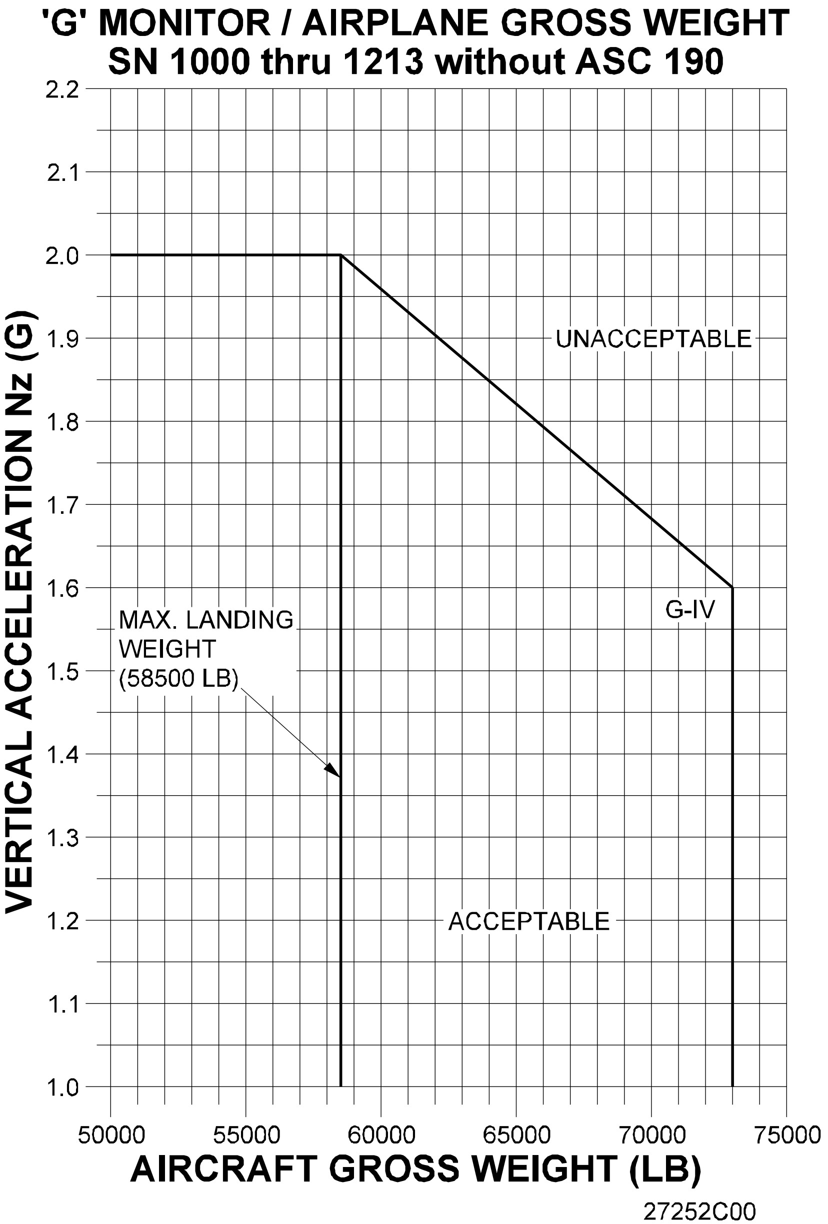


Figure 10. 'G' Monitor/Airplane Gross Weight: SN 1000-1213 Without ASC 190

2A-31-40: Central Warning System

1. General Description:

The Gulfstream IV central warning system provides the flight crew with indications of equipment status, operating conditions requiring action or indications that a system is in operation. Manual and automatic testing of the warning system indicators and indicating circuitry is also provided. Display of critical data is ensured through both automatic and manual reversionary features. Limited provisions for manipulation of display data based on personal preference are also included.

The central warning system is one of the many systems integrated into Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System and as such cannot be adequately described in the space available here. For comprehensive reference material on the central warning system, refer to Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV. For the purposes of this system description, the following central warning system units and components will be discussed:

- Avionics Standard Communications Bus
- Bus Controllers
- Fault Warning Computers
- Data Acquisition Units
- Standby Warning Lights Panel (SPZ-8000)
- Warning/Caution Inhibition Panel
- Tone Generator (SPZ-8000)
- Tone Generator (SPZ-8400)
- Maintenance Test Switch
- Trend and Limit Monitoring System

2. Description of Subsystems, Units and Components:

A. Avionics Standard Communications Bus:

Two independent Avionics Standard Communications Buses (ASCBs) provide data transfer within the central warning system. Although movement of data across the buses has bidirectional capability, it can only occur in one direction at a time. Three bus controllers (one inside each symbol generator) control operation of both ASCBs.

Systems deemed critical to airplane operation retain a private line as the primary means of communication and rely on ASCB as a backup.

B. Bus Controllers:

Bus controllers function by notifying subsystems, units and components when to transmit and receive across ASCB by assigning message priorities and inhibiting simultaneous transmissions.

A bus controller is incorporated into each of the three symbol generators in order to equal the level of symbol generator redundancy. Each bus controller is independent of the other two and only one is active at any given time. The other two, if operable, are in a standby status. Bus controller No. 1 normally assumes control of ASCB and retains control until a fault or failure occurs. At that time, bus controller No. 1 would automatically go off-line and the next bus controller in sequence would

assume control of ASCB. In addition, a blue BUS CTLR 1-2-3 FL advisory message would be displayed on the Crew Alerting System (CAS).

In addition to bus monitoring, each bus controller performs a self-test at fault warning computer and bus controller power-up. The self-test is inhibited, however, if the airplane is airborne. Failure of a bus controller self-test causes a blue BC 1-2-3 TEST FAIL advisory message to be displayed on CAS (enabled only on the ground).

C. Fault Warning Computers:

Two fault warning computers are incorporated into the central warning system. They are designated Fault Warning Computer (FWC) 1 (installed in the left radio rack) and FWC 2 (installed in the right radio rack). Each FWC receives ASCB inputs consisting of identical versions of engine data, airplane systems data and other miscellaneous data. Each FWC then places the data through identical algorithms. FWC outputs are transmitted across ASCB to the three symbol generators used by the central display system.

Although both FWCs may be available and functioning properly, only one is in control at any given time. Should the FWC in control fail, the CAS messages section will be replaced by a red “X”. Reversion to the opposite FWC (as well as routine selection of the FWC in control) is accomplished by the flight crew using either display controller. With an operable FWC in control, CAS messages are returned and the failed FWC will be identified by a blue FWC 1-2 FAIL advisory message.

The fault warning computers also contain an electronic checklist module. This feature is described in Section 2A-31-50, Central Display System.

D. Data Acquisition Units:

Two data acquisition units are incorporated into the central warning system. Data Acquisition Unit (DAU) 1 is installed in the left radio rack and is dedicated to the left engine. DAU 2 is installed in the right radio rack and is dedicated to the right engine. Each DAU receives its dedicated engine data plus other miscellaneous airplane systems data across ASCB and then transmits the data back across ASCB to the three symbol generators used by the central display system.

Each DAU has two independent channels, A and B; only one of which is in control at any given time. Should the channel in control fail, a blue DAU 1A-1B-2A-2B FL advisory message will be displayed on CAS. Reversion to the opposite channel (as well as routine selection of the channel in control) is accomplished by the flight crew using either display controller.

E. Standby Warning Lights Panel (SPZ-8000):

SPZ-8000 equipped airplanes have a Standby Warning Lights Panel (SWLP) installed on the copilot’s inboard skirt panel. Its 21 capsules are all red and cover most of the subject areas addressed by the warning (red) CAS messages. Because of capsule or CAS message space constraints, message legends of duplicated subjects are similar, but not identical, so that confusion should not result.

The SWLP will operate continuously if manual (MAN) is selected. If operated in automatic (AUTO), the panel will operate whenever EICAS display is lost on DU 3 or 4. The AUTO-MAN switch is located on the bottom of the SWLP. Preflight checks (using the warning lights dimming

and testing) may be accomplished with the SWLP in AUTO or MAN mode.

The following warning messages are displayed on the SWLP:

L ENGINE HOT	R ENGINE HOT	REV UNLOCKED
L OIL PRESS	R OIL PRESS	GND SPOILER
L FUEL PRESS	R FUEL PRESS	ACFT CONFIG
L FUEL FILTER	R FUEL FILTER	AFT EQUIP HOT
L PYLON HOT	R PYLON HOT	CAB PRESS LOW
APU FIRE	FIRE DET LOOP	CABIN DOORS
SMOKE DETECT	FLAME DETECT	—

When activated, the SWLP capsules illuminate flashing. The master warning light (“W” switchlight on glareshield warning/caution inhibition panel) also illuminates. Depressing the illuminated master warning light causes the flashing SWLP capsule(s) to revert to steady display. They will remain illuminated until the fault is corrected. The standby warning lights panel is removed from SPZ-8400 equipped aircraft. These airplanes have display unit (DU) reversionary capability, allowing engine instruments and crew alerting system (EICAS) display on DU 2, 3, 4 or 5.

F. Warning/Caution Inhibition Panel:

The primary flight crew interface with the central warning system is the warning/caution inhibition panel. Installed on both sides of the flight panel glareshield, they are shown and described in detail in Figure 11. For the purposes of this discussion, however, only the function of the WARN INHIBIT switch will be described.

To prevent the flight crew from being distracted during certain phases of flight (such as during takeoff and landing), annunciations that would normally accompany CAS messages in some cases can be inhibited. This is accomplished through use of the WARN INHIBIT switch that, when selected, inhibits aural tones and illumination of the master warning/master caution switches. It should be noted at this point that display of messages on CAS cannot be inhibited; rather, it is only the aural tones and warning/caution inhibition panel switch illumination that would normally accompany the message.

With the airplane on the ground and at least one valid radio altimeter signal present, depressing either WARN INHIBIT switch enables the inhibit mode for takeoff. After climbing through 400 feet AGL and retracting the landing gear, the inhibit mode is automatically cancelled. The flight crew may again initiate the inhibit mode on final approach after the landing gear is extended and locked, provided at least one valid radio altimeter signal is present. In all other flight conditions, the inhibit mode cannot be activated.

The following list summarizes annunciation presentation with the inhibit mode enabled:

(1) In both SPZ-8000 and SPZ-8400 equipped airplanes, all advisory (single chime) aural tones are inhibited.

(2) In both SPZ-8000 and SPZ-8400 equipped airplanes, the aural caution tone (double chime) and master caution switch illumination are inhibited, **except** with the amber CPL DATA INVALID caution message.

(3) In SPZ-8000 equipped airplanes with the SWLP in automatic (AUTO), the aural warning tone (triple chime) and master warning switch illumination are inhibited in conjunction with the following five (5) messages:

- APU FIRE
- CABIN DFRN-9.8
- CABIN PRESSURE LOW
- DAU 1-2 MISCMP-MSG
- MAIN DOOR

(4) In SPZ-8000 equipped airplanes with the SWLP in manual (MAN) and SPZ-8400 equipped airplanes, it is not possible to inhibit the aural warning tone or illumination of the master warning switch.

G. SPZ-8000 Tone Generator:

A tone generator is installed to provide audible sounds, calling the attention of the flight crew to an associated message or condition. A tone generator test panel is installed on the copilot's side console for tone testing. The aural tones provided and test positions are:

Type of Aural Tone:	Condition:	Tone Generator Test Panel Position:
Horn	Landing Gear Unsafe	1
Cricket	Airplane Overspeed	2
Beep	Altitude Alert	3
1 Chime	CAS Advisory	4
2 Chimes	CAS Caution	5
3 Chimes	CAS Warning	6
Lo/Lo/Lo	Autothrottle Disconnect	7
Lo/Hi/Lo	Manual Autopilot Disconnect	8

Warning and caution CAS alerts can be silenced by pressing the associated switch on the warning/caution inhibition panel. Advisory CAS alerts cannot be manually silenced, but will automatically silence after two (2) seconds. Airplane overspeed cannot be silenced as long as the overspeed condition exists. The landing gear unsafe warning can be silenced, either manually or automatically, in all cases except when the flaps are fully extended without the landing gear being down and locked. The airplane has a radio altimeter-activated automatic horn silencing system that will function above 1200 ft AGL except when the flaps are fully extended without the landing gear being down and locked.

H. SPZ-8400 Tone Generator:

A tone generator is incorporated into each fault warning computer to provide audible sounds, calling the attention of the flight crew to an associated message or condition. Tone volume is adjusted from the TONE prompt on the TEST page of the pilot's display controller. The aural tones provided by the tone generator are:

Type of Aural Tone:	Condition:
Klaxon™ (Hi/Lo - Hi/Lo)	Landing Gear Unsafe
Cricket	Airplane Overspeed
Beep	Altitude Alert
1 Chime	CAS Advisory [1]
2 Chimes	CAS Caution
3 Chimes	CAS Warning
Lo/Lo/Lo	A/T Disconnect
Lo/Hi/Lo	Manual A/P Disconnect

(1) Only those messages not resulting from deliberate crew action.

Warning and caution CAS alerts can be silenced by pressing the associated switch on the warning/caution inhibition panel. Advisory CAS alerts cannot be manually silenced, but will automatically silence after one cycle. Airplane overspeed cannot be silenced as long as the overspeed condition exists. The landing gear unsafe warning can be silenced, either manually or automatically, in all cases except when the flaps are fully extended without the landing gear being down and locked. The airplane has a radio altimeter-activated automatic horn silencing system that will function above 1200 ft AGL except when the flaps are fully extended without the landing gear being down and locked.

The following CAS advisory messages do not generate a tone when activated:

AC EXT POWER	FMS 3 ACTIVE	ISOLATION VLV OPEN
APU ALT OFF	FUEL INT TANK OPEN	MAINT SWITCH ON
L-R COWL AI ON	FUEL XFLOW OPEN	SPD BRAKE EXTENDED
DC EXT POWER	GND SPOILER UNARM	TONE GEN 1-2 FAIL
EXT BATT SWITCH ON	IRS 1-2 HI LAT ALN	L-R WING AI
FGC NOT USING IRS 1-2		

I. Maintenance Test Switch:

A maintenance test switch is installed above the left and right radio racks (Figure 12). Airplanes having ASC 221 have an additional switch on the copilot's circuit breaker panel. The maintenance test switches are labeled MAINT TEST and are guarded to OFF. When any MAINT TEST switch is selected ON, advisory CAS messages that are normally suppressed can be viewed. Trend and limit monitoring is also inhibited. A blue MAINT SWITCH ON advisory message is also displayed to alert the flight crew that the MAINT TEST switch is on.

J. Trend and Limit Monitoring System:

Each fault warning computer contains a trend and limit monitoring function designed to record, store and display the following types of data:

- Engine exceedances of the following maximum limits:
  - Turbine Gas Temperature (TGT)
  - Low Pressure Turbine Speed (LP RPM)
  - High Pressure Turbine Speed (HP RPM)
  - Low Pressure Turbine Vibration (LP EVM)

    - High Pressure Turbine Vibration (HP EVM)
    - Parameter that tripped the exceedance monitor
- APU exceedances of the following maximum limits:
    - Turbine Gas Temperature (TGT)
    - Turbine Speed (RPM)
    - Parameter that tripped the exceedance monitor
- Brake temperature exceedances and time of occurrence
- Parameter that tripped the exceedance monitor

Trend data recording is used to monitor the long term histories and relative health of the engines and APU. Engine trend data is automatically recorded when the airplane reaches approximately 100 knots on takeoff, again when the airplane establishes steady cruise flight and thereafter at approximately ninety (90) minute intervals. APU trend data is recorded just prior to the first engine start of a flight. In addition to these trend recordings, an engine start log is maintained to track the number of engine starts.

Engine limit data (hereafter referred to as exceedance data) is recorded when a TGT, HP, LP or EVM limit is exceeded on either engine. Recording also occurs when a fire indication is detected on either engine or when a fire test is performed on an operating engine while airborne. The recording includes pre- and post-exceedance data points so that a detailed time-versus-parameter plot of the exceedance may be constructed. Certain CAS messages are associated with engine exceedance detection and, if the exceedance is not corrected, the subsequent recording of the exceedance. On SPZ-8000 equipped airplanes, a blue ENGINE EXCEEDANCE advisory message is displayed when an exceedance is detected. On SPZ-8000 equipped airplanes having ASC 415 and SPZ-8400 equipped airplanes, the blue ENGINE EXCEEDANCE advisory message has been removed and replaced with a red ENGINE EXCEEDANCE warning message. If the detected exceedance is not corrected, it is recorded and a blue EXCEEDANCE RECORD advisory message is displayed (SPZ-8000 equipped airplanes having ASC 415 and SPZ-8400 equipped airplanes). The exceedance can then be viewed on the EXCEEDANCES system page.

APU exceedance data is recorded when an EGT or RPM limit is exceeded, a fire indication is detected or when an APU fire test is performed while airborne. Like engine limit data recording, APU limit recording includes pre- and post-exceedance data points so that a detailed time-versus-parameter plot of the exceedance may be constructed. CAS messages are also associated with APU exceedance detection and, if the exceedance is not corrected, the subsequent recording of the exceedance. A blue APU EXCEEDANCE advisory message is displayed when an exceedance is detected. If the detected exceedance is not corrected, it is recorded and a blue EXCEEDANCE RECORD advisory message is displayed (SPZ-8000 equipped airplanes having ASC 415 and SPZ-8400 equipped airplanes). The exceedance can then be viewed on the EXCEEDANCES system page.

Brake temperature exceedance data is recorded when a brake temperature exceeds 625°C. On SPZ-8000 equipped airplanes having ASC 415 and SPZ-8400 equipped airplanes, a blue EXCEEDANCE

RECORD advisory message is displayed. The exceedance can then be viewed on the EXCEEDANCES system page.

The EXCEEDANCES system page display holds the most recent exceedance data until airplane power is turned off or another exceedance is detected. All exceedances are held in the fault warning computer's memory, however. If the EXCEEDANCES page is selected before any exceedances are recorded, a white NO EXCEEDANCES RECORDED message is displayed on the page.

Although engine trend and limit data recording operates automatically, a manual request for recording can also be made by the flight crew. On airplanes SN 1253 and subsequent and SN 1000 through 1252 having ASC 255, a crew activated recording switch (labeled SNAPSHOT, shown in Figure 13) allows the flight crew to record maximum values and the time in exceedance for up to five (5) minutes on the ground or while in flight.

The memory allocated within each fault warning computer for trend and limit recording is a nonvolatile type of memory known as Electrically Erasable and Programmable Read Only Memory (EEPROM). On SPZ-8000 equipped airplanes, 64 kilobytes of memory is allocated, with provisions for an additional 64 kilobytes of growth memory. On SPZ-8400 equipped airplanes, 128 kilobytes of memory is allocated. When the fault warning computer is tested, a message indicating EEPROM memory usage is generated.

On SPZ-8000 equipped airplanes having ASC 415 and SPZ-8400 equipped airplanes, a blue T&L >80% FULL advisory message is displayed on CAS when 80% of the allocated EEPROM memory has been used. The flight crew should then notify maintenance personnel to download and erase the fault warning computer memory at the next convenient maintenance period. The T&L >80% FULL advisory message is displayed only when the airplane is on the ground at speeds of less than 50 knots.

The fault warning computer contains a dedicated output for data downloading. Downloading is performed using the data loader.

3. Controls and Indications:

(See Figure 11 through Figure 13.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
BUS CONT #1	CP	K-2	ESS DC Bus
BUS CONT #2	CP	L-2	ESS DC Bus (1)
BUS CONT #3	CP	M-2	ESS DC Bus (2)
FWC #1	CP	A-13	ESS DC Bus
FWC #2	CP	B-13	ESS DC Bus (3)
DAU #1A	CP	A-14	ESS DC Bus
DAU #2A	CP	B-14	ESS DC Bus (4)
DAU #1B	CP	C-14	ESS DC Bus (3)
DAU #2B	CP	D-14	ESS DC Bus
SPZ-8000 SHUT DN (5) (6)	CP	L-4	R MAIN DC Bus
TONE WARN #1	P	A-5	ESS DC Bus
TONE WARN #2	P	B-5	R MAIN DC Bus

[(1)] R MAIN DC bus on SPZ-8000 equipped airplanes.
[(2)] L MAIN DC bus on SPZ-8000 equipped airplanes.
[(3)] R MAIN DC bus on SPZ-8000 equipped airplanes.
[(4)] L MAIN DC bus on SPZ-8000 equipped airplanes.
[(5)] CB installed on airplanes SN 1156 and subs, and SN 1000-1155 with ASC 92A.
[(6)] CB labeled “SPZ-8400 SHUT DN” on SPZ-8400 equipped airplanes. CB labeled “SPZ-8000 SHUT DN” on SPZ-8000 equipped airplanes (when installed by ASC 92A).

B. Warning (Red) CAS Messages:

CAS Message:	Cause or Meaning:
DAU 1-2 MISCMP-MSG	Miscompare between data channels of a serious nature.
ENGINE EXCEEDANCE [(1)]	LP, HP or TGT above limits.

[(1)] SPZ-8000 equipped airplanes with ASC 415 and SPZ-8400 equipped airplanes.

C. Caution (Amber) CAS Messages:

CAS Message:	Cause or Meaning:
DAU 1-2 MISCMP ENG	Engine data miscompare between DAU ’A’ and ’B’ channels.
DAU 1-2 MISCMP MSG	EICAS amber message miscompare between DAU ’A’ and ’B’ channels.

D. Advisory (Blue) CAS Messages:

CAS Message:	Cause or Meaning:
APU EXCEEDANCE	Fault warning computer has recorded an exceedance.
BC 1-2-3 TEST FAIL	Bus controller power-up self test has failed.
BUS CTLR 1-2-3 FAIL	Indicated bus controller has failed.
DAU 1A-1B-2A-2B FL	Indicated DAU channel has failed.
DAU 1-2 MISCMP MSG	EICAS blue message miscompare between DAU ’A’ and ’B’ channel.
ENGINE EXCEEDANCE [(1)] EXCEEDANCE RECORD [(2)]	Fault warning computer has recorded an exceedance.
FWC 1-2 FAIL	Indicated fault warning computer has failed.
MAINT SWITCH ON	Maintenance switch is on.
T&L >80% FULL [(2)]	Fault warning computer memory is greater than 80% full.
TONE GEN 1-2 FAIL [(3)]	Aural warning tone generator has failed.

[(1)] Deleted by ASC 415 on SPZ-8000 equipped airplanes.
[(2)] SPZ-8000 equipped airplanes with ASC 415 and SPZ-8400 equipped airplanes.
[(3)] Presented as TONE GEN FAIL on SPZ-8000 equipped airplanes.

4. Limitations:

A. Flight Manual Limitations:

The are no Flight Manual limitations for the central warning system at the time of this revision.

B. System Notes:

(1) Bus Controllers:

The bus controller self-test will fail if power is momentarily removed

from both fault warning computers after the bus controllers are powered. If this occurs, the system can be reset and the message cleared by reapplying power to the airplane in the normal power-up sequence or by pulling and resetting all three bus controller circuit breakers.

(2) Exceedance Recording:

It is possible that display of the blue APU EXCEEDANCE, ENGINE EXCEEDANCE or EXCEEDANCE RECORD advisory messages could be caused by a fire. The flight crew should be alert for other indications of fire and take appropriate action as necessary.

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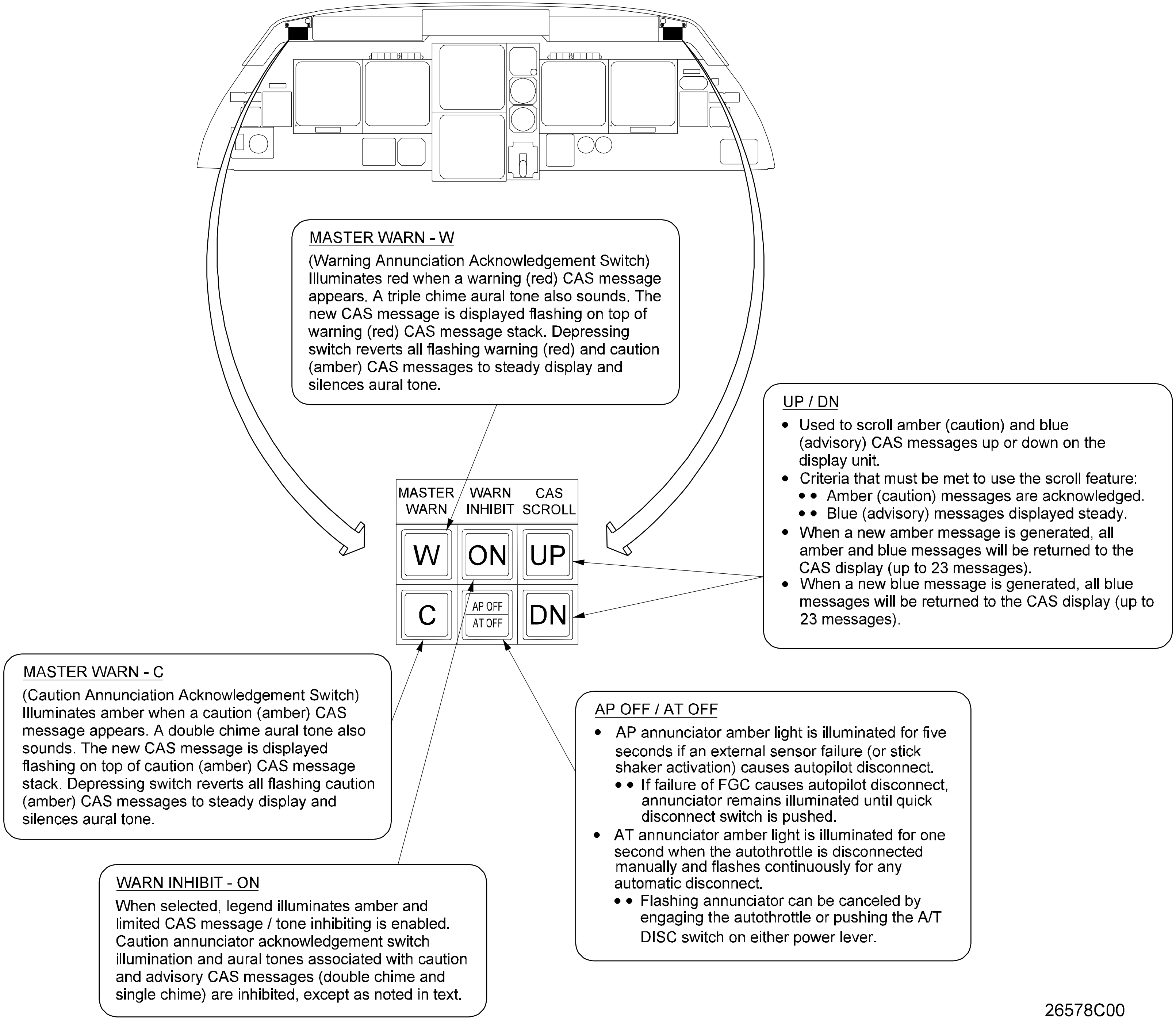

Figure 11. Warning/ Caution Inhibition Panel

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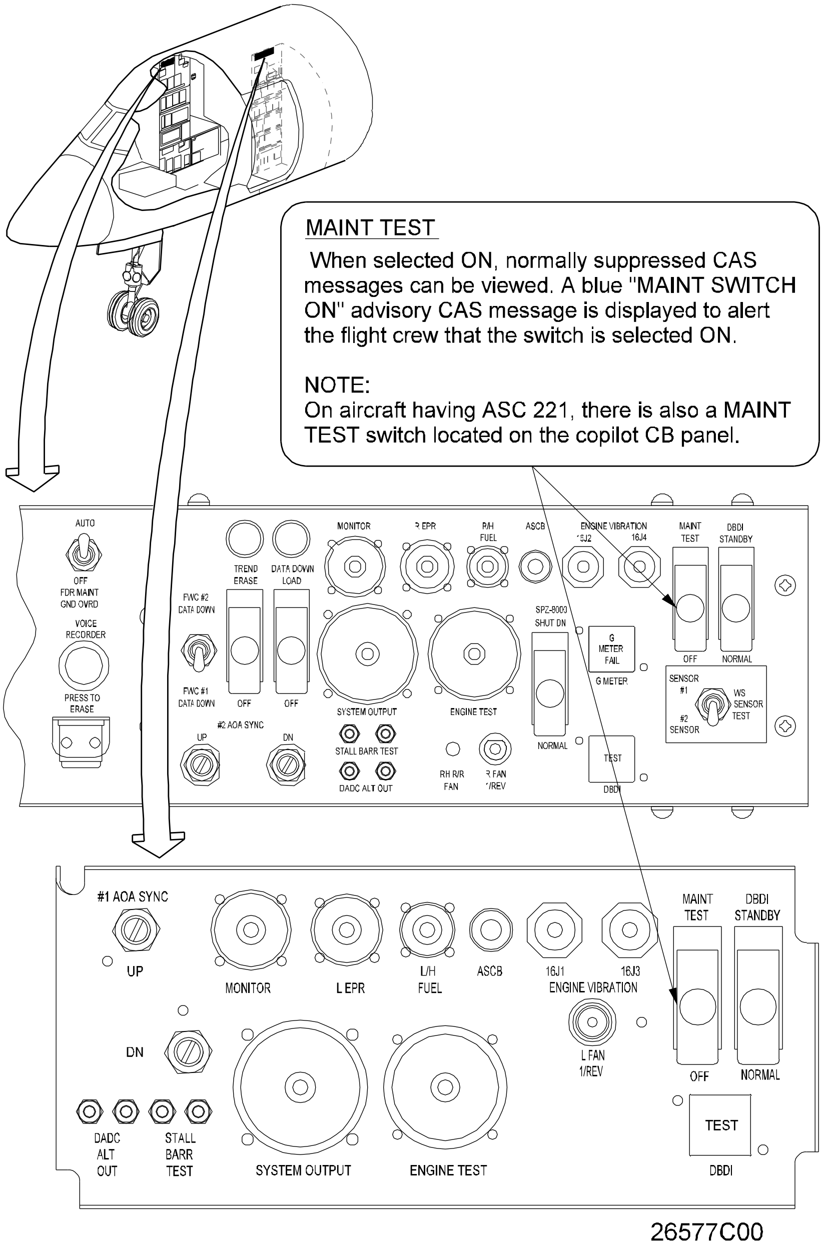
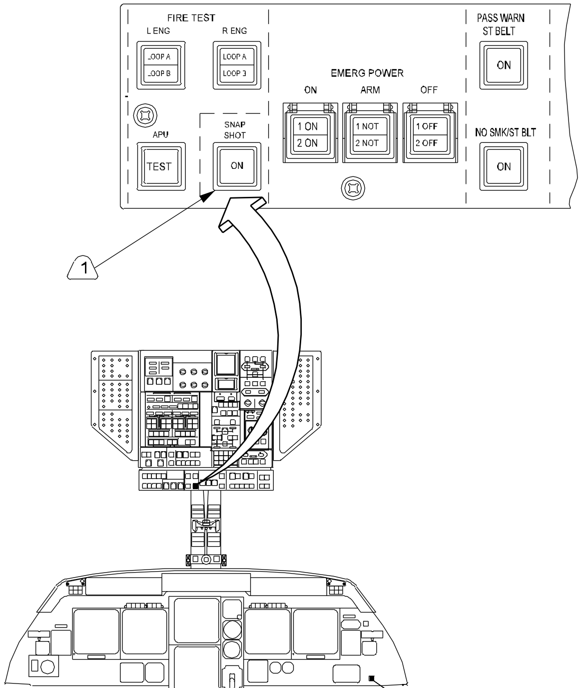


Figure 12. Maintenance Test Switch



NOTES:

1 AIRCRAFT 1253 AND SUBSEQUENT

2 AIRCRAFT 1000 - 1252 HAVING ASC 255

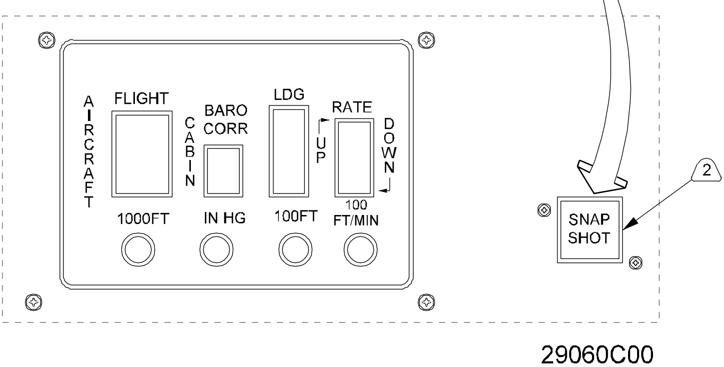


Figure 13. Snapshot Switch

2A-31-50: Central Display System

1. General Description:

The Gulfstream IV central display system provides the flight crew a means to observe visual displays of conditions in selected systems. To enhance system operation, six (6) Display Units (DUs) present integrated data including Primary Flight Display (PFD) data, Navigation Display (ND) data, Engine Instruments (EI) data and Crew Alerting System (CAS) data. PFD, ND and EICAS presentations can be further categorized as:

- Pitch and roll attitude
- Heading data
- Course orientation
- Flight director commands
- Flight path angle
- Weather presentations
- Mode and source annunciations
- Air data parameters
- Engine data
- Fault warning information
- Traffic Alert and Collision Avoidance System (TCAS) (optional)
- Enhanced Ground Proximity Warning System (EPWS) (optional)

The central display system is one of the many systems integrated into Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System and as such cannot be adequately described in the space available here. For comprehensive reference material on the central display system, refer to Honeywell's <u>SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV</u>. For the purposes of this system description, the following central display system units and components will be discussed:

- Symbol Generators
- Display Controllers
- Display Units
- Display Power Panel
- Display Unit Cooling
- Display Brightness Panel
- Display Switching and System Control Panel
- Electronic Checklist

2. Description of Subsystems, Units and Components:

A. Symbol Generators:

The three Symbol Generators (SGs) are the principal components of the central display system. Using ASCB, ARINC (Aeronautical Radio Incorporated) 429 and/or ARINC 453 data buses, the symbol generators interface with the other central display system components, flight management system, flight guidance system and navigation system to process, organize and display data in the desired format on the selected display unit. Each symbol generator has an incorporated bus controller

available to govern ASCB as required.

The symbol generators normally drive the display units as shown in the following table:

DU #:	Panel Position:	Used As:	Normally Driven By:
1	Far Left	Pilot's PFD	SG #1
2	Second From Left	Pilot's ND	SG #1
3	Upper Center	EI Display	SG #3
4	Lower Center	CAS Display	SG #3
5	Second From Right	Copilot's ND	SG #2
6	Far Right	Copilot's PFD	SG #2

Each symbol generator is powered up when the display units it normally serves is powered up. If one symbol generator should later fail, the remaining two operating symbol generators will continue providing the same displays with no degradation. This prevents a single symbol generator failure from rendering the central display system inoperative. If two symbol generators fail, the remaining operating symbol generator will drive all six DUs simultaneously, however, only four distinct display formats can be provided. In this case, the copilot's displays will be a repeat of the pilot's displays. In addition, the pilot's display controller will control central display system functions. If the pilot's display controller should fail at this stage, control would shift to the copilot's display controller.

B. Display Controllers:

Shown in Figure 17, the two (2) display controllers are used to control the central display system. The pilot's display controller is installed in the left side of the flight panel glareshield; the copilot's display controller is installed in the right side. Each display controller contains twenty pushbuttons, three knobs and a Cathode Ray Tube (CRT) screen. The twenty pushbuttons (ten function keys and ten line select keys) allow navigation data and sensor source selection, EFIS and EICAS format selection, bug setting, test functions and maintenance functions. The three knobs allow adjustment of menu parameters (SET), barometric pressure setting (BARO) and CRT screen brightness (BRT).

Each BARO knob is connected directly to the on-side Digital Air Data Computer (DADC): DADC #1 for the pilot's side, DADC #2 for the copilot's side. When the BARO knob has been turned past its barometric pressure setting limit, the altitude scale on the PFD will show a red X. Rotating the knob back to within its limit will remove the red X and return the scale to normal.

In addition to setting the local altimeter setting for the displayed air data source, the BARO knob also has a “Push-To-Standard” feature incorporated. By depressing the PUSH STD portion of the knob, the flight crew can select and display the numeric value of standard day barometric pressure (29.92 in Hg or 1013 Mb) or an iconic representation of this value, “STD”.

If a display controller fails, the associated PFD and ND revert to the following display source selections:

PFD:	ND:
On-side IRS, DADC, RA & NAV	MAP Format
RAD ALT Set = 200	On-side FMS
Bearing Pointers = AUTO	ID WAYPT
BARO = IN	
FD CMD = SC	

If both display controllers fail, EICAS uses DAU 1A, DAU 2B and FWC 1. Operation of the BARO knobs will be unaffected due to the direct connection to their on-side DADC.

C. Display Units:

The six display units (Figure 17) are interchangeable, high resolution, eight inch by eight inch cathode ray tubes. Driven by the symbol generators and commanded by the display controllers, the display units are used by the central display system to present air data and display information about the airplane's attitude, flight director, navigation, engines and aircraft systems. Once power is applied, it takes approximately ten (10) minutes for full color stabilization to take place.

Display units used as primary flight displays also present other data vital for safe airplane operation such as the navigation data source, radio altitude, course, heading and distance.

Display units used as navigation displays show navigation and flight plan data from the navigation receivers, flight management system, and weather radar in one of three desired formats:

- MAP: partial compass rose and range displays
- COMP: (compass) traditional HSI with 360° compass rose
- PLAN: "north up" representation of active flight plan

If the component providing data to the PFD or ND fails or provides invalid data, a red X will appear on that portion of the PFD or ND. If a symbol generator fails, a red X will appear across the entire display unit. If the display unit goes blank, display unit failure should be suspected.

D. Display Power Panel:

The six display units are selected on and off in pairs using the PILOT, EICAS and COPILOT switches on the display power panel (DISPLAYS section of the cockpit overhead panel, Figure 14). All three switches are guarded switches and when not selected on, an amber OFF legend is displayed in the switch. Design of the panel is "fail-operational" in that the affected display units will be powered should a failure occur within the panel.

E. Display Unit Cooling:

Two fans are installed to provide circulation of cooling air to the display units. On airplanes Serial Number (SN) 1000 through 1155 not having Aircraft Service Change (ASC) 87, the cooling fans automatically start and remain on any time electrical power is applied to the airplane, regardless of whether the display units are on or off. The fans may be shut off, however, by pulling the DISPLAYS FAN #1 and DISPLAYS FAN #2 circuit breakers (copilot's circuit breaker panel, D-5 and D-6, respectively).

On airplanes SN 1156 and subsequent and SN 1000 through 1155 having ASC 87, the display unit cooling fans (and also the display controllers) are shut off when the airplane is on the ground and all three DISPLAYS switches are selected OFF. When the airplane is airborne, the cooling fans and display controllers are operational at all times. In addition to being able to control the cooling fans without exercising the circuit breakers, incorporation of ASC 87 eliminates the possibility of inadvertently operating the display units without the cooling fans operating.

F. Display Brightness Panel:

DU brightness is controlled by the DISPLAY BRIGHTNESS panel located on the cockpit center pedestal and shown in Figure 15. Three (3) dual concentric potentiometers (knob sets consisting of an inner and outer knob) are provided. Each knob set is dedicated to a pair of DUs: pilot’s, EICAS and copilot’s. The inner and outer knobs control the DUs in the pair.

Display brightness is also controlled by a photo-electric sensor in each DU that will automatically adjust display brightness based on ambient light levels.

G. Display Switching and System Control Panel:

(See Figure 18 for SPZ-8000 and Figure 19 for SPZ-8400.)

The display switching and system control panel is located on the cockpit overhead panel. It provides the flight crew with reversionary control of the central display system in the event of display unit or symbol generator failures. The control panel is divided into two sections, labeled DISPLAY SWITCHING and SYMBOL GENERATOR CONTROL. The function of each section is outlined as follows:

(1) DISPLAY SWITCHING **for SPZ-8000 equipped airplanes** is accomplished by three (3) selector knobs: PILOT, EICAS and COPILOT.

If either PFD fails, selecting PFD XFER with the associated DISPLAY SWITCHING knob moves the PFD display to the ND. If an ND fails, ND OFF is selected.

If either the Engine Instruments (EI) display or the Crew Alerting System (CAS) display fails, the remaining operating display can provide a compacted presentation of both displays. Selection of the EICAS DISPLAY SWITCHING knob to TOP CMPT presents all information on the EI display; selection to BOT CMPT presents all information on the CAS display.

(2) DISPLAY SWITCHING **for SPZ-8400 equipped airplanes** is accomplished automatically by the central warning system in the following manner:

- Failure of either PFD switches that PFD to the associated ND
- Failure of either ND shuts off that ND
- Failure of the EI display causes all of EICAS to be displayed compacted on the CAS display
- Failure of the CAS display causes all of EICAS to be displayed compacted on the EI display

Manual switching provisions are incorporated to override the automatic switching feature through the use of two (2) selector

knobs: PILOT and COPILOT. Manual switching is also required to restore normal operation after automatic switching has taken place. Note that manual switching only affects DU 2 and DU 5. The available manual selections are described as follows.

(a) PILOT Selector:

- NORM - normal presentation
- PFD - displays PFD information on DU 2
- ENG / EICAS - with DU 3 failed, EI data is displayed on DU 2

(b) COPILOT Selector:

- NORM - normal presentation
- PFD - display of PFD information on DU 5
- CAS / EICAS - with DU 4 inoperative, normal CAS data is displayed on DU 5

(3) SYMBOL GENERATOR CONTROL **for both SPZ-8000 and SPZ-8400 equipped airplanes** provides three (3) selectors: PILOT, EICAS and COPILOT. Each selector provides reversionary control of its associated SG, as shown in Figure 18 and Figure 19 and summarized in the following table:

DISPLAY SYSTEM CONTROL Selector and Position:			**Display System and Symbol Generator Used:**		
PILOT	**EICAS**	**COPILOT**	**Pilot's**	**EICAS**	**Copilot**
NORM	NORM	NORM	1	3	2
ALT	NORM	NORM	3	3	2
ALT	ALT	NORM	2	2	2D
ALT	ALT	ALT	1	3	2
ALT	NORM	ALT	3	3	3D
NORM	NORM	ALT	1	3	3
NORM	ALT	NORM	1	2	2
NORM	ALT	ALT	1	1	1D

(1) NORM = Normal
ALT = Alternate
D = Duplicate of source data in use by pilot's display system

H. Electronic Checklist:

Electronic checklist modules are installed in the fault warning computers of all SPZ-8400 equipped airplanes, in SPZ-8000 equipped airplanes SN 1144 and subsequent, and in SPZ-8000 equipped airplanes SN 1000 through 1143 having ASC 178 incorporated. The checklist is controlled with either the ND joystick and associated switches (Figure 16) or the display controller.

The electronic checklist can be displayed with the ND in the MAP, COMP or PLAN mode. When displayed, it appears in the lower center portion of the ND. The electronic checklist has the following features:

- 12 lines of text per checklist page (maximum)
- Display for NORMAL, ABNORMAL and EMERGENCY procedures (EMERGENCY procedures normally displayed on copilot's ND only)

- Automatic display (auto-callup) of EMERGENCY procedures (normally displayed on copilot's ND only)

When selected by the display controller, the electronic checklist is displayed on the on-side ND. A typical checklist is composed of data generated by the selected fault warning computer and contains the following:

- Label (NORMAL/ABNORMAL/EMERGENCY)
- Cursor
- Procedure title
- Procedure page index
- Procedure step description

The electronic checklist can be selected by only one display controller at a time. Once selected by one display controller, the checklist is displayed on the on-side ND and the ability to select the checklist for display is removed from the cross-side display controller.

The title page of the electronic checklist (labeled TITLE PAGE) is the first item displayed in the NORMAL index. The TITLE PAGE must be reviewed by the flight crew before any other selection on the NORMAL, ABNORMAL or EMERGENCY index can be made. For the emergency procedures permitted to auto-callup, reviewing the TITLE PAGE is not required, however. For through flights, reselecting the TITLE PAGE resets all NORMAL checklists, making them once again available for use.

The five (5) auto-callup checklists are:

- ENGINE FIRE
- ENGINE OVERHEAT
- APU FIRE
- LOSS OF PRESSURIZATION
- THRUST REVERSER UNLOCK

The ENGINE FIRE checklist has priority over all other checklists. Regardless of the checklist in progress, the ENGINE FIRE checklist will be displayed when engine fire warnings are triggered. When all line items in the ENGINE FIRE checklist have been acknowledged by the flight crew, the previous checklist in progress will be displayed. If any one of the other four auto-callup checklists is in progress when a second auto-callup checklist of equal priority is activated, the second checklist will wait in queue. When all line items in the first checklist have been acknowledged by the flight crew, the second checklist will be displayed.

If an engine or APU fire test is accomplished while the BEFORE STARTING ENGINES checklist is displayed, the auto-callup feature will cause the ENGINE FIRE or APU FIRE checklist to be displayed.

If the selected fault warning computer should fail while a checklist is displayed, a red CHECKLIST UNAVAILABLE message is displayed in the checklist window below the title page.

3. Controls and Indications:

(See Figure 16 and Figure 17.)

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
DISPLAY CONT #1	CP	F-5	ESS DC Bus [(1)]
DISPLAY CONT #2	CP	G-5	R MAIN DC Bus
DISPLAY UNIT #1	CP	E-6	ESS DC Bus
DISPLAY UNIT #2	CP	F-6	R MAIN DC Bus
DISPLAY UNIT #3	CP	G-6	ESS DC Bus
DISPLAY UNIT #4	CP	H-6	R MAIN DC Bus
DISPLAY UNIT #5	CP	I-6	L MAIN DC Bus
DISPLAY UNIT #6	CP	J-6	R MAIN DC Bus
DISPLAYS FAN #1	CP	D-5	ESS DC Bus [(2)]
DISPLAYS FAN #2	CP	D-6	R MAIN DC Bus [(3)]
SYM GEN #1	CP	K-3	ESS DC Bus
SYM GEN #2	CP	L-3	R MAIN DC Bus
SYM GEN #3	CP	M-3	L MAIN DC Bus

[(1)] L MAIN DC bus on SPZ-8000 equipped airplanes.
[(2)] ESS AC bus ϕA on SN 1000 & SN 1002-1095 (excluding 1034) not having ASC 49.
[(3)] R MAIN AC bus on SN 1000 & SN 1002-1095 (excluding 1034) not having ASC 49.

B. Warning (Red) CAS Messages:

CAS Message:	Cause or Meaning:
CHECK PFD 1-2 [(1)]	Hazardously misleading information on both PFDs.

[(1)] Presented as CHECK PFD on SPZ-8000 equipped airplanes.

C. Caution (Amber) CAS Messages:

CAS Message:	Cause or Meaning:
CHECK DU 1-2-3-4-5-6	Information on indicated display unit may be incorrect.
DU FAN 1-2 FAIL	Respective display unit cooling fan has failed.

D. Advisory (Blue) CAS Messages:

CAS Message:	Cause or Meaning:
CHECKLIST MISMATCH	Different checklists installed in FWC 1 and FWC 2.
DC CONFIG MISMATCH [(1)]	Disagreement between the 2 display controllers' configuration.
DISP CTLR 1-2 FAIL	A display controller has failed.
DU 1-2-3-4-5-6 HOT	DU hot (266° F [130° C]).
PROG MSG 1-2 FAIL [(1)]	Programmable message modules have failed.
PROG MSG 1-2 MISMATCH [(1)]	Programmable message modules do not contain the same messages.
SG 1-2-3 FAIL	Indicated symbol generator has failed.
SG 1-2-3 HOT	Indicated symbol generator is hot.

[(1)] SPZ-8400 equipped airplanes.

4. Limitations:

A. Flight Manual Limitations:

(1) Electronic Checklist:

(a) **For SPZ-8000 equipped airplanes:** electronic checklist document number 1159AV41201-40 corresponds to this revision of the Gulfstream Aerospace GIV Airplane Flight Manual and Gulfstream Aerospace GIV Operating Manual.

(b) **For SPZ-8400 equipped airplanes:** electronic checklist document number 1159AV41202-40 corresponds to this revision of the Gulfstream Aerospace GIV Airplane Flight Manual and Gulfstream Aerospace GIV Operating Manual.

(c) **Acceptable checklists:**

- Electronic Checklist Document Numbers 1159AV41201-20 and 1159AV41202-20 are acceptable for use until the checklist modules can be updated.
- Electronic checklist document numbers 1159AV41201-8 and 1159AV41202-10 are acceptable for use until the checklist modules can be updated.

(d) **Verifying checklist version:** checklist version can be verified by selecting TITLE PAGE of electronic checklist.

B. System Notes:

(1) Display Units:

To extend the service life of the display units, it is recommended that:

- conditions of extreme heat or cold be corrected before turning on the display units
- the display units should be selected OFF when not needed
- the display unit fans be selected OFF when display units are selected OFF
- all six display units are turned on whenever the EICAS display units are needed

(2) Electronic Checklist:

Any change to the airplane that results in the electronic checklist being incompatible with the required flight crew procedure to the extent that the electronic checklist could be considered hazardously misleading will require the electronic checklist to be either revised or disabled entirely.

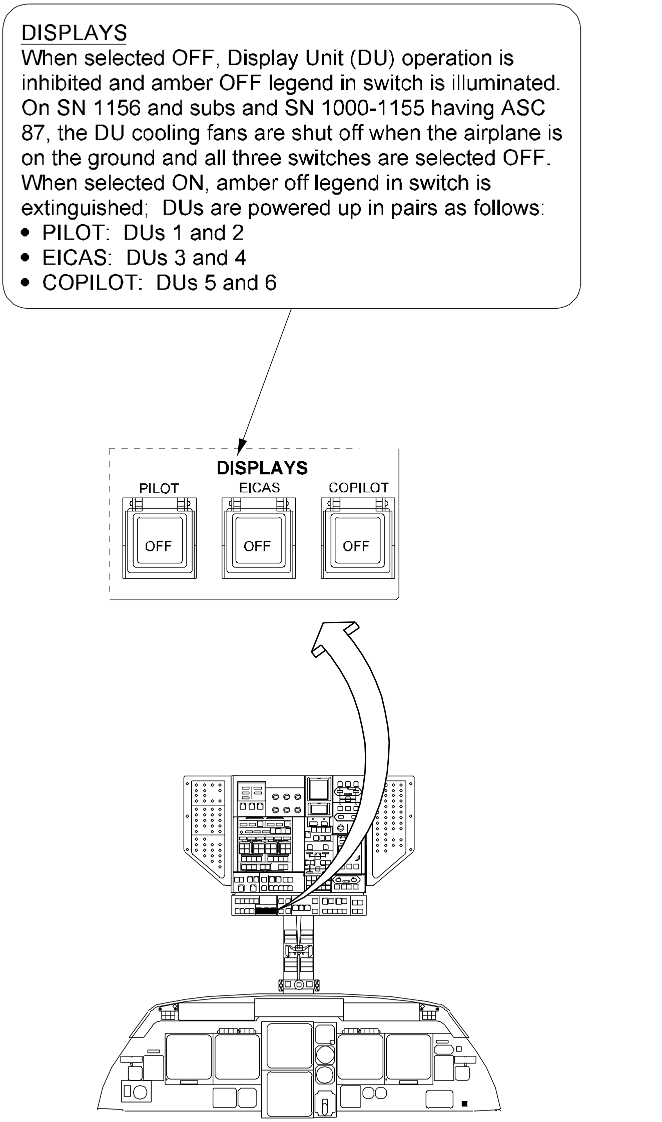


Figure 14. Display Power Panel

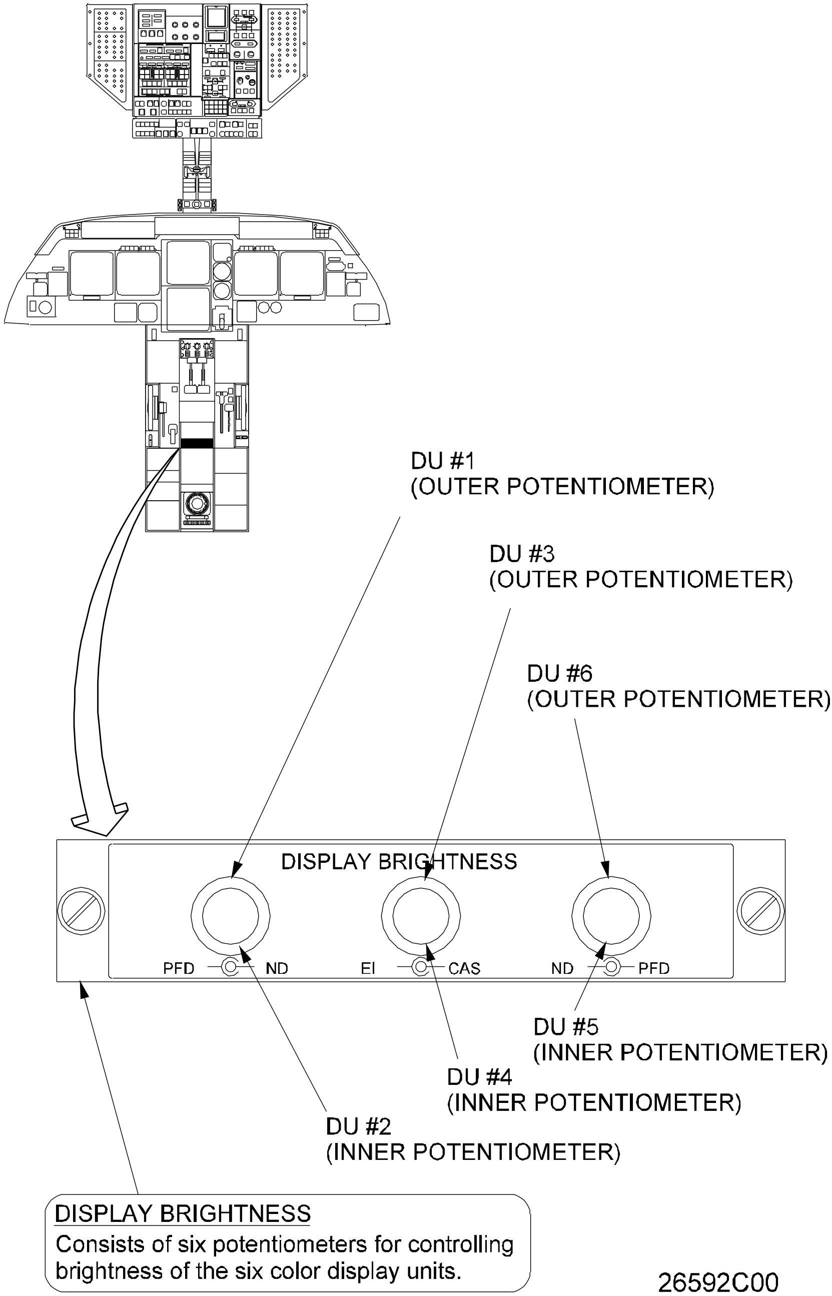


Figure 15. Display Brightness Panel

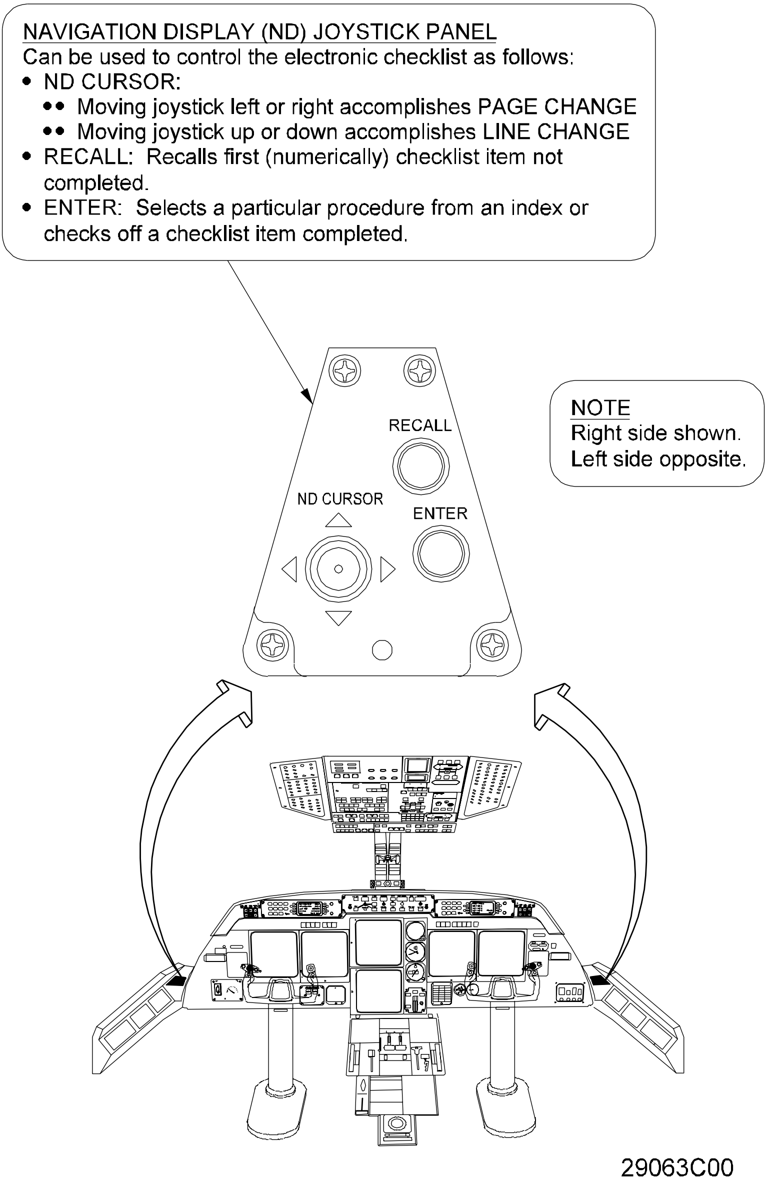


Figure 16. Navigation Display Joystick Panel

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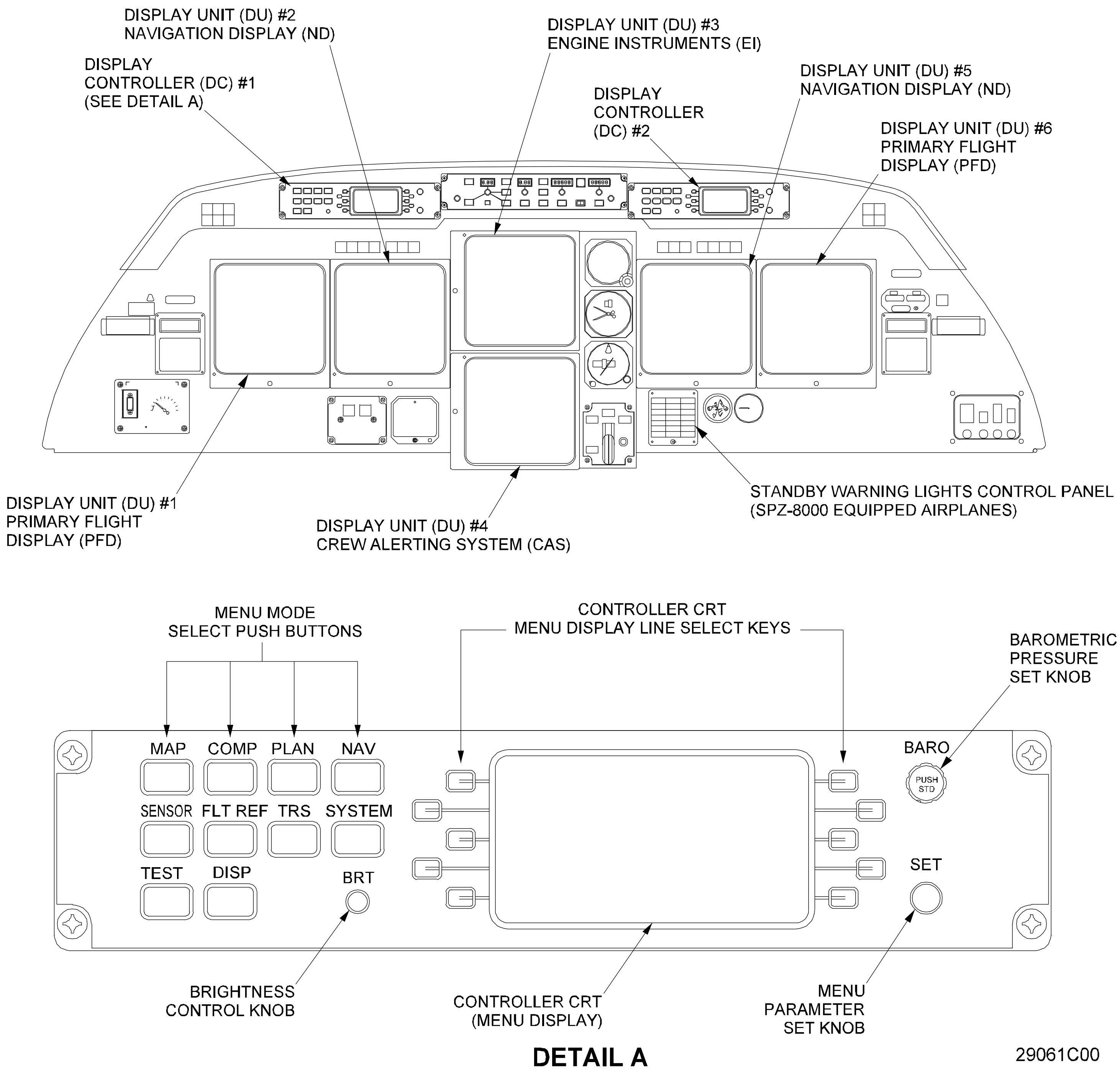


Figure 17. Display Units and Display Controllers

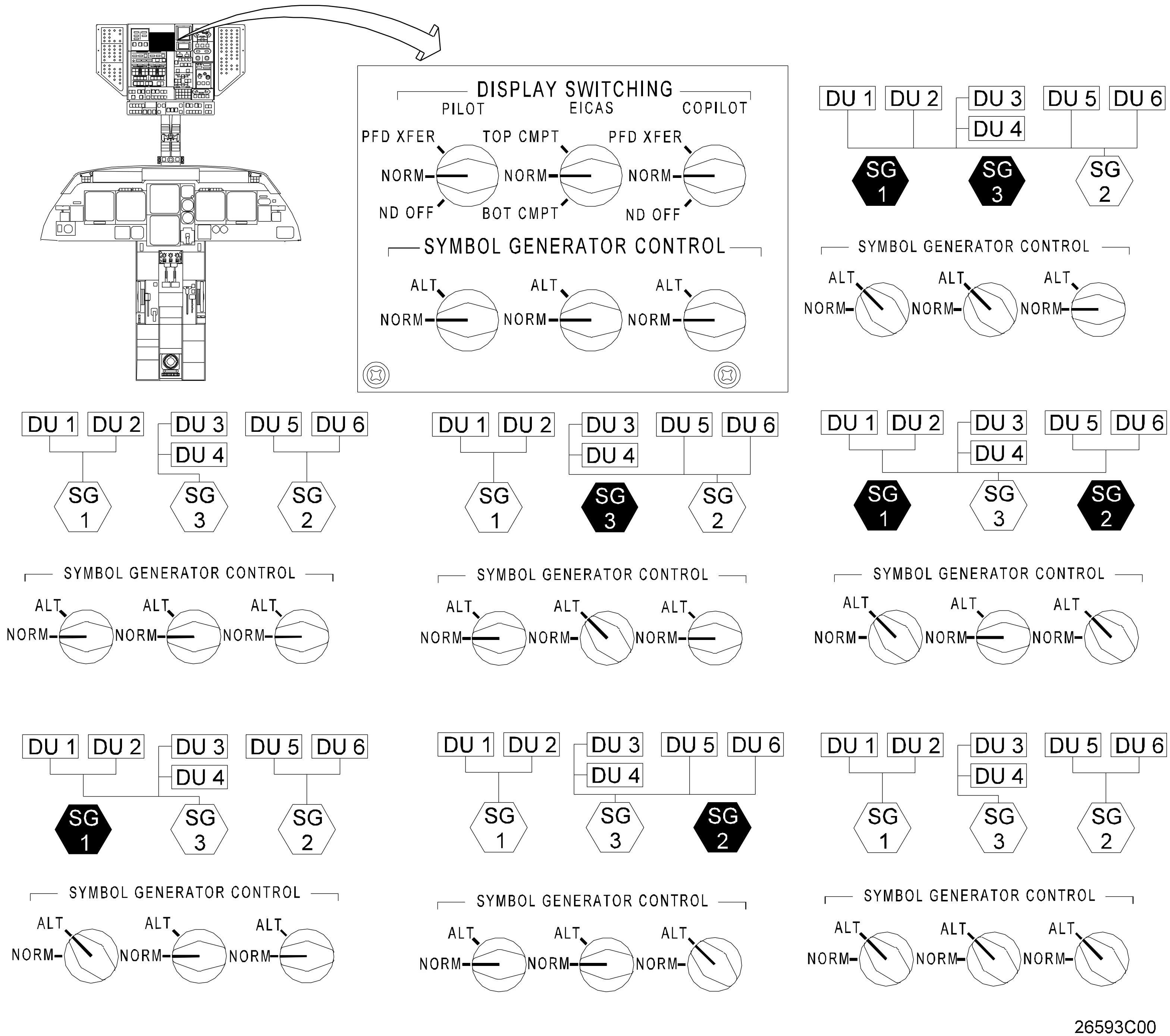


Figure 18. SPZ-8000 Display Switching and Display System Control Panel

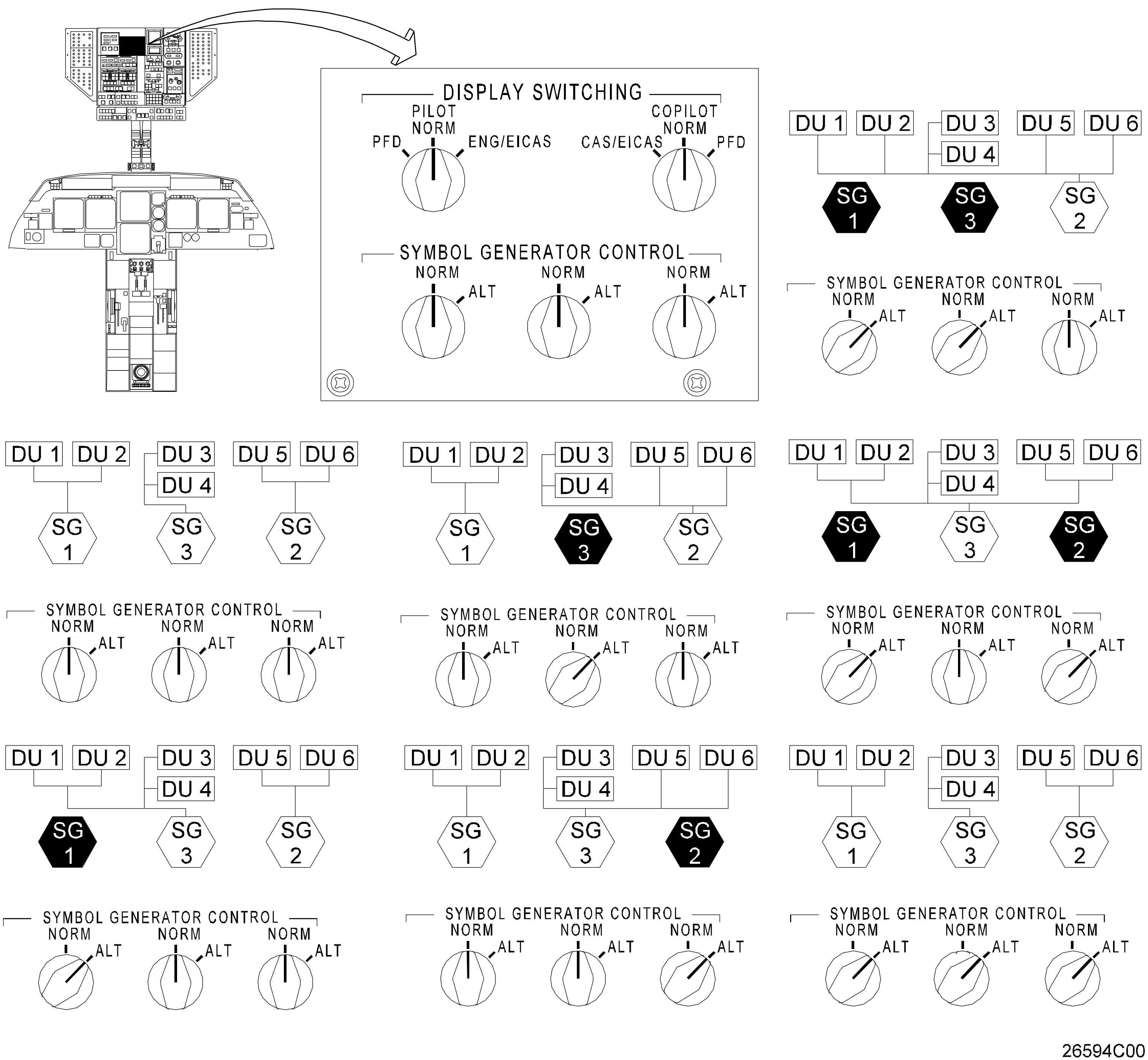


Figure 19. SPZ-8400 Display Switching and Display System Control Panel

2A-32-10: General Description

The Gulfstream IV landing gear has dual wheels on both the main and steerable nose gear. The main gear retract laterally into wheel wells on either side of the fuselage keel at the wing root area. The auto-centered nose gear retracts forward into a wheel well beneath the cockpit. Integrated gear doors close upon completion of gear extension and retraction providing an aerodynamically conformal fuselage surface.

Landing gear operation normally uses 3000 psi pressure from the Combined hydraulic system. Extension and retraction selections are made with the cockpit landing gear handle that is mechanically linked to the landing gear selector valve. The selector valve directs hydraulic pressure in a defined sequence to open the gear doors, extend or retract the gear and close the doors when the gear have reached the selected position. If Combined hydraulic system pressure is not available, the Utility system pressure may be used for landing gear operation, provided adequate fluid remains in the Combined system. In the event that Combined system fluid is lost, a single-use emergency gear extension system operated with a 3000 psi nitrogen gas bottle will extend the landing gear, but leave the gear doors open. The landing gear operating system is shown in Figure 1.

Landing gear position is indicated on the gear control panel by a green light for each gear and a red light in the landing gear control handle. Illumination of the red landing gear handle light indicates a disagreement between the selected and actual position of one or more of the landing gear. The individual gear green lights come on when the corresponding landing gear is down and locked.

In addition to the red landing gear position disagreement light in the gear handle, a warning horn/klaxon will sound if the landing gear is not down and locked and the aircraft configuration, altitude and/or power setting approximates the landing configuration.

A proximity, or nutcracker switch is installed on each gear. As the weight of the aircraft compresses the shock struts of the landing gear, the nutcracker switches close completing circuits for relays to aircraft systems that only operate on the ground. Conversely, when airborne, the nutcraker switches open, providing an in-the-air signal to systems that operate only in flight.

Main landing gear brakes are air-cooled multiple carbon-fiber discs with anti-skid protection. Overheat protection for the tires/wheels is provided by fusible plugs in the wheels that melt, releasing tire pressure, if high temperature thresholds are exceeded. Brake temperatures are monitored (on aircraft S/Ns 1000 to 1155 with ASC 167 and S/N 1156 and subsequent) and indicated on cockpit displays.

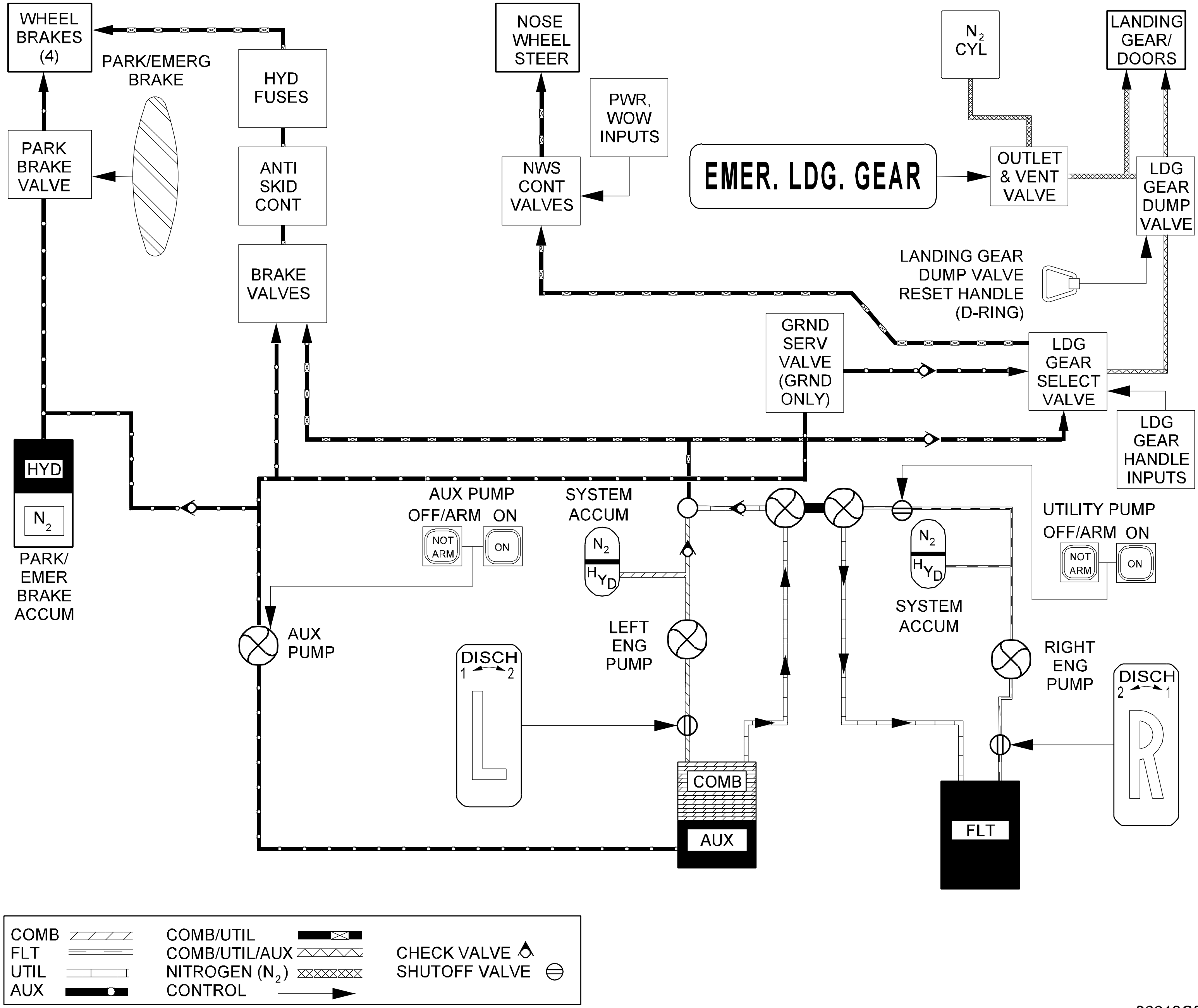
The Combined hydraulic system provides 3000 psi for normal brake operation. Hydraulic fuses are incorporated into the brake hydraulic lines and close to prevent fluid loss if a hydraulic line is damaged or cracked. If Combined/Utility system pressure is not available, but brake system hydraulic lines are intact, the Auxiliary hydraulic system can be used for brake operation. If no hydraulic pressure is available from aircraft systems, accumulator pressure from the Parking/Emergency brake system will provide approximately five to six brake applications without anti-skid protection.

Control of the brake system is either through a hydro-mechanical analog brake system (HMAB) for aircraft S/Ns 1000 - 1213 with ASC 307 and S/Ns 1214 and subsequent, or a brake-by-wire system for aircraft S/Ns 1000 - 1213 without ASC 307. In the analog system, cockpit brake pedal application is mechanically linked to the system brake metering valves that apply hydraulic pressure proportional to pedal depression. In the

brake-by-wire system, cockpit brake pedals incorporate linear variable differential transducers (LVDTs) that supply pedal deflection signals to a brake electronic control unit (ECU) that opens the left and right brake control valves proportional to LVDT commands. Brake feel is supplied by bungees that apply proportional resistance to pedal deflection. Anti-skid circuitry is integrated into the brake-by-wire system. Analog braking uses a separate anti-skid control box.

The Landing Gear System is composed of the following components.

- 2A-32-20: Landing Gear and Doors
- 2A-32-30: Extension and Retraction
- 2A-32-40: Wheels and Brakes
- 2A-32-50: Nose Wheel Steering



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Simplified Landing Gear Hydraulic System Diagram
Figure 1

2A-32-20: Landing Gear and Doors System

1. General Description

The major components of each main landing gear assembly are: a landing gear door, a shock strut, a sidebrace actuator/downlock mechanism and an uplock mechanism. Each of the main landing gear assemblies has dual wheels with integrated brakes mounted on an articulated trailing arm attached to the landing gear structural post. The pneumatic/hydraulic shock strut is attached between the wheel end of the articulated trailing arm and the top of the structural post. In flight with the landing gear down, the shock strut extends and the articulated trailing arm pivots downward, positioning the wheels for ground contact and compression of the shock strut. The major components of the main landing gear are shown in Figure 2 and Figure 3.

A side brace hydraulic actuator/downlock extends and retracts the main gear. In the extended position, hydraulic pressure positions mechanical downlocks that prevent gear retraction. The downlocks include mechanical springs loads that maintain the locked position if hydraulic pressure is lost. In gear retraction, hydraulic pressure compresses the springs, displaces the downlocks and allows gear retraction inward into the wheel well in the wing root next to the fuselage keel. An uplock roller mounted on the dual wheel axle assembly engages an uplock hook in the wheel well and rotates to latch the main gear in the uplocked position. The main landing gear door, hinged at the inboard side of the wheel well, closes after the gear retracts and mates with the fairing door panel mounted on the outside of the main gear assembly to seal the wheel well.

The nose gear assembly includes steerable dual wheels mounted on a shock strut attached to a trunnion and a hinged two-piece truss brace. A drag brace between the truss brace and the shock strut provides additional structural support and limits the extension of the nose gear shock strut. The truss brace and trunnion pivot axes allow the hydraulic actuator cylinder to retract the nose gear assembly forward and up and extend the gear aft and down. See the illustration in Figure 4.

When the nose gear is fully extended, the forward and aft sections of the truss brace move over center to lock the nose gear down. A downlock hydraulic cylinder extends an actuating arm holding the truss brace in the over center position. Springs are also incorporated in the over center downlock to maintain the over center lock in the event of hydraulic failure. When the nose gear retracts, the downlock actuator arm moves aft, allowing the forward and aft portions of the truss brace to fold at the hinge point. As the nose gear reaches the fully retracted position, a roller attached to the dual wheel axle assembly pushes aside a spring latch and engages an uplock hook, with the spring latch then falling into place securing the roller into the uplock hook. A hydraulically actuated pushrod then rotates the uplock hook upward to provide positive nose gear latching.

Two symmetrical nose wheel doors, hinged on the outside of the forward nose wheel well and mechanically linked together, operate in conjunction with the nose gear. Timing valves integrated into the extension/retraction cycle operate the hydraulic door actuating cylinder in the correct sequence. When the nose gear is retracted and seated in the uplock hook, the doors close to cover the forward wheel well, with the fairing panel attached to the rear of the nose gear drag brace enclosing the aft portion of the wheel well.

Indication of landing gear position is provided to the flight crew through lights on the landing gear control panel, an aural warning horn/klaxon, and by messages on the Crew Alerting System (CAS) in the event of malfunction.

2. Description of Subsystems, Units and Components

A. Landing Gear Doors

The landing gear doors are shaped to conform to the surrounding fuselage areas, providing an aerodynamically smooth covering of the wheel well areas when closed. The main landing gear wheel well is enclosed by a single door in conjunction with a fairing panel attached to the outside of the gear strut. The nose gear wheel well is enclosed by two symmetrical curved doors and a fairing panel attached to the aft side of the nose gear. When the landing gear is fully extended, the gear doors close to reduce drag and air noise. All gear door movement is sequenced by mechanical linkages operating hydraulic actuators to coordinate actuation with landing gear extension and retraction.

Procedures exist to open and close the doors on the ground in order to accomplish maintenance tasks. Hydraulic power from the Auxiliary system and operation of a ground service valve, located adjacent to the nose wheel well, allow movement of the gear doors by pressurizing the landing gear on the ground. The ground service valve is shown in Figure 5.

B. Shock Struts

Standard oleo-pneumatic (gaseous nitrogen and hydraulic fluid) shock struts are incorporated into each gear assembly. The shock struts consist of a movable stainless steel piston within a cylinder containing hydraulic fluid pressurized with gaseous nitrogen. O-ring seals maintain strut pressure and allow the movement of the steel piston. The struts absorb the shock of landing, and provide damping during taxi, takeoff and landing rollout. An air/oil filler valve is provided at the top of each strut for servicing. During normal operations, approximately three to five inches of inner cylinder chrome is exposed at the bottom of each main landing gear strut, depending on aircraft gross weight and outside air temperature. A placard attached to upper portion of the landing gear strut indicates the correct extension for ambient conditions.

C. Extension / Downlock Mechanisms

(1) Main Landing Gear Sidebrace Actuator / Downlock Mechanism

The sidebrace hydraulic piston and actuator arm extends the main landing gear outboard and down from the wheel well and locks the landing gear in the down position using internal locking keys located in the lower end of actuator cylinder. The keys slide into annular slots when the sidebrace actuator reaches full extension. A lock ram positioned by down hydraulic pressure and a spring maintain the keys in the slots preventing any further actuator movement. Spring pressure is necessary to maintain the keys in the locked position in the event of hydraulic pressure failure. The movement of the keys into the slots also provides the input for the downlock microswitch that provides the cockpit indication that the gear is down and locked. When the gear is selected up, hydraulic pressure is ported to the other side of the lock ram, displacing the keys from the slots and overriding spring pressure, allowing the actuator to move in the

retract direction.

The sidebrace actuators have provisions for inserting ground lock pins to maintain the main landing gear in the extended position during ground operations. See the illustration in Figure 6.

(2) **Nose Landing Gear Extend / Retract Actuator**

Unlike the main landing gear sidebrace actuator that pushes the main gear into the extended position, the nose gear actuator pulls the nose gear aft and down to the extended position. One end of the actuator is attached to the aft structural wall of the nose wheel well. The other end is attached to the hinged nose gear trunnion that pivots aft into the extended position.

(3) **Nose Landing Gear Downlock**

Nose gear downlock protection is provided by the over center position of the hinged two part truss brace with integrated downlock actuator and spring. The forward and aft sections of the truss brace unfold as the nose gear extends. As the truss brace unfolds, it forces open the “C” shaped suitcase springs. When the nose gear is fully extended, a hydraulic downlock actuator and linkage pulls the truss brace slightly (approximately 0.050 inch) over center at the hinge point. The truss brace hinge is also held in the over center position by the energy of the “C” shaped suitcase springs that close slightly in the over center position. The springs are necessary to maintain the nose gear in the downlocked position in the event of hydraulic pressure failure. Once locked into the over center position, hydraulic pressure on the retract side of the downlock actuator is required to push the truss brace past the over center position and overcome spring pressure.

The truss brace hinge has a provision for inserting a down lock pin to secure the nose gear in the extended position. See the illustration in Figure 6.

(4) **Downlock Switches**

Microswitches installed on the main and nose landing gear provide downlock electrical signals for the landing gear position indicators on the cockpit control panel. On the main landing gear, when the internal locking keys move into the annular groves of the actuator in the fully extended position, a plunger with an integrated cam moves, depressing the contact of the downlock microswitch.

The downlock microswitch on the nose gear is located on the truss brace. As the truss brace unfolds to the over center position the contact on the nose gear microswitch is depressed sending a downlocked signal to the landing gear control panel.

D. Uplock Mechanisms

(1) **Main and Nose Landing Gear Uplock Mechanisms**

The uplock mechanisms of the main and nose landing gear operate in an identical manner, although the components are shaped somewhat differently due to the direction of movement of the nose and main landing gear in the extension/retraction cycle. Operation of the main gear uplocks is described as typical of all landing gear installations. The landing gear uplocks are shown in Figure 2 and

Figure 4.

When the main landing gear is fully retracted into the wheel well, the gear is locked into position by a hook and latch that engage an uplock roller mounted on the front of the dual wheel axle housing. The uplock hook and latch mechanism is mounted on the interior forward side of the wheel well. The uplock hook is mechanically positioned by the uplock roller on the landing gear and the uplock latch is positioned by dual springs. (The uplock hook also has dual springs that will maintain the hook in the locked position in event of hydraulic failure.) The integrated latch assembly mechanically blocks movement of the uplock hook until the latch is moved aside by the gear uplock roller as the gear reaches the retracted position. When the latch is moved by the gear uplock roller from the position blocking movement of the uplock hook, the uplock roller enters the concave portion of the uplock hook as the hook pivots into a cradling position. Spring tension on the latch drops the latch into position behind the uplock roller, securing the uplock roller in the uplock hook. The action of the spring held latch counteracts any tendency for the uplock roller to rebound out of the uplock hook and retains the uplock roller in the grasp of the uplock hook after system hydraulic pressure is released.

CAUTION

WHEN THE LANDING GEAR IS EXTENDED, THE GEAR UPLOCK HOOKS ARE OPEN AND HYDRAULIC PRESSURE IS REMOVED FROM THE UPLOCK ACTUATORS. IT IS POSSIBLE TO MANUALLY PUSH UP THE UPLOCK LATCH AND ROTATE THE UPLOCK HOOKS TO THE LOCKED POSITION. SUBSEQUENT RETRACTION OF THE GEAR WILL RESULT IN THE LANDING GEAR DOORS CLOSING PREMATURELY, AND THE LANDING GEAR WILL IMPACT THE DOORS, FAILING TO RETRACT. ALL UPLOCKS SHOULD BE VERIFIED OPEN DURING PREFLIGHT WALK AROUND INSPECTION.

3. Limitations

A. Maximum Tire Speeds

Maximum tire speeds for installed configuration are shown in the following table:

Aircraft Serial Number	Configuration	Nose Tire	Main Tire	Overall Limit
1000-1213	Without ASC 190 / 266	182 kts / 210 mph	182 kts / 210 mph	182 kts / 210 mph
1000-1213	With ASC 266	182 kts / 210 mph	195 kts / 225 mph	182 kts / 210 mph
1000-1213	With ASC 190	195 kts / 225 mph	195 kts / 225 mph	195 kts / 225 mph

Aircraft Serial Number	Configuration	Nose Tire	Main Tire	Overall Limit
1214 and subsequent	Production	195 kts / 225 mph	195 kts / 225 mph	195 kts / 225 mph

B. Nose Strut Servicing

Recommended nose strut servicing is shown in the following table.

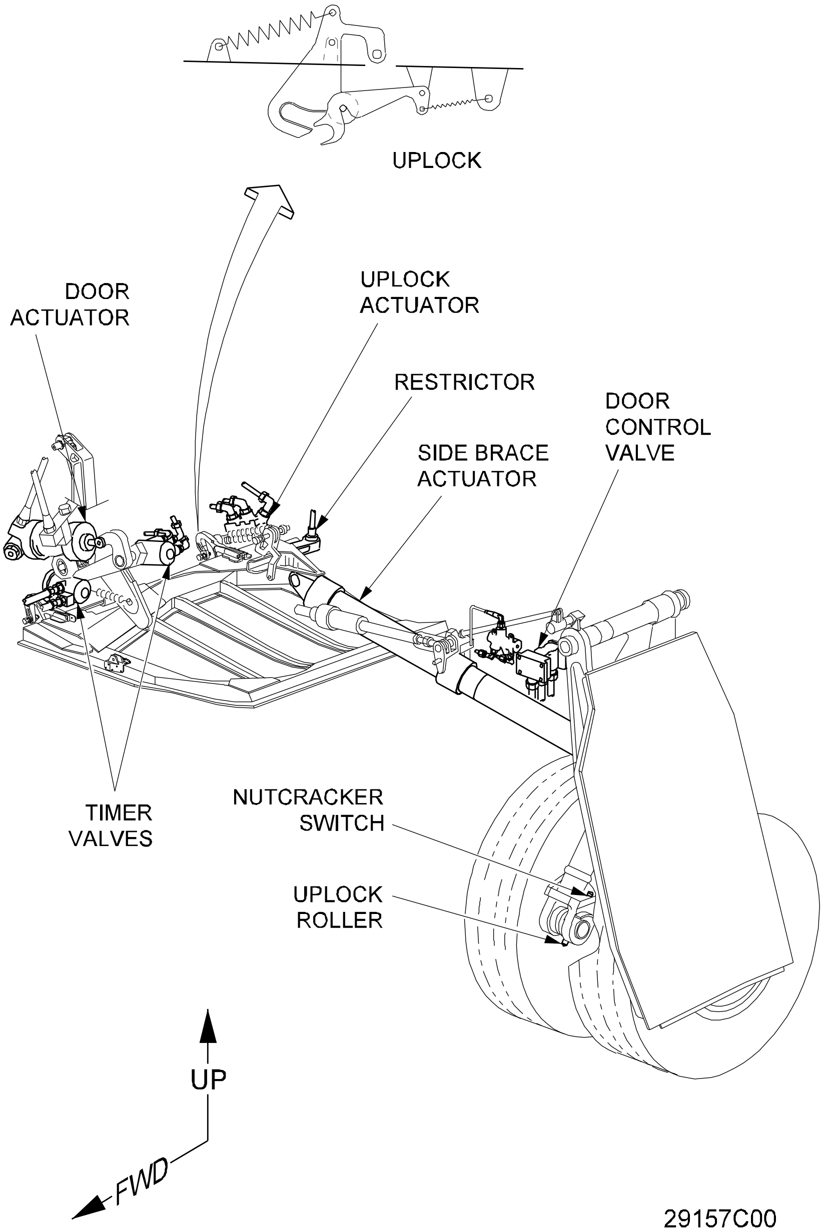
Service Temp:	Cold Day Service (-40° F / -40° C)	Std Day Service (70° F / 21° C)	Hot Day Service (110° F / 43° C)
Piston Extension (inches)	**Strut Pressure (psig)**		
13.04	188	224	246
11.24	220	264	289
9.46	260	312	341
8.34	300	365	399
7.34	350	419	457
6.54	400	475	518
5.84	450	537	586
5.30	500	596	651
4.94	550	644	703
4.54	600	706	770
4.24	650	761	830
3.94	700	825	900
3.64	750	901	982
3.44	800	959	1045
3.24	850	1025	1117
3.04	900	1100	1198
2.94	950	1142	1244
2.84	1000	1187	1293
2.74	1050	1236	1346
2.64	1100	1289	1403
2.44	1150	1408	1533
2.34	1200	1477	1607

NOTE:

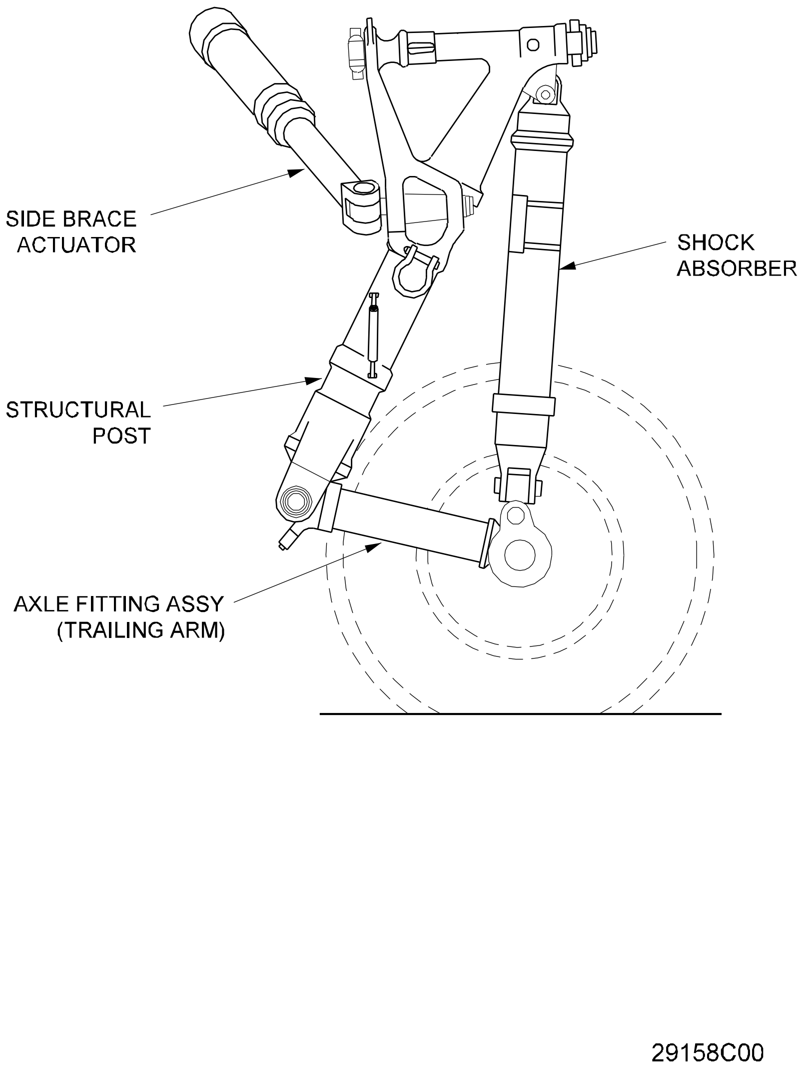
(1) Piston extension is measured from bottom surface of gland nut to black circumferential line on piston / axle.

(2) This table is used only for checking and adjusting strut service pressure and should not be used for full strut servicing.

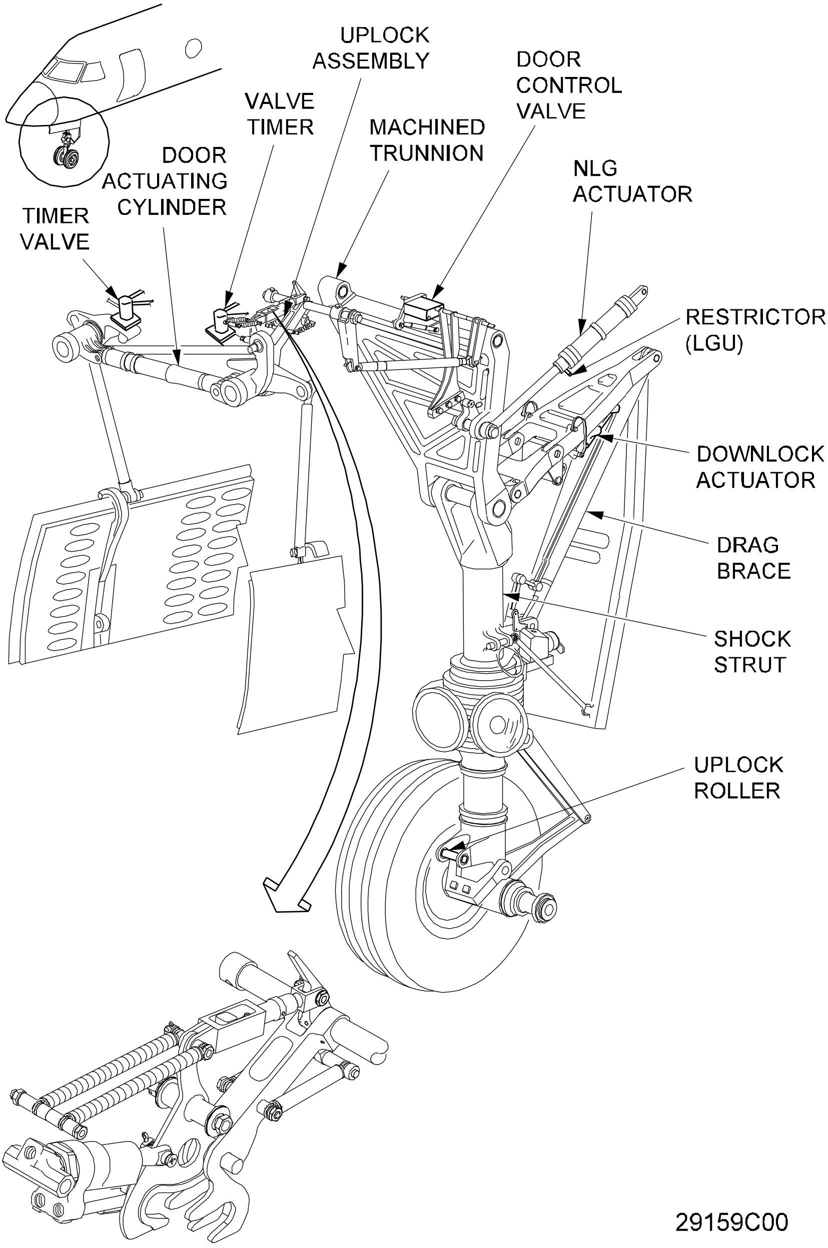
(3) Interpolate between temperatures for proper service pressure.



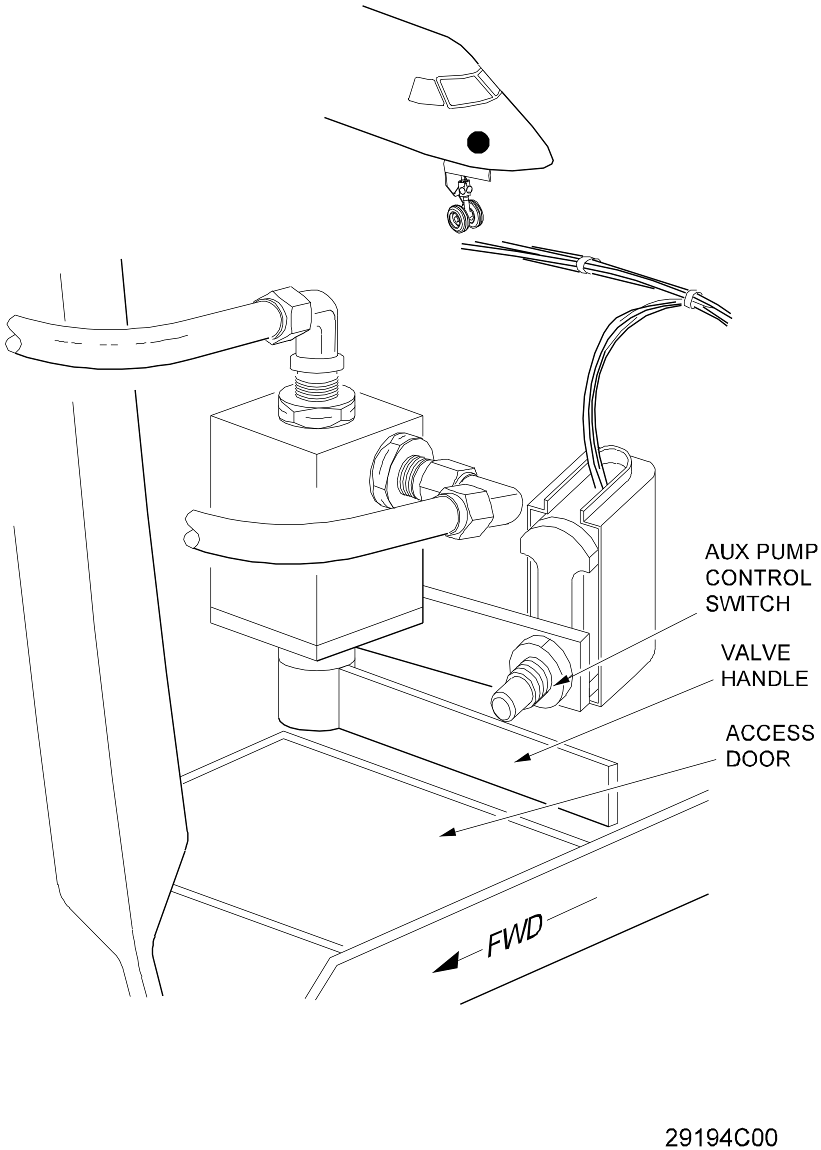
Main Landing Gear Components
Figure 2



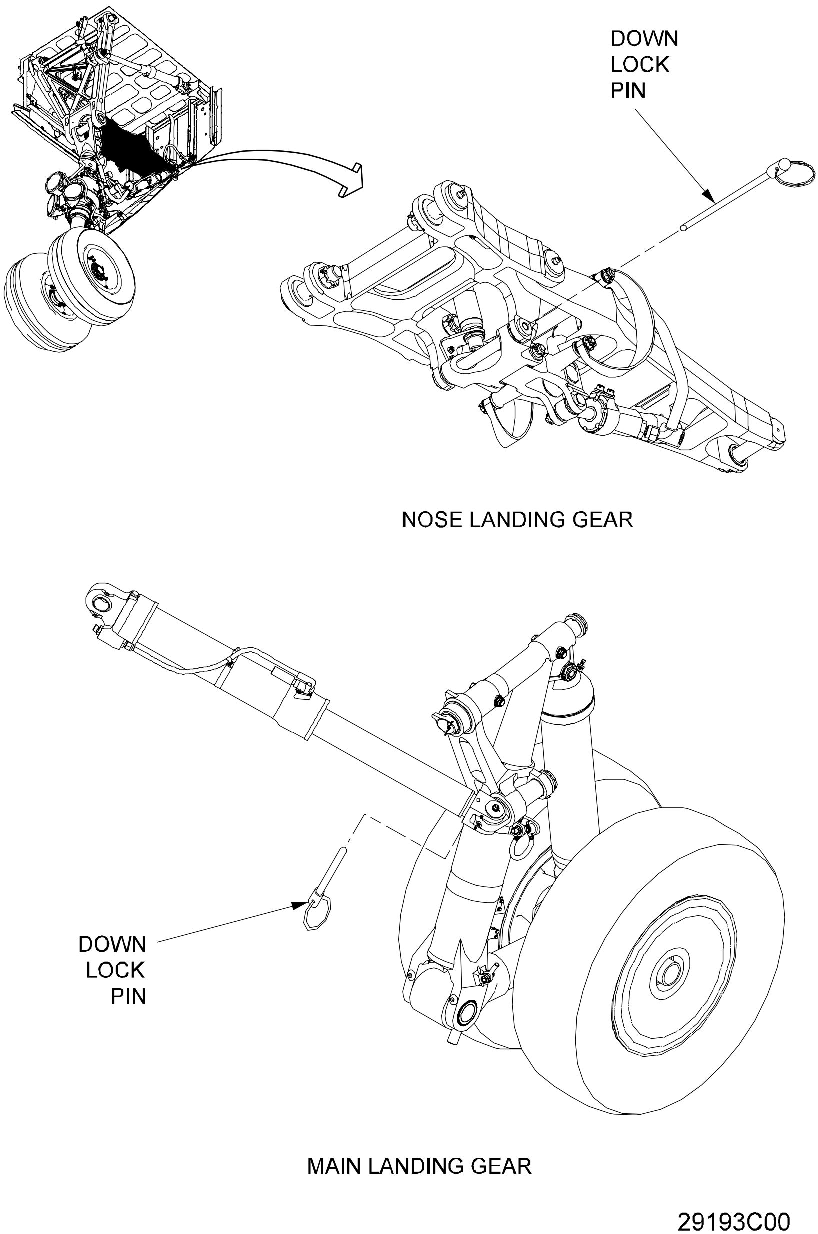
Main Landing Gear Structure
Figure 3



Nose Landing Gear Components
Figure 4



Ground Service Valve
Figure 5



Landing Gear Safety Pins
Figure 6

END

2A-32-30: Extension and Retraction System

1. General Description

Landing gear extension and retraction is controlled with the landing gear lever on the copilot side of the center instrument panel. The landing gear lever is connected to a mechanical linkage behind the instrument panel that extends to the landing gear selector valve mounted in the unpressurized nose section. The linkage is shown in Figure 7. Moving the landing gear handle mechanically opens and closes ports in the selector valve, routing Combined system hydraulic pressure to actuators for the main and nose gear and doors. If Combined system pressure is not available, but hydraulic lines and fluid quantity are intact, Utility system pressure will operate the landing gear using normal control linkages. If no hydraulic pressure is available, but system lines remain intact, the landing gear may be lowered by pressurizing the hydraulic lines with nitrogen gas stored in an emergency bottle located in the nose wheel well. The emergency extension system may only be used once and cannot retract the landing gear.

A nutcracker (squat) switch is installed on each landing gear. The nutcracker switches are closed when the aircraft is on the ground and the pneumatic-oleo struts are compressed with the aircraft weight. For this reason the switches are sometimes referred to as weight-on-wheels switches. The switches provide sensing information to aircraft systems / sub-systems that operate only in the air or only on the ground.

To facilitate maintenance on the landing gear or on components located in the wheel wells, the Auxiliary hydraulic system may be used to operate the landing gear system and/or the landing gear doors. A ground service valve, located inside a hinged panel near the nose wheel well, is positioned and held with safety locking pin to port Auxiliary system pressure to operate the doors only. If the aircraft is raised on jacks, the Auxiliary system may also be used to raise and lower the landing gear.

2. Description of Subsystems, Units and Components

A. Landing Gear Control Panel

The landing gear control panel, located on the copilot side of the center instrument panel, contains the gear lever, position and warning indicators and safety features. See Figure 8. Landing gear position and warning lights on the panel are powered by the Emergency DC bus and may be dimmed and tested with switches on the cockpit side consoles. The landing gear warning horn/klaxon is powered by the Essential DC bus.

(1) Landing Gear Lever:

The landing gear lever is selected into either the UP or DOWN detents on the control panel. To move the lever from one selection to the other, the lever must first be moved to the left to clear the detent. Lever movement is mechanically linked to the landing gear hydraulic selector valve in the nose compartment behind the radome. The selector valve routes Combined or Utility system pressure to the extend or retract sides of actuators for the landing gear and doors. Lever position has no effect on operation of the emergency landing gear extension system, but the position of the lever is integral to a

correct landing gear position indication.

The transparent wheel shaped handle of the landing gear lever has two clear bulbs for night illumination and a red bulb for annunciation of disagreement between gear lever position and the selected position of the landing gear (when gear selected down) or landing gear doors not closed (when gear selected up). With the gear lever up, the red light illuminates until all landing gear are up and the gear doors closed. With the gear lever down, the red light illuminates until all landing gear are down and locked. The warning light circuits are configured such that a ground is supplied to the red light, powering the light, whenever there is a difference between the gear lever selected position and proximity switches on the landing gear downlocks (DOWN) or proximity switches on the gear doors (UP). Agreement between selected and actual position removes the ground from the circuit.

(2) Ground Safety Lock and LOCK RELEASE Button:

On the ground with weight on the aircraft wheels (nutcracker switches on both main landing gear compressed), the landing gear lever is blocked from movement out of the DOWN detent by a solenoid actuated pin. The pin prevents inadvertent retraction of the landing gear due to unintentional movement of the landing gear lever. If the blocking solenoid or nutcracker switches malfunction during takeoff in the transition to the weight off wheels mode, preventing movement of the landing gear lever to the UP position, pressing the LOCK RELEASE button will manually move the blocking pin clear of the landing gear lever.

(3) Down and Locked Lights:

Three green light modules, one for each landing gear, are installed above the landing gear lever. Each light is an independent circuit, illuminating only when the associated gear downlock proximity switch contact is closed. The green lights do not annunciate landing gear door uplock engagement, and are also separate from the red warning light in the landing gear lever handle. Thus, if a landing gear does not fully extend and engage the downlock switch, a red light will illuminate in the gear lever handle and the green light associated with the malfunctioning landing gear will not illuminate, identifying the malfunctioning gear. However, if a landing gear does not fully retract (downlock not engaged, but gear door uplock not engaged) the red light in the landing gear lever handle will illuminate, but none of the green lights will illuminate.

(4) Gear Warning Horn and HORN SILENCE Switchlight:

If the landing gear are not down and locked during flight conditions associated with landing, a warning horn (aircraft with SPZ-8000 are equipped with a horn, aircraft with SPZ-8400 are equipped with a klaxon) will sound, alerting the crew that the landing gear is not in the correct position. In some conditions, the gear warning horn may be silenced by pressing the HORN SILENCE switchlight on the landing gear control panel to the left of the landing gear lever, or by pressing either of the HORN SILENCE buttons on the outside of the power levers. In other conditions, the warning horn can be silenced

only by configuring the aircraft correctly for landing. If the HORN SILENCE switchlight or buttons have been pressed, the HORN SILENCE switchlight will illuminate.

The following conditions will sound the landing gear warning horn:

- The aircraft at or below 1,200 feet radio altitude and a power lever retarded to 70% or less and all landing gear not down and locked. The warning horn may be silenced in this condition. If the power lever(s) is subsequently advanced to more than 70%, the HORN SILENCE switchlight will extinguish and the horn warning circuit will rearm.
- The flaps selected to more than 22° and all landing gear not down and locked regardless of altitude. The warning horn cannot be silenced and will sound until the aircraft configuration is corrected.

B. Normal Retraction and Extension System

(1) Retraction Sequencing

(a) Landing gear lever is placed in UP position

(b) Red light in landing gear lever handle is illuminated

(c) Landing gear selector valve is moved to RETRACT position

(d) Hydraulic pressure is routed to OPEN side of landing gear door actuators

(e) Landing gear doors open

(f) The timer valves route hydraulic pressure to the RETRACT side of the main landing gear sidebrace actuators, unlocking the internal downlock keys, and to the nose gear actuator and release the nose gear downlock actuator

(g) The nose gear downlock actuator unlocks

(h) Three green DOWN AND LOCKED lights on landing gear control panel are extinguished

(i) Landing gear retracts and locks into uplock hooks

(j) Hydraulic pressure is routed to CLOSE side of landing gear door actuators

(k) Landing gear doors close

(l) Red light in landing gear lever handle is extinguished

(2) Extension Sequencing

(a) Landing gear lever is placed in DOWN position

(b) Red light in landing gear lever handle is illuminated

(c) Landing gear selector valve is moved to EXTEND position

(d) Hydraulic pressure is routed to OPEN side of landing gear door actuators

(e) Landing gear doors open

(f) Hydraulic pressure is routed to UNLOCK side of landing gear uplock actuators

(g) Uplock hooks unlock

(h) Hydraulic pressure is routed to EXTEND side of main landing gear sidebrace actuators and nose landing gear extend / retract actuator

(i) Landing gear extends down and downlocks engage

(j) Three green DOWN AND LOCKED lights on landing gear control panel are illuminated

(k) Red light in landing gear lever handle is extinguished

(l) Hydraulic pressure is routed to CLOSE side of landing gear door actuators

(m) Landing gear doors close

C. Emergency Extension System

The landing gear emergency extension system will operate only if the hydraulic lines for gear extension are intact. The system substitutes gas pressure for hydraulic pressure to move the actuators of the landing gear components. Emergency gear extension takes approximately six seconds to extend and lock the landing gear. Emergency extension nitrogen pressure will not close the landing gear doors after the landing gear are down and locked, the gear doors remain open. Once activated, emergency system pressure remains in the three actuators of each landing gear (uplock release, door open and gear extend) until the pressure is vented overboard when the emergency extension T-handle is stowed. Controls for the emergency extension system are shown in Figure 9.

(1) Units and Components

(a) Nitrogen Bottle

Pressurized nitrogen used for emergency landing gear extension is stored in a bottle located on the left side of the nose wheel well. A fully charged bottle contains 150 cubic inches of nitrogen pressurized to 3,000 psi at 70° F. The bottle has a pressure relief valve set at 3,750 psi to prevent damage to aircraft components should gas pressure exceed the structural limits of the bottle. The location of the emergency extension bottle is shown in Figure 10.

(b) Gear Emergency Extension Cable and T-Handle

The emergency extension T-handle is located on the forward end of the copilot side console and is labeled EMER LDG GEAR. The T-handle is connected by a flexible cable to the pressurized nitrogen bottle. Pulling the T-handle to full extension opens the bottle outlet valve, releasing compressed nitrogen into the landing gear hydraulic lines. If the emergency T-handle is subsequently returned to the fully stowed position, the nitrogen bottle vent port opens, and nitrogen pressure is discharged through an overboard vent.

(c) Dump Valves

Compressed nitrogen enters the hydraulic gear extension lines through dump valves integrated into the normal hydraulic system pressure lines. Nitrogen pressure activates the dump valves to close off the normal hydraulic system pressure lines and return hydraulic fluid present in the gear

extend, gear retract and gear door lines and actuators to the system reservoir. This action prevents any hydraulic lock in the landing gear actuators. Nitrogen then pressurizes the unlock port of the three gear uplock actuators, the open port of the three door actuators and the extend port of the three landing gear actuators.

(d) Dump Valve Reset Handle

A reset handle (D-ring), shown in Figure 9, is located on the inside of the copilot side console and labeled EMERGENCY LANDING GEAR DUMP VALVE RESET. The D-ring is mechanically connected to the hydraulic system dump valves. Pulling the D-ring returns the dump valves to the normal hydraulic system setting. If emergency extension of the landing gear with the pressurized nitrogen bottle has been satisfactorily completed, the landing gear would normally be left extended until a landing could be made. If however, the landing gear must be retracted and hydraulic fluid and Combined or Utility system pressure is available, resetting the dump valves will allow retraction of the landing gear.

(2) Landing Gear Emergency Extension Sequence

(a) Landing gear lever is placed in DOWN position

(b) Red light in landing gear lever handle is illuminated

(c) EMER LDG GEAR T-handle is pulled to full limit of travel

(d) Compressed nitrogen is released to landing gear dump valves

(e) Dump valves isolate landing gear extend lines from remainder of hydraulic system and open returns for hydraulic fluid in actuators to hydraulic reservoirs

(f) Pressurized nitrogen is routed to OPEN side of landing gear door actuators, UNLOCK side of landing gear uplock actuators, EXTEND side of main landing gear sidebrace actuators and nose landing gear extend / retract actuator

(g) Landing gear doors open

(h) Uplock actuators unlock

(i) Landing gear extends down and locks

(j) Three green DOWN AND LOCKED lights on landing gear control panel illuminate

(k) Red light in landing gear lever handle extinguishes

(l) Landing gear doors remain open

D. Nutcracker System

The nutcracker (squat) switch system provides AIR or GROUND sensing to aircraft systems and components. A nutcracker switch is installed on each landing gear. The nutcracker switch contacts are depressed when the landing gear oleo-pneumatic struts are compressed by the weight of the aircraft on the ground. When the aircraft is in flight, the struts extend, releasing pressure on the nutcracker switches and opening the switch

contacts. The nutcracker switch system is connected to seventeen relays. Circuits may be opened or closed corresponding to AIR or GROUND states of operation by wiring systems and components through the nutcracker switch relays.

The left and right main landing gear nutcracker switch relays are powered by the Essential DC bus. The nose gear nutcracker switch relay is powered by the Emergency DC bus. The main gear nutcracker switches are incorporated into a test circuit that verifies switch integrity. (The nose landing gear nutcracker switch is not tested.) The test switch is located on the center pedestal aft of the throttle quadrant. Each nutcracker switch has a dedicated circuit breaker. If a nutcracker switch fails in the ground (closed) position, pulling the respective circuit breaker will change the switch input to relays to the air position.

Nutcracker switch inputs from the main landing gear control the landing gear lever safety lock solenoid and the nose landing gear nutcracker switch weight-on-wheels signal is necessary for nosewheel steering. For a list of nutcracker system relay inputs to aircraft systems and components, see the following table.

Ground Mode (Weight on Wheels)	**Flight Mode (Weight off Wheels)**
Thrust reverser operation Engine ground idle Ground spoiler operation APU control Air flow control Cockpit clocks FMCS EDS Autopilot Cockpit voice recorder ILS Engine starting Speedbrake / Flap alarm Cabin pressure control EICAS DAU / FWC #1 EICAS DAU / FWC #2 G-meter Stall barrier	Thrust reverser REVERSE ALERT light Engine flight idle Angle of attack indication Gear lever downlock solenoid pin retract Pitot heat VHF nav Brake-by-wire Flight data recorder

NOTE:

Failure of the nutcracker system will result in inoperative and/or abnormal operation of aircraft systems, depending upon whether the nutcracker system fails in the ground or flight mode. See the following table for nutcracker failure effects.

In flight with Nutcracker System failed in ground mode	On the ground with Nutcracker System failed in flight mode (Aircraft with ASC 166A)	On the ground with Nutcracker System failed in flight mode (Aircraft without ASC 166A)
Landing gear lever cannot be positioned to up unless the solenoid lock release button is pressed	Red REV UNLOCK light may illuminate when thrust reversers are used. Thrust reversers will stow at low ground speed.	Thrust reversers inoperative
Auto-pressurization will maintain 0.25 in differential, and cabin altitude will climb with aircraft. Manual pressurization is available.	Red ACFT CONFIG anunciation with speed brake deployment	Red ACFT CONFIG anunciation with speed brake deployment
Thrust reversers will operate in flight if power levers are retarded to idle.	Ground spoilers will deploy with wheel spin-up, but will stow at low ground speed.	Ground spoilers will deploy with wheel spin-up, but will stow at low ground speed.
If Ground Spoiler switch is armed, ground spoilers will deploy in flight if power levers are retarded to idle.	Anti-skid may not operate. All braking may be inoperative at low ground speed unless anti-skid is selected off.	Anti-skid may not operate. All braking may be inoperative at low ground speed unless anti-skid is selected off.
Stick shaker and stick pusher are inoperative	Engine idle set to 67% HP at low ground speed	Engine idle set to 67% HP at low ground speed
	APU bleed air inoperative	APU bleed air inoperative
	Pressurization outflow valve will not automatically open, valve must be manually opened.	Pressurization outflow valve will not automatically open, valve must be manually opened.
	Stick shaker and stick pusher remain armed.	Stick shaker and stick pusher remain armed.
	Landing gear lever handle not protected by lock release solenoid and gear may be retracted.	Landing gear lever handle not protected by lock release solenoid and gear may be retracted.
	Engines may not be restarted after shutdown	Engines may not be restarted after shutdown

3. Controls and Indications

A. Circuit Breakers (CBs)

Circuit Breaker Name	CB Panel	Location	Power Source
LDG GEAR POS IND	P	E-5 (1)	Emergency DC Bus
LDG WARN HORN	P	E-4	Essential DC Bus

Circuit Breaker Name	CB Panel	Location	Power Source
L NUTCRACKER	CPO	A-13	Essential DC Bus
NUTCRACKER	CPO	B-13	Essential DC Bus
R NUTCRACKER	CPO	C-13	Essential DC Bus
NUTCRACKER BAT PWR	CPO	D-13	Battery Bus (2)

NOTE(S):

(1) S/N 1000-1279; B-5 on S/N 1280 and subsequent.

(2) Aircraft with ASC 242.

B. Warning (Red) Messages and Annunciations

CAS Message	SWLP Indication	Cause or Meaning
ACFT CONFIGURATION	ACFT CONFIG	Landing Gear not down and locked with radar altitude ≤ 1200 ft and power lever < 70%, or Landing Gear not down and locked with Flaps > 22° at any altitude

NOTE:

See 2A-27-60, Flaps System for a description of the Configuration Warning System

C. Other Warning Annunciations

Annunciation	Cause or Meaning
Warning horn (SPZ-8000 equipped aircraft) or Klaxon™ hi-lo, hi-lo (SPZ-8400 equipped aircraft) sounds	Landing gear unsafe

D. Nutcracker Switch Test

The Nutcracker Switch Test button on the center pedestal is used to verify that the main landing gear nutcracker switches are in the air (weight-off-wheels) mode during flight (the landing gear retracted). Pushing the button should result in a green indication for the L (left) and R (right) strut nutcracker switches on the main landing gear, indicating the air mode. See the illustration in Figure 11. The switch should not be activated on the ground, since this would place the nutcracker switches in the air mode, interrupting the circuits to aircraft systems that are dependent upon ground (weight-on-wheels) mode for proper operation (this action would retract the solenoid pin from the landing gear lever, allowing the handle to be placed in the up position).

E. Brake Nutcracker Override Switch (Brake-by-Wire System Only)

During flight with the nutcracker switches in the air (weight-off-wheels) mode, hydraulic pressure is removed from the main landing gear brakes. Brake operation and correct brake pressure application may be tested in flight by depressing the Brake Nutcracker Override Switch on the center pedestal, shown in Figure 18. With the switch depressed, brake pressure may be applied with the landing gear retracted. With the EICAS selected to

the hydraulics system page (SPZ-8000) or the brakes system page (SPZ-8400) depressing either set of brake pedals will result in the applied brake pressure indicated on the brake system page, allowing verification of brake operation prior to landing.

4. Limitations

A. Landing Gear Extended Speed (V_{LE} / M_{LE})

Do not exceed 250 KCAS / 0.70 MT with landing gear extended (gear doors open or closed).

B. Landing Gear Operation Speeds (V_{LO} / M_{LO})

(1) Normal Operation:

Do not lower or raise landing gear at speeds in excess of 225 KCAS / 0.70 MT.

(2) Alternate Operation:

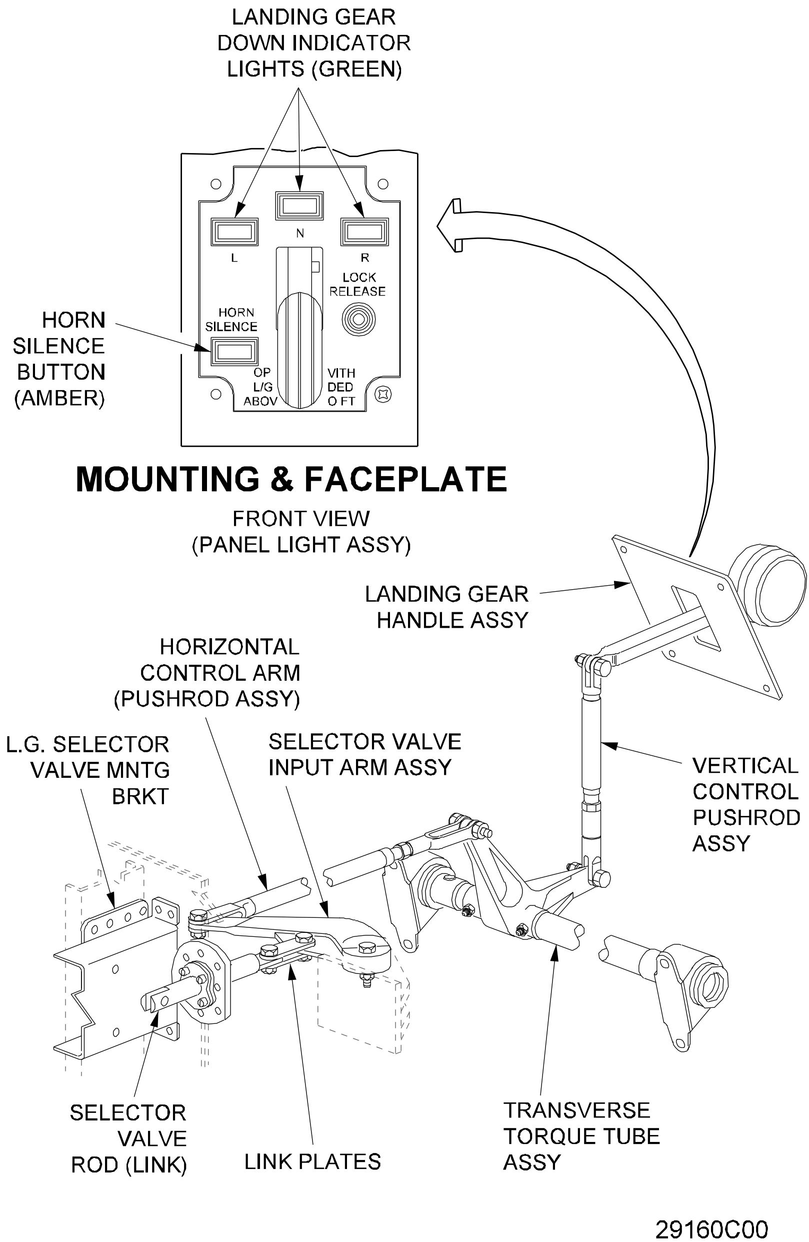
Do not lower landing gear using alternate system at speeds in excess of 175 KCAS.

C. Landing Gear Extension / Operation Altitudes

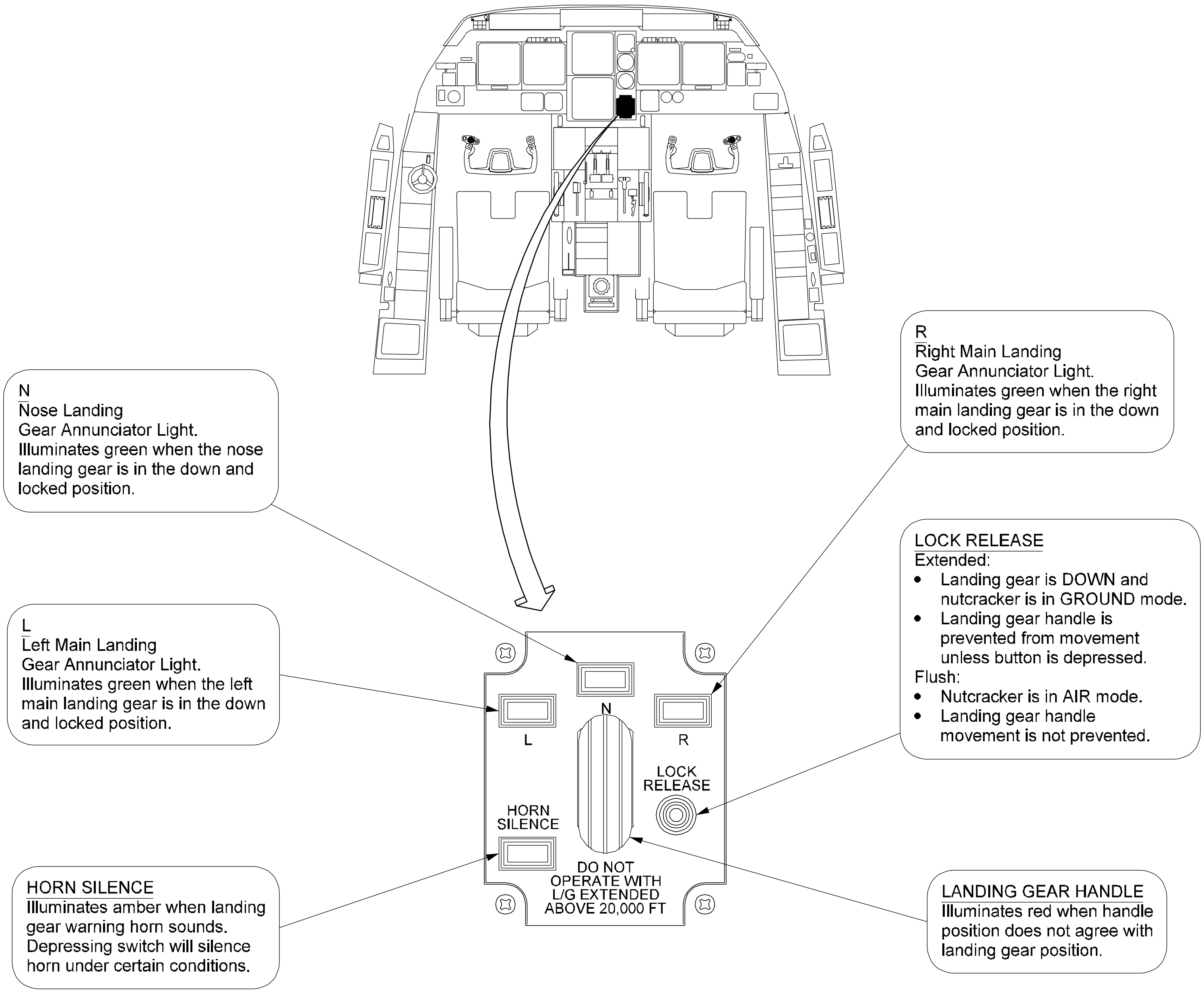
Maximum operating altitude for extending landing gear or flying with landing gear extended is 20,000 ft. MSL.

D. Speed Brakes Extension With Landing Gear Extended

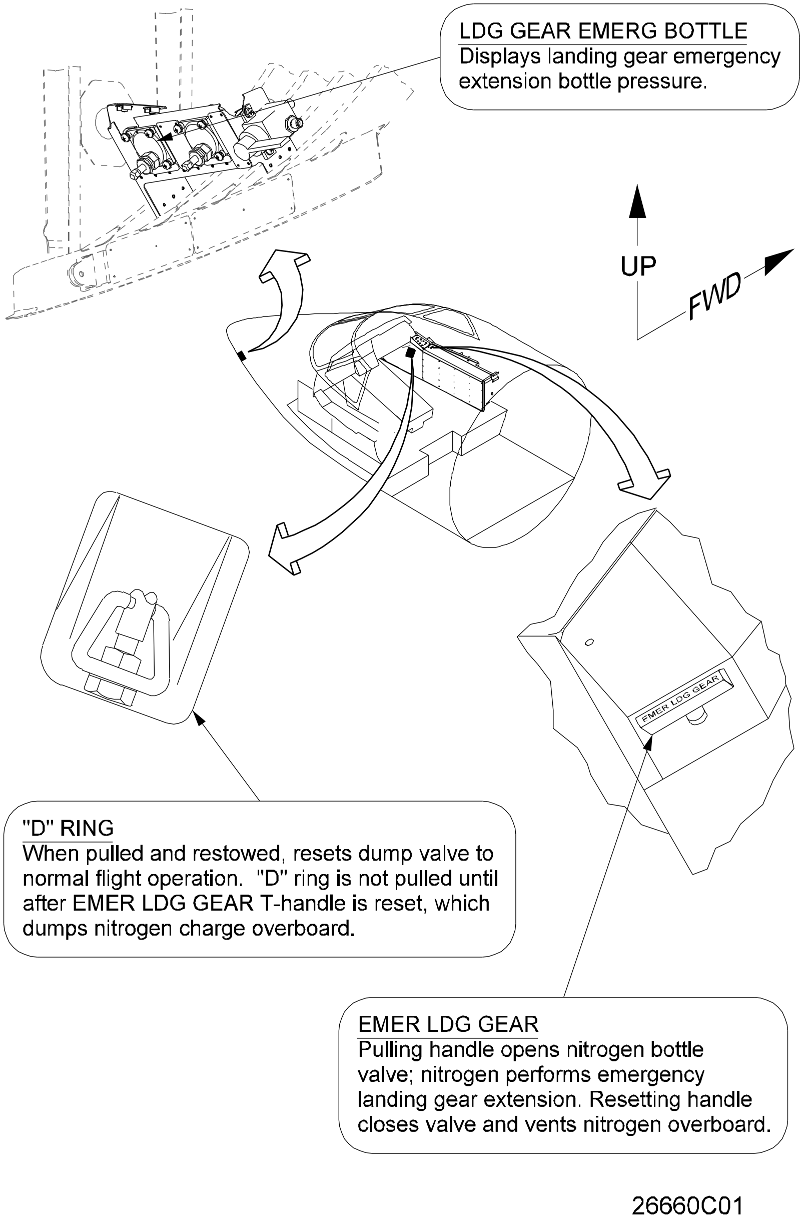
During flight, speed brakes are not to be extended with flaps set at 39°, or with the landing gear extended.



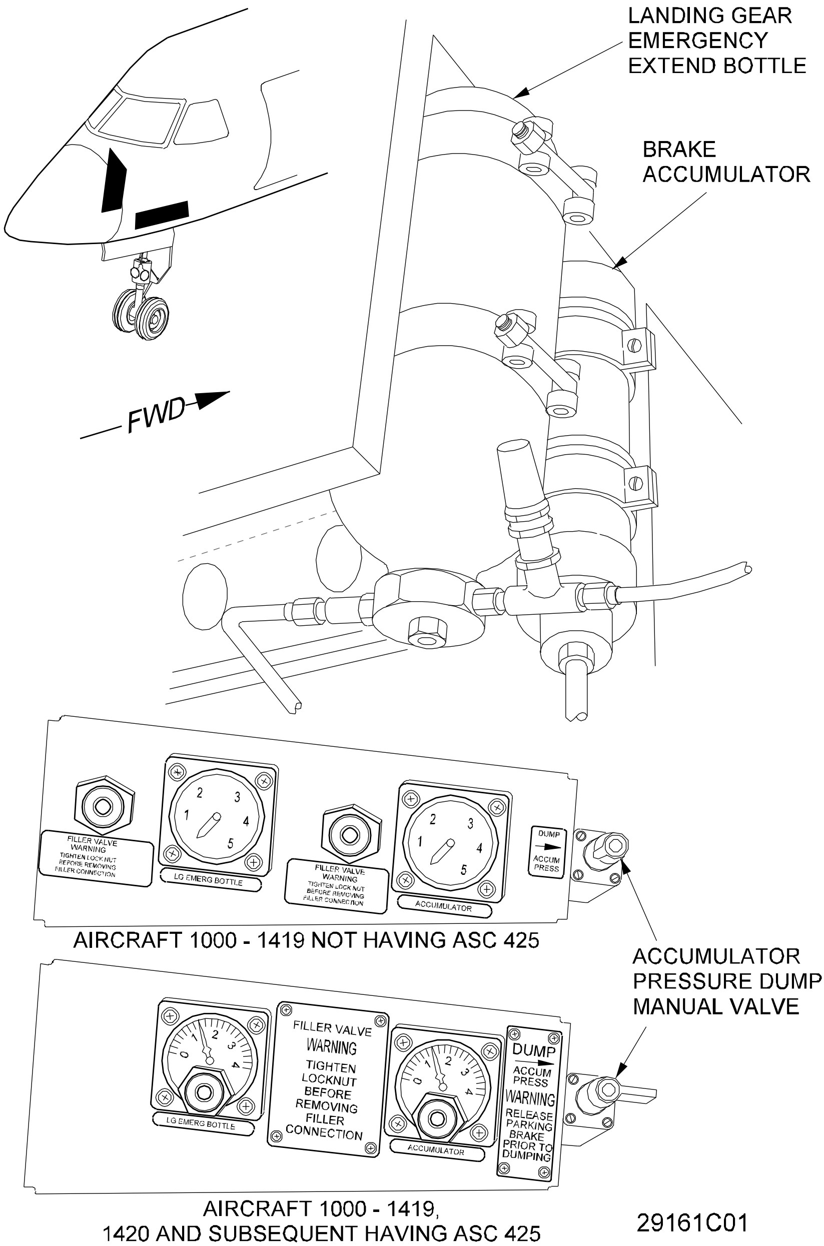
Landing Gear Lever Linkage
Figure 7



Landing Gear Control Panel
Figure 8



Emergency Gear Extension Controls
Figure 9



Emergency Extension Bottle
Figure 10

NUTCRKR SW TEST pushbutton

Momentary action pushbutton, L and R halves of pushbutton will illuminate with amber background when pressed if Left and Right MLG nutcracker switches are in AIR (weight-off-wheels) mode. NOTE: Pressing pushbutton while on ground (weight-on-wheels) will place MLG nutcracker switches in AIR mode. Operation of some aircraft systems will be altered:

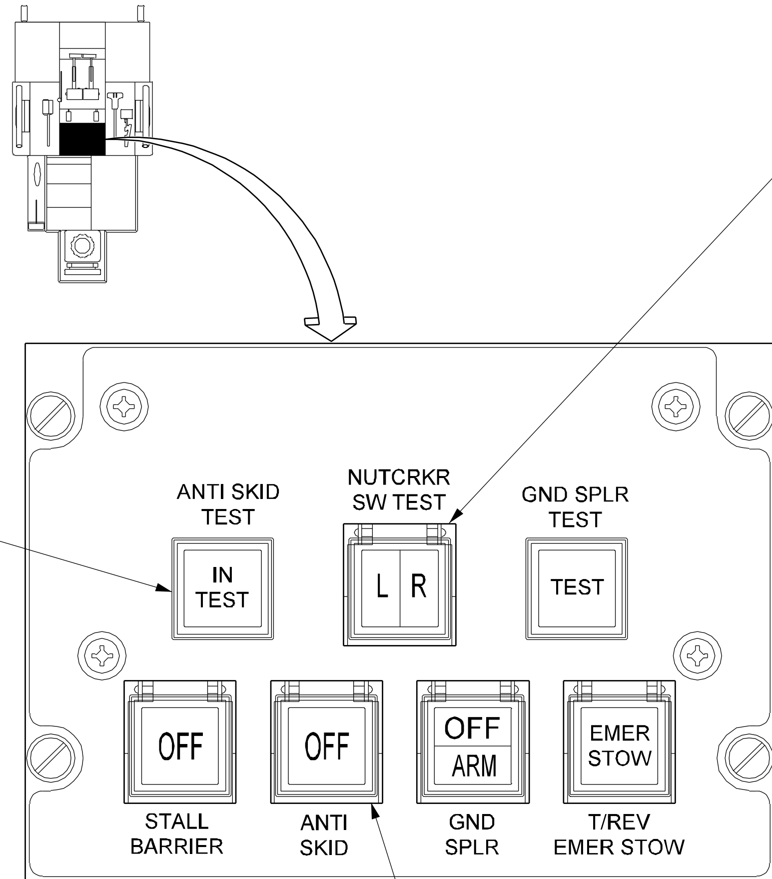
- Nose wheel steering inoperative
- Thrust reversers inoperative
- Ground spoilers inoperative
- Landing gear lever solenoid will retract and landing gear handle may be moved to retract position

ANTI SKID TEST pushbutton

Pushing momentary action ANTI SKID TEST pushbutton initiates BIT test of anti-skid system. BIT test should only be initiated when aircraft speed is less than 12 knots. During the BIT test the following indications are displayed on EICAS:

- Green IN TEST light in pushbutton illuminated
- ANTI SKID FAIL (amber)
- BRAKE FAIL (amber)
- BRAKE MAINT REQ'D (blue)
- SYSTEM TEST light (amber) on front of Skid Control Box is illuminated

If system fault is detected, system will remain in BIT test mode, and all lights / annunciations will remain on. Faults are isolated using FAULT ISOLATION switch on front of Skid Control Box.



ANTI SKID pushbutton

Alternate action pushbutton selects Anti-skid system OFF (amber) or ON

29199C00

Nutcracker Switch Test Pushbutton
Figure 11

END

2A-32-40: Wheels and Brakes System

1. General Description

The GIV aircraft is equipped with one of two types of main landing gear wheels/ brakes and tires, and one of two types of braking systems, depending upon serial number (S/N) and/or Aircraft Service Change (ASC) retrofitted into the aircraft.

Aircraft S/N 1000 through S/N 1213 (with the exception of S/N 1183) are equipped with main landing gear wheels and brakes manufactured by Aircraft Braking Systems (ABS) and tires manufactured by Goodyear, speed rated at 182 knots. Beginning with S/N 1214, aircraft were equipped with main landing gear wheels / brakes manufactured by Dunlop, and Goodyear tires. The later model GIV wheels/brakes and tires are speed rated at 195 knots, and thus support increased speeds associated with higher gross weight takeoffs. Any aircraft not equipped with Dunlop wheels/brakes may optionally have them retrofitted with the installation of ASC 266.

Aircraft S/N 1000 through S/N 1213 (with the exception of S/N 1183) are equipped with an electronic braking system that transmits signals proportional to cockpit brake pedal movement to a Brake Electronic Control Unit that then applies hydraulic pressure to the brakes in response to the variable electric signal. This type of braking system is commonly referred to as Brake-by-Wire. Beginning with S/N 1214, aircraft are equipped with a Hydro-Mechanical Analog Braking system (HMAB). In the HMAB system, the cockpit brake pedals are mechanically connected to valves that apply hydraulic pressure to the brakes in response to brake pedal displacement. Aircraft produced prior to S/N 1214 may have the HMAB system retrofitted with the installation of ASC 307. Both braking systems incorporate full anti-skid protection. Each of the primary braking systems is supplemented with an independent emergency/parking brake system that uses stored accumulator pressure for brake application.

2. Nose Landing Gear Wheels And Tires

The nose landing gear has dual wheels. Each wheel is 22 x 6.6 inches and made of two forged aluminum halves. The two wheel halves are mated together by eight bolts, with the inner joint fitted with an O-ring seal, forming an airtight structure for mounting tubeless tires. The tires are 21 x 7.25 -10 rated at 182 knots for aircraft S/N 1000-1213 (except S/N 1183) without ASC 190 installed. For aircraft 1214 and subsequent, and S/N 1000-1213 with ASC 190, the nose wheel tire speed rating is 195 knots. For more information concerning nose wheel tire and wheel selection, see GIV Illustrated Parts Catalog section 32.

3. Main Landing Gear Wheels and Tires

Each main landing gear has dual wheels. The wheels are forged aluminum with a removable flange to facilitate tire servicing. The flange is mated to the wheel with an O-ring that provides and airtight seal for the tubeless tires. Each wheel is mounted to the landing gear axle with two tapered roller bearings. A brake assembly is integrated into the wheel, fitting into the space between the bearing housing and inside wheel rim.

S/Ns 1000 through 1213 (except 1183) have wheels and brakes manufactured by ABS, and Goodyear 34 x 9.25-16, 18 ply rating, type VII tires, speed rated at 182 knots. Aircraft in this production sequence may be retrofitted with ASC 190 that

increases tire speed rating to 195 knots, and / or with ASC 266 that installs Dunlop wheels, brakes and tires that provide increased braking capability. On aircraft S/N 1214 and subsequent, Dunlop wheels, brakes and tires are production installed, supporting higher aircraft gross weights and consequent higher tire speeds. The Dunlop wheels have a greater radius between the bearing housing and the inside of the wheel hub to accommodate the larger braking package. This requires a lower profile tire, measuring 34 x 9.25-18, in order for the tire and wheel combination to fit into the wheel wells.

ABS wheels have three fusible plugs that melt at 430°F, releasing tire pressure if the wheel overheats. Dunlop wheels have four fusible plugs that melt at 390°F and a wheel safety plug that will deflate the tire if pressure reaches 375 - 650 psi.

4. Brake Assemblies

Brakes manufactured by ABS have four rotating discs (rotors) three stationary discs (stators) an end plate and a pressure plate. See Figure 12 for an illustration of an ABS brake. All of these elements are composed of carbon-metallic alloy, and are referred to as a whole as the disc stack or heat pack. The stators are attached to the torque tube that in turn is bolted to the aircraft gear assembly and holds the wheel bearings. The rotors are attached to the wheel, with notches in the rotors fitting keys on the interior of the wheel hub. The wheel with attached rotors turns on the wheel bearing, with the rotors spinning between the brake stators. Brake bolts attached at the brake housing and fastened to the outside of the end plate maintain sufficient clearance between the rotors and the stators to permit the wheels to turn freely. Five hydraulic actuating pistons are built in the brake housing. When the brakes are applied, hydraulic pressure is applied to the pistons that then move outward from the housing squeezing the pressure plate, and reducing the clearance between the rotors and stators. The surfaces of the rotors and stators are pressed against each other, producing the friction that slows the spinning wheel.

Dunlop brakes have three rotors, two inner stators of double thickness, and a stator of single thickness at the pressure plate and the end plate. The components of a Dunlop brake are shown in Figure 13. Operation of the Dunlop brakes is similar to that of the ABS brakes, but because of the greater energy absorbing mass afforded by the increased diameter of the wheel and resulting larger surface area of the rotors and stators, a higher braking efficiency is attained.

Both brake types have provisions for brake temperature monitoring, anti-skid protection, and application of hydraulic pressure at a reduced level (400 ± 50 psi) to stop wheel spin as the landing gear are retracted into the wheel well.

2A-32-41: Hydro-Mechanical Analog Braking (HMAB) System

1. General Description

The Hydro-Mechanical Analog Braking (HMAB) system is installed on aircraft S/N 1214 and subsequent, and is available for retrofit on aircraft S/N 1000 through 1213 with the installation of ASC 307. The HMAB system applies up to 3000 psi hydraulic pressure from the Combined system to the main landing gear brakes in response to cockpit brake pedal commands. The left and right pilot and copilot brake pedals are mechanically linked to the left and right brake metering valves in the nose wheel well. The metering valves open in response to brake pedal deflection, porting increased pressure with greater brake pedal application. From the brake metering valves, pressurized hydraulic fluid is routed to the augmenter valve in the main wheel well. The augmenter valve increases hydraulic fluid flow

rates and intensifies hydraulic pressure to compensate for any system response loss caused by the distance between the brake metering valves in the nose wheel well and the main landing gear. From the augmenter valve, hydraulic pressure then passes through the anti-skid control valve modules, hydraulic line fuses, pressure transducers and brake shuttle valves and each brake piston assembly.

Hydraulic fuses detect a broken or damaged hydraulic line indicated by increased fluid flow and close to prevent system fluid loss. The anti-skid control valve modules regulate hydraulic pressure to the brake assemblies to maintain wheel rotation during braking. Pressure transducers electronically transmit readings of hydraulic pressure applied to the brakes to the Data Acquisition Units (DAUs). The DAUs formulate this information for display on the cockpit Engine Instrument/ Crew Alerting System (EICAS). See Figure 14 for a schematic of the HMAB system.

The left and right brake metering valves are plumbed to both the Combined and Auxiliary hydraulic systems. If Combined system pressure drops below 1500 psi, internal stops in the metering valves shift, allowing Auxiliary system pressure to operate the brakes with a slight increase in pedal effort required. The Auxiliary hydraulic pump will pressurize automatically (if armed) when the loss of Combined system pressure requires increased brake pedal travel and subsequent engagement of the pedal limit switches. The limit switches in combination with the Combined system low pressure switch complete the circuit to power the Auxiliary pump. When the Auxiliary system is in use, the augmenter valve is bypassed, and pressure is routed through alternate anti-skid modules to the emergency brake shuttles and hydraulic fuses, and then to the brakes.

If Combined or Auxiliary hydraulic pressure are not available, a parking / emergency brake system with stored accumulator pressure may be used to stop the aircraft. The brake accumulator is in the nose wheel well, and has a nitrogen gas pre-charge of 1200 psi. The accumulator is shown in Figure 10. When normal hydraulic pressure is initially applied to the aircraft, the accumulator is fully pressurized with hydraulic fluid at 3000 psi. Accumulator pressure may be read at the accumulator in the nose wheel well, or on the indicator on the copilot lower instrument panel. The accumulator, when fully charged contains sufficient pressurized hydraulic fluid for approximately 5 to 6 brake applications. The PARK / EMER BRAKE handle on the cockpit center console is connect by cable to the modulating valve on the right side of the nose wheel well. Brake modulation can be accomplished by pulling out the handle between ¼ and ½ inch. Handle travel beyond approximately ½ inch results in the application of full accumulator pressure. No anti-skid is available with emergency braking. To apply accumulator pressure for parking the aircraft, the PARK / EMER BRAKE handle is pulled up and then rotated clockwise. To release the PARK / EMER BRAKE, pull the handle up and rotate counterclockwise.

An anti-rotation valve is incorporated in the braking system to stop wheel spin upon landing gear retraction. When the landing gear is down and locked, the anti-rotation valve is closed, but as the gear retracts, pressure from the main landing gear timer valve is routed to the anti-rotation valve where it is reduced to 400±50 psi, and applied to the wheel brakes through the parking / emergency brake lines.

2A-32-42: Hydro-Mechanical Analog Braking (HMAB) Anti-skid Protection

1. General Description

NOTE:

The following system description applies to aircraft S/N 1214 and subsequent, S/N 1000 - 1213 with ASC 307 installed, and S/N 1183.

Hydro-mechanical Analog Braking (HMAB) anti-skid protection includes a primary system and an alternate system. The primary system provides anti-skid protection during normal operations using the Combined hydraulic system. The alternate system provides anti-skid protection during a hydraulic malfunction of the Combined system, and uses the Auxiliary hydraulic system for braking.

Anti-skid protection for the main landing gear brakes operates by regulating hydraulic pressure applied to the brakes to maintain wheel rotation. HMAB anti-skid is effective at taxi speeds above ten (10) kts. (Below 10 knots, the main wheel brakes may be locked by brake pedal commands in order to turn the aircraft within a minimum radius.) Hydraulic pressure is metered by the anti-skid control valve modules in response to commands from the skid control box or Electronic Control Unit (ECU). The ECU, located in the left avionics equipment rack aft of the cockpit, monitors wheel speeds using data from the Wheel Speed Sensors (WSSs). Each main landing gear wheel is equipped with a WSS mounted on the wheel axle that is spun by the rotation of the wheel. The WSSs act as generators, producing an alternating current (AC) signal at a frequency proportional to wheel revolution. The ECU compares the frequency of the AC signals produced by the rotation of the wheels to detect a skidding wheel, indicated when one wheel is not rotating as fast as others. If the ECU detects a rotational speed difference on one of the main landing gear wheels during braking, it signals the anti-skid control valve module to reduce hydraulic pressure to the brake on the slower wheel and to the corresponding wheel on the opposite strut. The ECU controls the hydraulic pressure to wheel brakes in pairs - inboard wheels and outboard wheels. By regulating the same hydraulic pressure to symmetrically paired wheel brakes, aircraft directional control is easier to maintain.

The ECU has three circuit boards for monitoring wheel speeds and controlling brake application. Two boards are used during primary anti-skid system operation: one board for the inboard wheels on each strut, and one board for the outboard wheels on each strut. The third circuit board is used during alternate anti-skid operation and controls the application of Auxiliary system hydraulic pressure to both the inboard and outboard wheel brakes on each strut. Each circuit board is powered separately for redundancy. The board for the outboard wheels is powered by the Emergency Battery 2B Bus through circuit breaker OUTBD ANTI SKID, located at position B-2 on the Copilot Overhead (CPO). The inboard wheel circuit board is powered by the Emergency Battery 1A Bus through circuit breaker INBD ANTI SKID located at position B-1 on the CPO. The circuit board for the alternate anti-skid system is powered by Emergency Battery 2B Bus through the WHEEL SPEED circuit breaker at location C-10 on the CPO.

The ECU circuit boards communicate with three anti-skid control valve modules. There are two modules controlled by the primary anti-skid system circuit boards, one for each main landing gear strut. The third module is controlled by the alternate anti-skid board, and is linked to all four main landing gear wheels. The

alternate module and corresponding circuit board are always electrically powered, even if the primary system components are operating the anti-skid system.

During primary system operation, the circuit board for the outboard wheels compares and controls outboard wheel speed though the WSSs and the left and right strut anti-skid control valve modules. The inboard wheels of each strut are monitored and controlled in the same way. During alternate system operation with Auxiliary hydraulic system pressure, a single circuit board and anti-skid control valve module controls all four wheel brakes. The alternate anti-skid system is not directly linked to the WSSs, it relies upon the inboard and outboard ECU circuit boards for wheel speed monitoring and comparison. Unlike the primary system that controls symmetrically paired wheel brakes, the alternate system controls each strut as a unit. If, for instance, a decrease in wheel speed is detected in the left inboard wheel, the alternate system reduces hydraulic pressure application to both the inboard and outboard wheel brakes of the left strut.

Additional protective features are provided by the anti-skid system. To avoid flat-spotted and/or blown tires, touchdown protection prevents hydraulic pressure from reaching the wheel brakes until the nutcracker (squat) switch is in the ground mode (weight-on-wheels) or the wheels have spun up to twenty-two (22) knots. When the wheels are lowered prior to landing, the nutcracker switch is in the flight (weight-off-wheels) mode and the wheels are not rotating. In this condition, with the anti-skid system on, hydraulic pressure to the brakes is blocked, since the lack of wheel rotation signals a locked wheel. On touchdown, compression of the nutcracker switch to the ground (weight-on-wheels) mode starts a seven second delay during which hydraulic pressure remains blocked to the wheel brakes unless the wheels spin up and attain a speed equivalent of 22 knots. The time delay provides protection for the wheels/brakes in the event of a bounce upon landing that would compress the nutcracker switch and allow application the brakes prior to the subsequent touchdown. Normal hydraulic pressure is available at wheel speeds above 22 knots regardless of nutcracker switch position. Should a nutcracker switch fail in the flight, or weight-off-wheels mode, normal braking with full anti-skid protection is available until the aircraft slows to below 22 knots. Below wheel speeds of 22 knots, the logic of the touchdown protection circuitry reverts to nutcracker switch position. The brakes on the wheels of the strut with the normally operating nutcracker switch will operate with full anti-skid (since the seven second delay has expired), but the brakes on the wheels of the strut with the inoperable nutcracker switch will be without hydraulic pressure, since the nutcracker switch did not transition to the ground (weight-on-wheels) mode to start the timing of the seven second delay. This hazardous condition of full anti-skid braking present on one strut and no braking available on the other strut is avoided by selecting anti-skid off prior to slowing the aircraft below 22 knots.

Similar protection acts as a backup to normal anti-skid protection to avoid locked wheels on the ground during roll out. If a rotational speed difference of approximately thirty-three percent (33%), (as measured between inboard wheels or between outboard wheels) is detected on one wheel, hydraulic pressure is completely removed from the slow wheel brake. This feature is useful in circumstances of severe hydroplaning or icing conditions where wheel rotation is minimal or nonexistent. Removing all hydraulic brake pressure to the locked or non-rotating wheel prevents damage / blow outs when the aircraft exits conditions of very low friction. This feature differs from touchdown wheel protection in that brake pressure is removed from a single wheel, rather than both symmetrical wheels.

In addition to the wheel speeds supplied by the WSSs to the anti-skid system, wheel rotation data is supplied to the automatic spoiler system to operate ground spoilers if there is a failure in the nutcracker switch system. When automatic ground spoilers are armed, they normally deploy when the nutcracker switch on each main landing gear is in the ground, or weight-on-wheels mode. If a nutcracker switch fails, wheel speed can initiate spoiler deployment. If at least one WSS on each strut supplies a signal equivalent to a speed of approximately forty-eight (48) knots, the ECU will supply a signal to the ground spoiler system to deploy. However, as the aircraft slows, the strength of the signal supplied by the WSSs will degrade, and when wheel rotation has slowed to approximately thirty-nine (39) to twenty-three (23) knots, the ground spoilers will retract.

2A-32-43: Electronic Braking System (Brake-by-Wire)

1. General Description

The Electronic or Brake-by-Wire Braking System uses electronic signals in place of mechanical linkages to transmit cockpit brake pedal commands to the main wheel brakes. All wiring is dual channel, providing redundancy for system components. DC power to channel 1 is provided by the forward emergency battery through circuit breaker BCS CHL #1, located on the CPO at position A-2 , and channel 2 is powered from the DC Essential Bus through circuit breaker BCS CHL #2, located on the CPO at position B-2. Anti-skid protection for braking is fully integrated into the system using information from wheel speed sensors (WSSs). The wheel speed sensor circuits are powered by the Left Main DC Bus through the WHEEL SPEED circuit breaker, located on the CPO at position C-10.

Pilot and copilot left and right brake pedals are equipped with linear variable differential transducers (LVDTs). As a brake pedal is depressed, an electrical signal proportionate to pedal displacement is generated. Brake pedal feel is provided by bungees incorporated into the pedal structure. The electrical signals from the LVDTs are sent to the Electronic Control Unit (ECU). If both pilots depress their respective brake pedals, only the signal corresponding to the greatest pressure is sent (the electric signals are not additive).

The ECU is the primary interface between system components. Circuit boards in the ECU control Combined hydraulic system braking and anti-skid functions, Auxiliary hydraulic system braking and pulsed anti-skid braking. One circuit board is dedicated to the Built-in Test Equipment (BITE). The ECU receives signals from the LVDTs, WSSs, and commands from cockpit switch panel selections (Anti-skid ON / OFF, Brake Test, and Brake Nutcracker Override). In response to received data, the ECU is programmed to apply braking in response to LVDT commands, reduce brake application if a wheel skid is detected, or provide a fault detected signal to the DAUs for annunciation on EICAS displays. Other ECU functions include wheel speed data for control of automatic ground spoiler deployment, prevention of a locked wheel on landing, and wheel spindown on landing gear retraction.

The ECU controls the main wheel brakes with inputs to the Hydraulic Brake Control Module (HBCM). The HBCM is connected to pressure lines from the Combined hydraulic system and the Auxiliary hydraulic system. A simplified diagram of the hydraulic components of the Brake-by-Wire system is shown in Figure 15. During normal operations, Combined system pressure displaces a shuttle valve in the HBCM and is routed to right and left brake control valves that are electronically operated by the ECU. The ECU signals the valves to open

proportional to LVDT inputs. Combined system pressure is then directed through brake lines incorporating hydraulic fuses and pressure transducers to the shuttle valves at the wheel brakes. Indication of hydraulic pressure applied to the brakes is available on the cockpit EICAS displays using data transmitted from the pressure transducers to the DAUs. If there is a failure of the Combined hydraulic system, the Auxiliary hydraulic system will provide wheel braking. A pressure switch in the HBCM monitors Combined hydraulic system pressure. If Combined system pressure falls, the HBCM will automatically switch to the Auxiliary system when all of the following conditions are met: (with AUX Pump armed) Combined system pressure drops below 1500 psi, either main landing gear nutcracker switch is in the ground mode, and a brake pedal (LVDT) is depressed to a least 10% of full travel. As Auxiliary system pressure increases, it displaces the HBCM shuttle valve that closes off Combined system pressure input and routes Auxiliary system pressure to the brake control valves. All normal brake system functions are available with Auxiliary hydraulic power.

If Combined or Auxiliary hydraulic pressure is not available, a parking/emergency brake system with stored accumulator pressure may be used to stop the aircraft. The brake accumulator is in the nose wheel well, and has a nitrogen gas pre-charge of 1200 psi. The accumulator is shown in Figure 10. When normal hydraulic pressure is initially applied to the aircraft, the accumulator is fully pressurized with hydraulic fluid at 3000 psi. Accumulator pressure may be read at the accumulator in the nose wheel well, or on the indicator on the copilot lower instrument panel. The accumulator, when fully charged contains sufficient pressurized hydraulic fluid for approximately 5 to 6 brake applications. The PARK / EMER BRAKE handle on the cockpit center console is connect by cable to the modulating valve at the accumulator. Brake modulation can be accomplished by pulling out the handle between ¼ and ½ inch. Handle travel beyond approximately ½ inch results in the application of full accumulator pressure. No anti-skid is available with emergency braking. To apply accumulator pressure for parking the aircraft, the PARK / EMER BRAKE handle is pulled up and then rotated clockwise. To release the PARK/EMER BRAKE, pull the handle up and rotate counterclockwise.

The Brake-by-Wire system provides automatic spindown of the wheels during landing gear retraction. When the landing gear lever is selected to the up position and at least one nutcracker (squat) switch is in the air (weight off wheels) mode, the ECU signals an initial partial brake application to slow wheel spin, followed by a full brake application to stop wheel spin. The initial brake application is for two (2) seconds, the subsequent application is for eight (8) seconds, with the hydraulic pressure applied ranging from 400 psi to 1500 psi.

2A-32-44: Electronic (Brake-By-Wire) Anti-skid Protection

1. General Description

NOTE:

This section pertains to aircraft S/N 1000 - 1213, excluding 1183, without ASC 307.

To prevent wheels from skidding during braking, the Electronic or Brake-by-Wire System provides anti-skid protection. Anti-skid is operable between ten (10) and one hundred seventy (170) knots. (Below 10 knots, the main landing gear wheels may be locked to accomplish tight-radius turns.) Anti-skid is accomplished by

maintaining wheel rotation during braking. Each wheel assembly is equipped with a Wheel Speed Sensor (WSS). The speed sensors are mounted each wheel hub and act as generators driven by wheel rotation. As a wheel turns, the WSS produces alternating current (AC) voltage at a frequency proportional to wheel speed. The variable frequency voltage of each wheel is monitored by the Brake Electronic Control Unit (ECU). Circuit boards in the ECU are programmed to compare the frequencies, and therefore the speeds, of adjacent wheels on each strut (the speed of the outboard wheel on the left strut is compared with the speed of the inboard wheel of the left strut). If the ECU detects a difference in rotational speed between the two wheels, the circuit boards in the ECU determine that the slow wheel is in a skid, and the ECU signals the brake control valve on the Hydraulic Brake Control Module (HBCM) to reduce hydraulic pressure to both wheel brakes on the strut with the skidding wheel. When the rotational speed of the slow wheel recovers to match that of the faster wheel, hydraulic pressure is restored to both wheel brakes. Anti-skid protection is also available using Auxiliary system pressure if the Combined system fails.

If the anti-skid system fails or is disabled, a type of anti-skid braking with degraded performance is available. This type of braking is termed open-loop pulsed anti-skid. This mode does not compare wheel rotational speeds. Brake pressure is proportional to brake pedal LVDT displacement, but is applied only three (3) to five (5) times per second, resulting in pressure fluctuations from zero to commanded pressure levels. Because full brake pressure is not constantly applied, wheel skids are less likely. The abrupt fluctuations between zero and full commanded brake pressure can be felt in the brake pedals, thus this mode of braking is popularly called “bang-bang” braking.

The anti-skid Wheel Speed Sensors in combination with the nutcracker (squat) switches are also used to provide data for other braking functions and protective features. With the anti-skid system selected on, the nutcracker switches block brake application to the wheels when the aircraft is in flight (weight-off-wheels). When the landing gear is lowered in preparation for landing, brake pressure should not be available until the aircraft is on the ground and both the nutcracker switches compressed (weight-on-wheels). An additional locked wheel circuit in the ECU protects against a failure of the nutcracker switch system that would allow application of the wheel brakes prior to landing, causing flat-spotted or blown tires and/or wheel damage. The circuit incorporates the following logic. When the landing gear is initially lowered, the wheels do not rotate. The WSSs signal the lack of rotation to the ECU that equates this condition to locked wheels and blocks hydraulic pressure to the brakes. The locked wheel signal is preserved in the ECU until the aircraft wheels spin up on runway contact and reach a speed equivalent of thirty-five (35) knots as sensed by the WSSs, or until one of the nutcracker switches compresses and enters the ground (weight-on-wheels) mode. A time delay in the circuit will also maintain the locked wheel signal for seven seconds after the nutcracker system transitions from flight to ground mode in order to provide protection in the event of an aircraft bounce on landing. If a nutcracker system malfunction prevents a switch from achieving the ground (weight-on-wheels) mode, the locked wheel circuit preserves brake system operation with anti-skid protection until the aircraft slows to thirty-five (35) knots or below, at which time anti-skid protection is lost, but full pulsed braking is available if the anti-skid selected off.

In conditions of severe hydroplanning, a separate locked wheel circuit is effective at aircraft wheel speeds above twenty-six knots. This circuit releases hydraulic

pressure to the brake on a wheel that is rotating at a speed thirty percent (30%) less than the speed of the other paired wheel. This protection decreases the risk of loss of directional control when the aircraft exits the hydroplanning condition, and would otherwise encounter a sudden full brake application.

Speed signals from the WSSs are used to provide a ground spoiler activation signal in the event of a malfunction in the nutcracker switches. Automatic ground spoiler activation is provided on landing when the system is armed, flaps are extended to a least 22° and both nutcracker switches are in the ground (weight-on-wheels) mode. If a nutcracker switch fails to transition to the ground mode on landing , then speed signals from the WSSs can prompt activation. When at least one wheel on each strut spins up to approximately fifty-six (56) knots, the ground spoilers are activated. The spoiler activation signal provided by the WSSs is preserved until the wheels slow to approximately thirty-nine (39) to twenty-two (22) knots, at which time the electrical signal generated by the WSSs is too weak to maintain automatic ground spoiler deployment and the spoilers will stow.

For aircraft S/N 1000 - 1143 with ASC 166/166A incorporated, and S/N 1144 and subsequent, a circuit similar to the ground spoiler activation circuit is installed for activation of the thrust reversers. If WSSs on at least one wheel of each strut indicate a rotation speed of fifty-seven (57) knots, thrust reversers may be deployed even if the nutcracker system has not transferred to the ground (weight-on-wheels) mode.

2A-32-45: Brake Temperature Monitoring System (BTMS)

1. General Description

The temperature of each main landing gear brake is monitored by a probe incorporated into one of the brake assembly bolts on each wheel. The probes are resistance type devices (RTDs) that vary resistance to electrical current with temperature at a predetermined rate. Brake temperatures sensed by the probes are transmitted for view on cockpit displays. Electrical power for the temperature probes and monitoring circuits is provided by 28V DC through the BTMS circuit breaker at position D-10 on the CPO.

Temperature readings and overheat warning displays are presented in two different formats. For aircraft with SPZ 8000 DAFCS, a Brake Temperature Monitoring System (BTMS) panel, is installed in the cockpit center console beginning with aircraft S/N 1156, and available for installation in prior aircraft S/Ns as ASC 167. For aircraft with SPZ 8400 DAFCS, temperature readings and overheat warnings are displayed on the EICAS.

2. Brake Temperature Monitoring System (SPZ-8000 AIRCRAFT)

A. Description

A Brake Temperature Monitor System (BTMS) panel is installed in aircraft S/Ns 1156 through 1252 (except S/N 1236) and those aircraft S/Ns 1000-1155 retrofitted with ASC 137. The BTMS panel, shown in Figure 16, is located on the cockpit center console (location optional) and contains a temperature gage with readout selector, an overheat light, and a test switch. The temperature readout selector is normally positioned to ALL. In this position, the highest temperature of the four main landing gear brakes is displayed on the temperature gage. The selector is rotated to monitor the temperature of each individual brake. If temperature limits for the type of brake installed are exceeded, the OVHT light on the BTMS panel

illuminates. If the OVHT light on the BTMS panel is illuminated, rotating the temperature display selector through the brake positions will determine which brake(s) is overheating. On aircraft S/Ns 1156-1167, a temperature limit exceedance also prompts the illumination of the master caution lights, and an illumination of the BRAKE TEMP light installed above the pilot primary flight display. An amber BRAKE OVHT message is also annunciated on the EICAS for aircraft S/Ns 1168 -1252.

NOTE:

Temperature limits for ABS brakes are 400° ±10°C - the OVHT light will remain on until the brake(s) has cooled below 370°C. For Dunlop brakes the temperature limit is 625° ±25° C - the OVHT light will remain on until the brake(s) has cooled to below 595°C.

The TEST push-button on the BTMS verifies that system circuits are operating properly. With the temperature selector switch in the ALL position, the test function checks all four wheel circuits and the over temperature warning circuit. A proper test is indicated by illumination of the OVHT light and a temperature gage reading as follows:

- 425°C to 475°C: Airplanes having ASC 167 (ABS brakes)
- 875°C to 975°C: Airplanes having ASC 346 (Dunlop brakes)

If either of these conditions is not valid, selecting each wheel brake with the readout selector and pressing the TEST push-button will identify the faulty circuit. A temperature reading of 0°C indicates a short in the selected sensor circuit.

3. Brake Temperature Monitoring System (SPZ-8400 EQUIPPED AIRCRAFT)

A. Description

The Brake Temperature Monitor System (BTMS) on aircraft with SPZ 8400 DAFCS installed is mechanically and electrically the same as that installed on prior S/N aircraft. Only the display format differs. Temperature data from the brake temperature probes is supplied to the Data Acquisition Units (DAUs) that communicate the data for presentation on the CAS display. The Brake page is selected to the CAS with Line Select Key (LSK) entries on the Display Controller.

On the CAS, brake temperatures are formatted as bar graphs rising vertically with temperature. A center bar displays temperature scale increments in C° X 100, with the temperature of the inner and outer brakes on the left and right struts aligned on either side of the scale. Above the bar graph display is a presentation of applied brake pressure for left and right brakes in increments of psi X 100. Whenever the landing gear is extended, a readout of the peak temperature recorded during the last brake application in °C and identification of the brake with the peak temperature is displayed beneath the bar graphs. The existing record of prior brake application peak temperature is erased whenever the landing gear is lowered for landing, and when the aircraft accelerates faster than 60 knots on take off roll, initiating a new cycle for temperature recording.

If a brake temperature exceeds 625°C ±25°C, the bar graph display for that brake will be displayed in amber, and a BRAKE OVHT caution message is

displayed on the CAS. The maximum temperature and time (in GMT) of the exceedance is recorded and is available for display by selecting the exceedance page on the CAS. The bar graph display will return to the normal white color display and the BRAKE OVHT caution message will clear when the brake cools to 595°C or below.

If an individual brake sensor or monitor circuit malfunctions, the corresponding bar graph is replaced with a red X. If a DAU fails, both brake temperature graphs on the strut monitored by the DAU are replaced by X's and the peak temperature readout is replaced with amber dashes.

For additional information on the Brake System Temperature displays, refer to the Honeywell SPZ-8000 or SPZ-8400 Digital Automatic Flight Control System Pilot Manual for the Gulfstream IV.

4. Controls and Indications:

A. Circuit Breakers (CBs)

The wheels and brakes are protected by the following circuit breakers:

Circuit Breaker Name	CB Panel	Location	Power Source
WHEEL SPEED	CPO	C-10	Left Main 28V DC Bus **(1)** Emergency 28V DC Bus 2B **(2)**
BTMS **(3)**	CPO	D-10	Right Main 28V DC Bus
APPLIED BRAKE PRESSURE	CPO	D-6	Essential 28V DC Bus
BCS CHL #1 **(1)**	CPO	A-2	Fwd Emergency Battery 28V DC Bus
BCS CHL #2 **(1)**	CPO	B-2	Essential 28V DC Bus
INBD ANTI SKID **(2)**	CPO	A-2	Emergency 28V DC Bus 1A
OUTBD ANTI SKID **(2)**	CPO	B-2	Emergency 28V DC Bus 2B
WHEEL BRK ACCUM PRESS	CPO	D-3	Essential 28V DC Bus

NOTE(S):

(1) Electronic Brake-by-Wire System

(2) Hydro-Mechanical Analog Braking System

(3) Aircraft with SPZ-8000

B. Caution (Amber) CAS Messages:

Caution CAS messages associated with wheels and brakes are:

CAS Message	Cause or Meaning
ANTI-SKID FAIL (Brake-by-Wire	Normal anti-skid control circuit or other failure: system has switched to open loop pulsed anti-skid control (bang-bang braking) **OR:** ANTI-SKID switch selected to OFF
ANTI-SKID FAIL (Hydro-Mechanical Analog Braking)	Miscompare in 28V DC power supplies or in nutcracker switches **OR:** ANTI-SKID switch selected to OFF or system circuit problems **OR:** Auxiliary hydraulic system fails to energize or pressure fails to rise within three (3) seconds after brake pedals are depressed

CAS Message	Cause or Meaning
BRAKE FAIL (Brake-by-Wire)	Failure of brake system. BCS cannot provide braking
BRAKE FAIL (Hydro-Mechanical Analog Braking)	Loss of Combined and Auxiliary hydraulic system pressure. No braking is available
BRAKE OVHT	One or more brake assemblies has exceeded temperature limit: • 400°C for S/N 1156-1213 and aircraft with ASC 167 • 625°C for S/N 1214 and subsequent, and S/N 1000-1213 with ASC 346
BRAKE PEDAL (Brake-by-Wire)	Malfunctioning brake pedal

C. Advisory (Blue) CAS Messages

Advisory CAS messages associated with the brake system are:

CAS Message	Cause or Meaning
BRAKE MAINT REQ'D (Brake-by-Wire)	(For S/N 1000-1213 with ASC 190, 266, or 296) If message is displayed after landing gear retraction and extinguishes approximately five (5) seconds later, the auto spin-down feature has failed
BRAKE MAINT REQ'D (Hydro-Mechanical Analog Braking)	(For S/N 1214 and subsequent) If message is displayed during takeoff, landing ground roll, or during flight with the landing gear extended, a wheel speed sensor (WSS) miscompare lasting more than five (5) seconds has been detected

5. Built-in Test (BIT) - HMAB System

The Anti-skid Test pushbutton, shown in Figure 11, is used to initiate a system BIT test from the cockpit. Pressing the pushbutton performs the same function as pressing the System Test button on the face of the Skid Control Box (ECU) in the avionics equipment rack, illustrated in Figure 17. For a valid test, the following indications should appear.

- The Anti-skid Test switch illuminates IN TEST
- Amber ANTISKID FAIL and BRAKE FAIL messages appear on EICAS
- Blue BRAKE MAINT REQ'D message appears on EICAS
- System Test light illuminates on the Skid Control Box (ECU)

6. Built-in Test Equipment (BITE) – Brake-by Wire System

Pressing the BRAKE TEST pushbutton, shown in Figure 18 will initiate a brake system test. During the test, the following indications are displayed.

- Amber ANTISKID FAIL, BRAKE FAIL, and BRAKE PEDAL FAIL messages appear on EICAS
- Blue BRAKE MAINT REQ'D message appears on the EICAS
- The pushbutton illuminates IN TEST
- The BITE indicator illuminates on the ECU

If one (or more) of the indications fails to appear during the BITE test, a fault is

indicated. Faults may be isolated by rotating the selector on the front of the ECU. See the illustration in Figure 19

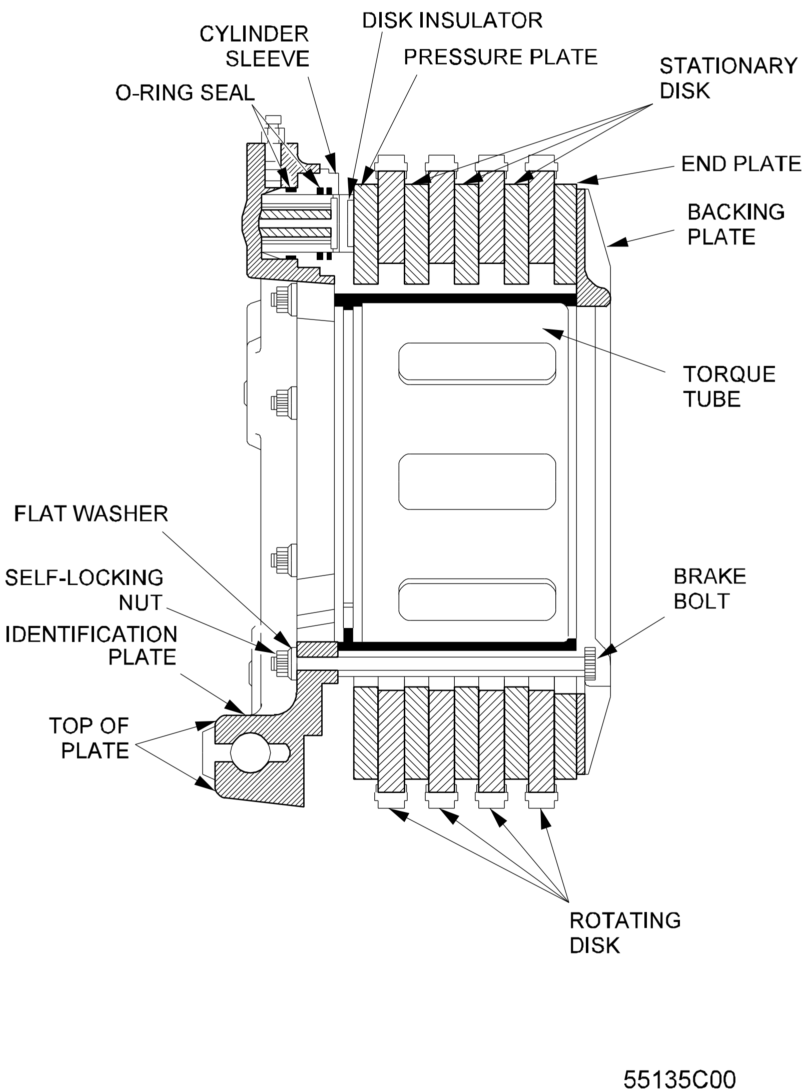
NOTE:

BIT or BITE tests should be performed at taxi speeds less than 10 knots. For the duration of the BIT or BITE tests, normal anti-skid operation is interrupted.

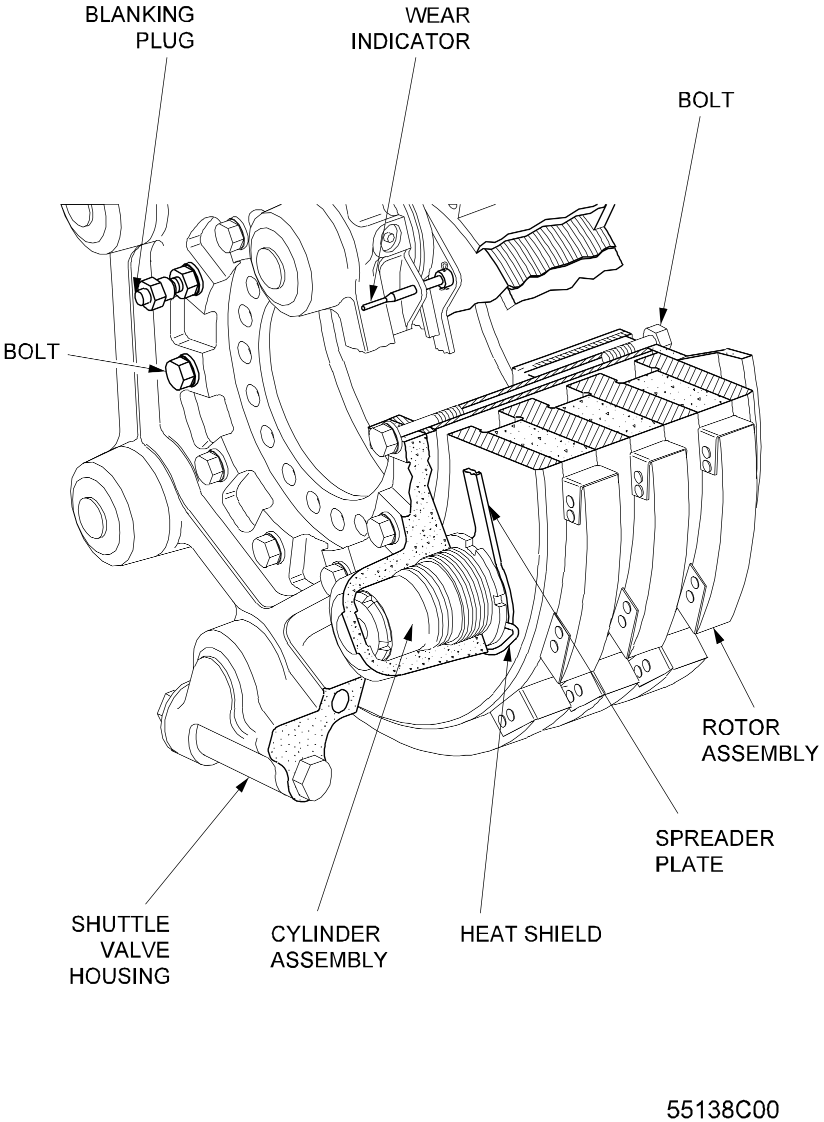
7. **Limitations:**

Takeoff is prohibited with BRAKE FAIL or BRAKE PEDAL message displayed.

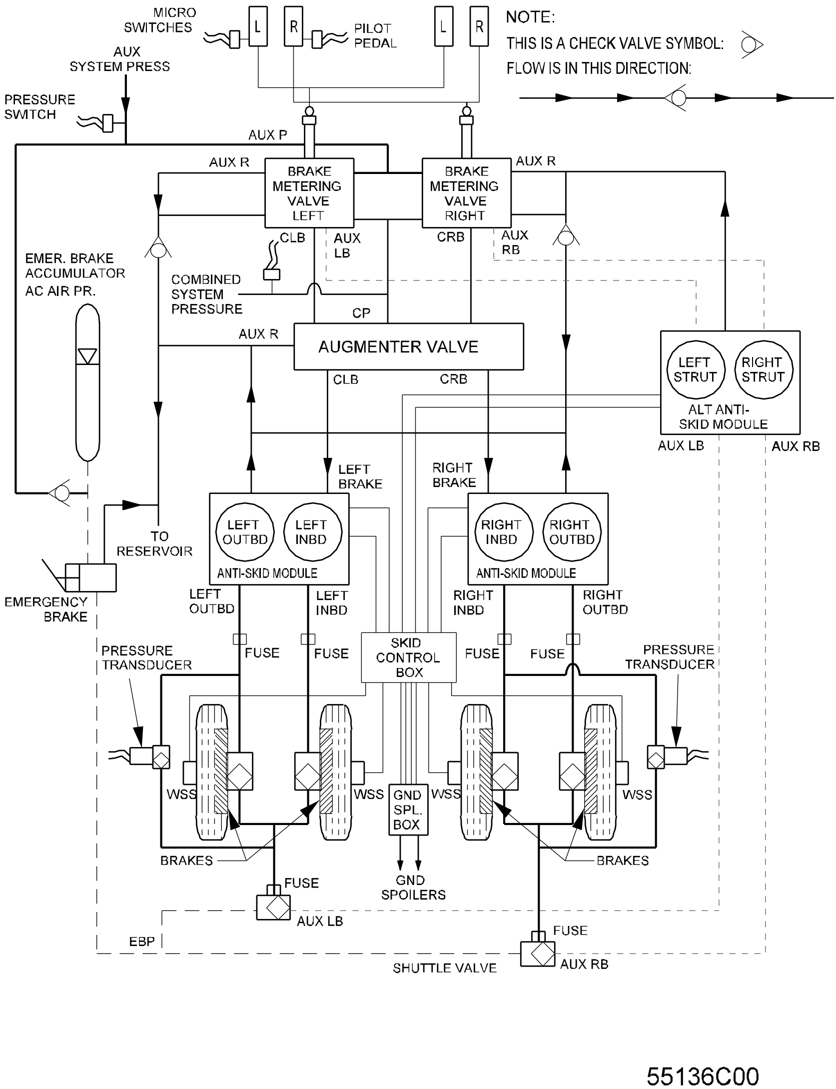
Takeoff is permitted with Anti-Skid system inoperative, provided Ground Spoilers are operative, 20° flaps are used, and the Cowl and Wing Anti-Icing systems are not used.



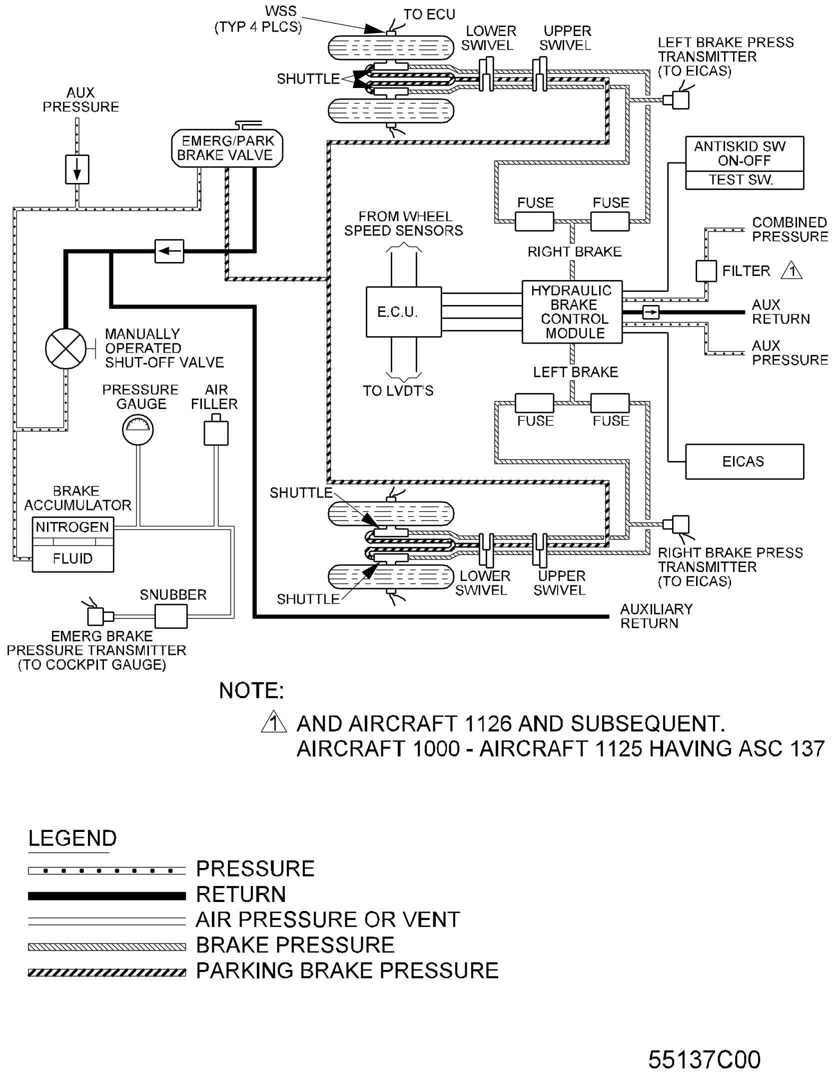
ABS Brake Components
Figure 12



Dunlop Brake Components
Figure 13



HMAB Hydraulic Schematic
Figure 14



Brake-by-Wire Hydraulic Schematic
Figure 15

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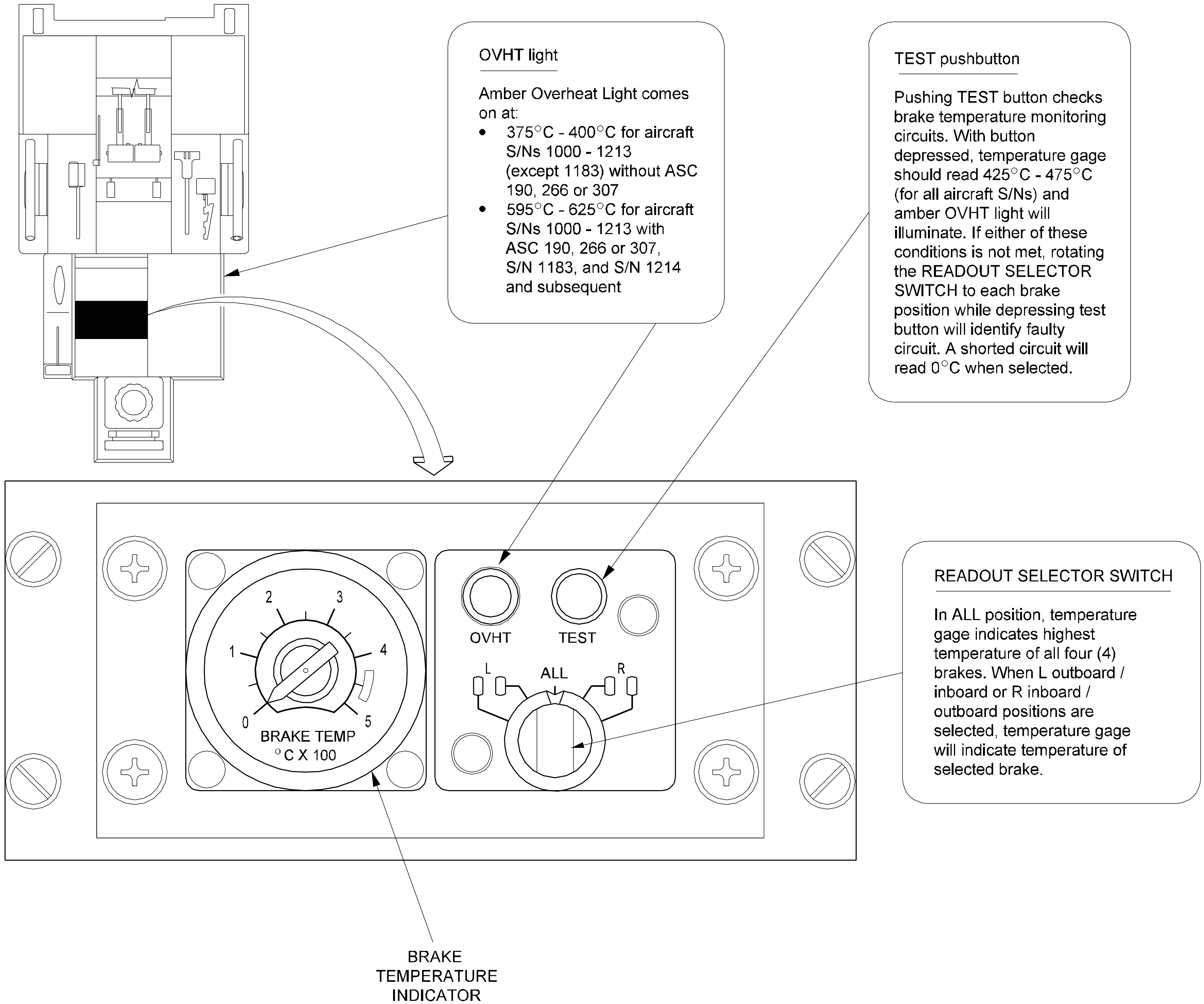
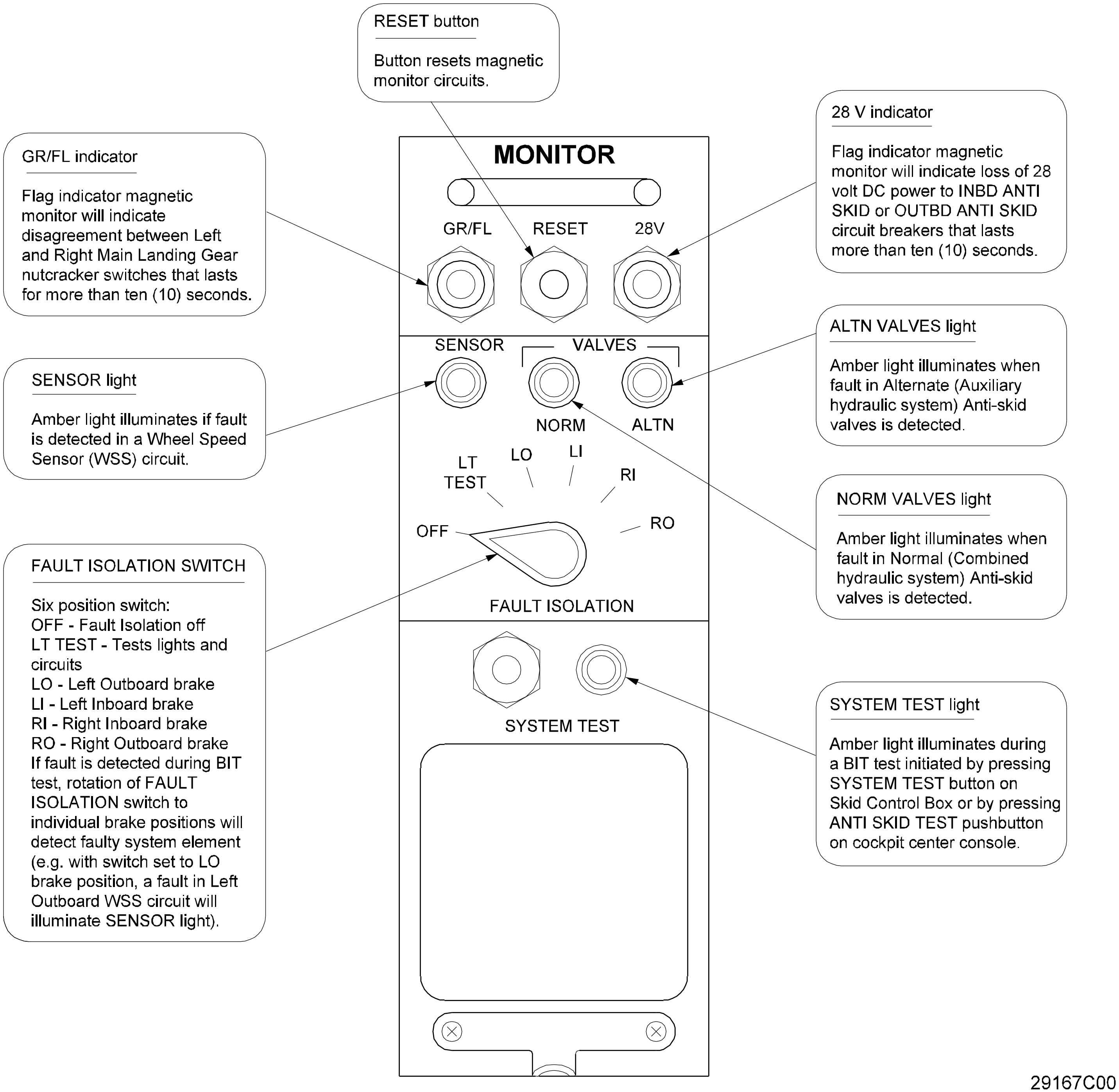


Brake Temperature Monitoring System Panel (SPZ-8000 Aircraft)
Figure 16



Skid Control Box (ECU)
for HMAB Anti-skid
Figure 17

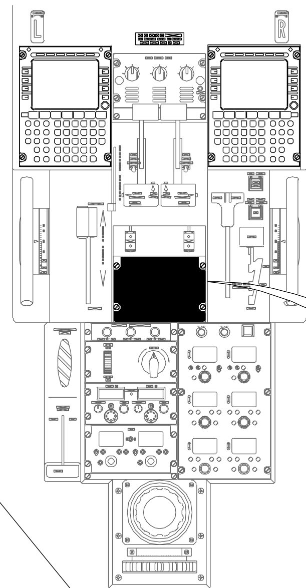
BRAKE NUTCRACKER ORIDE pushbutton

Momentary action amber ON switch, when pressed in air (weight-off-wheels) overrides air signal of MLG nutcracker switches, sending ground signal (weight-on-wheels) to brake system ECU, allowing brake pressure application with brake pedals, monitored on EICAS.

NUTCRKR SW TEST pushbutton

Momentary action pushbutton, L and R halves of pushbutton will illuminate with amber background for when pressed if Left and Right MLG nutcracker switches are in AIR (weight-off-wheels) mode. NOTE: Pressing pushbutton while on ground (weight-on-wheels) will place MLG nutcracker switches in AIR mode. Operation of some aircraft systems will be altered:

- Nose wheel steering inoperative
- Thrust reversers inoperative
- Ground spoilers inoperative
- Landing gear lever solenoid will retract and landing gear handle may be moved to retract position

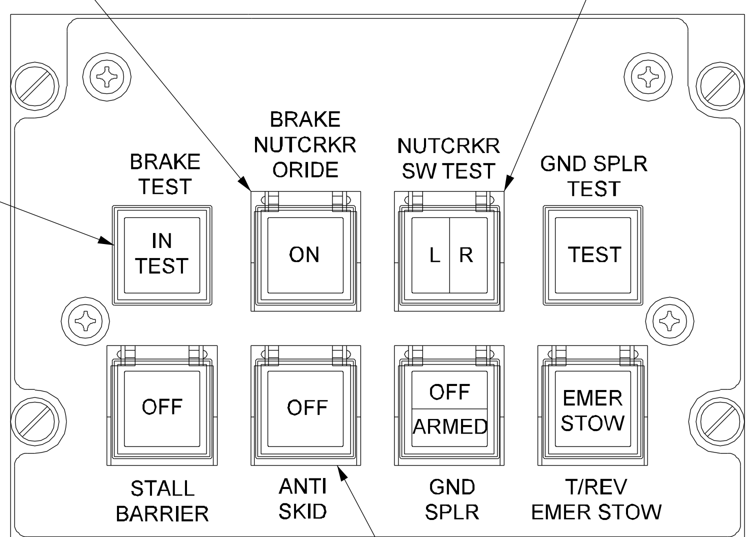


BRAKE TEST pushbutton

NOTE: Test should only be initiated if aircraft speed is less than 12 knots.
Momentary pushbutton initiates a BITE test in ECU (BITE test may also be initiated with toggle switch on front of ECU box). When pushbutton is depressed, IN TEST green light in pushbutton will illuminate for approximately three (3) seconds during which BRAKE test takes place. The following annunciations will be displayed on CAS:

- BRAKE MAINT REQ'D (blue)
- ANTISKID FAIL (amber)
- BRAKE FAIL (amber)
- BRAKE PEDAL FAIL (amber)

System fault is indicated if BRAKE TEST light and / or CAS annunciations are not present.



ANTI SKID pushbutton

Alternate action pushbutton selects Anti-skid system OFF (amber) or ON

29168C00

Brake Test Pushbutton for
Brake-by-Wire System
Figure 18

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FAULT light

ON (amber) when fault detected in anti-skid system during BITE test

LEFT light

ON (amber) when fault detected (FAULT light ON) in left hand component of anti-skid system and FAULT ISOLATION SELECTOR SWITCH rotated to faulty component type

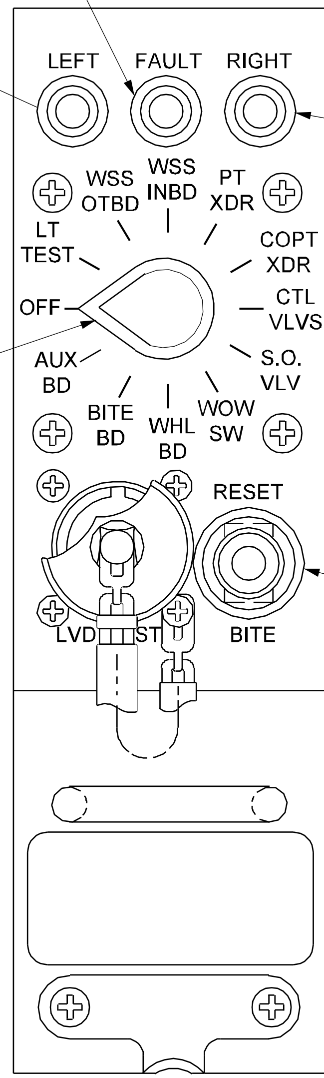
RIGHT light

ON (amber) when fault detected (FAULT light ON) in right hand component of anti-skid system and FAULT ISOLATION SELECTOR SWITCH rotated to faulty component type

FAULT ISOLATION SELECTOR SWITCH

Selector rotates through twelve (12) positions to diagnose source of FAULT light:

- OFF - selector off
- LT TEST - tests lights on ECU
- WSS OTBD - selects outboard wheels speed sensors
- WSS INBD - selects inboard wheels speed sensors
- PT XDR - selects pilot brake pedal LVDTs
- COPT XDR - selects copilot brake pedal LVDTs
- CTL VLVS - selects HBCM brake control valve
- S.O. VLV - selects HBCM shutoff valve
- WOW SW - selects main landing gear nutcracker switches
- WHL BD - selects Wheel circuit board
- BITE BD - selects BITE circuit board
- AUX BD - selects Auxiliary circuit board



BITE / RESET toggle switch

Initiates a BITE test of anti-skid system / resets BITE system and extinguishes amber lights (LEFT / FAULT / RIGHT) if illuminated during BITE test

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Brake-by-Wire BITE / Fault Indications
Figure 19

END

2A-32-50: Nose Wheel Steering System

1. General Description:

The aircraft has a steerable nose wheel operated by an electronic and hydraulic steer-by-wire system using inputs from a control wheel mounted on the pilot side console or inputs from rudder pedal position. Hydraulic pressure from the Combined powers nose wheel steering. The nose wheel may be turned 78° (± 2°) left or right of center with the cockpit control wheel, or 7° (±1°) using only rudder pedal input. Control wheel and rudder pedal inputs are additive, but total nose wheel displacement from center is limited to 80° (±2°) by mechanical stops. Nose wheel steering must be selected ON with the guarded system switch next to the control wheel. The system switch and other system components and controls are shown in Figure 20. With the nose wheel steering activated, moving the control wheel turns a dual potentiometer located beneath the control wheel inside the side console. Rudder pedal movement turns a similar potentiometer located under the copilot floorboard. Electrical signals proportional to control wheel / rudder displacement are routed from the potentiometers an Electronic Control Module (ECM) installed in the left hand avionics rack. The ECM sums the control wheel and rudder pedal inputs to obtain the desired steering command. The steering command is compared to the present position of the nose wheel, and an electric signal is transmitted to the Electro-Hydraulic Servo Valve (EHSV) in the nose wheel well, porting hydraulic pressure to the steering actuator that turns the nose wheel in the desired direction by moving the torque linkage connecting the actuator and the nose wheel. The steering control wheel is returned to the neutral, or straight ahead position by centering springs if it is released. The rudder pedals are centered by the bungees that provide artificial feel for rudder inputs.

To ensure that the nose wheel remains centered prior to nose wheel retraction into the nose wheel well, internal centering cams are incorporated in the shock strut, and the steering system is restricted to ground operation only by several safeguards. The landing gear lever handle must be in the down position to complete the electrical circuits for steering. Hydraulic pressure for steering is provided through the nose landing gear extend lines. The hydraulic pressure must pass through two (2) shut-off valves (SOVs) before reaching the EHSV and steering actuator. See the schematic in Figure 21. The first, SOV, #1, is powered by 28V DC through circuit breaker STEER BY WIRE #1 (position C-12) on the COP. For the SOV to open, the guarded PWR STEER switch next to the control wheel must be on, and the nose gear down lock switch must be engaged. SOV #2 is powered by 28V DC through circuit breaker STEER BY WIRE #2 (position D-12) on the COP, and requires that the nose gear nutcracker (squat) switch be compressed in the ground (weight-on-wheels) position for the valve to admit hydraulic pressure for the steering actuator. When the nose gear nutcracker switch is compressed, the ECM initiates a one second delay before responding to steering commands from the rudder pedals or the control wheel, maintaining the nose wheel in the centered or straight ahead direction. The delay avoids abrupt steering commands caused by large rudder pedal inputs during crosswind landings.

The nose wheel steering linkage may be disconnected for aircraft towing by removing the connecting pin on the torque link between the actuator and the nose

wheel. A tow bar is then connected to the nose wheel axle.

Several Aircraft Service Changes (ASCs) have modified the nose wheel steering system from the original configuration.

- ASC 176: modifies aircraft S/Ns 1000-1242 (modification incorporated into aircraft S/N 1243 and subsequent) to include additional circuits that monitor operation of the system and prompt a STEER BY WIRE FAIL message on the CAS if any of the following occur: nose wheel nutcracker switch stuck in the ground (weight-on-wheels) position, causing the No. 2 SOV to remain in the open position, loss of electrical power to the system, an improperly installed ECU or ECU connectors, or the No. 2 SOV fails to open after touchdown and nose wheel nutcracker switch compression.
- ASC 260A: improves the performance of the nose wheel steering in aircraft S/Ns 1000-1242. This is accomplished by increasing the hydraulic line size to the nose wheel steering, thus increasing hydraulic fluid flow. In addition, modifications are made to the handwheel, ECU and hydraulic components within the system. These improvements are production installed in aircraft S/N 1243 and subsequent.
- ASC 302A: available for installation on all aircraft, installs a separate switch that shuts off rudder pedal nose wheel steering and restricts steering to the hand control wheel only. The two (2) position switch is labelled NORMAL and HANDWHEEL ONLY. When HANDWHEEL ONLY is selected, a blue light adjacent to the switch is illuminated. For aircraft S/Ns 1000-1252 without SPZ 8400, a light installation, mounted above the pilot Primary Flight Display (PFD) will illuminate with the legend RUD STRG OFF when nose wheel steering is selected to HANDWHEEL ONLY. For aircraft S/Ns 1000-1252 with SPZ 8400 installed, and aircraft S/N 1253 and subsequent an annunciation of RUDDER STRG OFF will appear on the CAS when the steering switch is selected to HANDWEEL ONLY.
- ASC 302A AM2: installs an additional light above the copilot PFD, identical to the one installed on the pilot side for aircraft S/Ns 1000-1252 without SPZ 8400. The additional light legend is also RUD STRG OFF, and the light will illuminate in conjunction with the light on the pilot side when nose wheel steering is selected to HANDWHEEL ONLY.
- ASC 302B: clarifies installation of the ASC 302 series by combining all previous releases into a single document. The ASC 302 series consists of:
    - ASC 302: initial installation of NWS select option
    - ASC 302 AM1: corrects man-hours for ASC 302
    - ASC 302 AM2: rudder potentiometer adjustments
    - ASC 302A: incorporated ASC 302, AM1 and AM 2; installed new select panel; added indicator light (SPZ-8000) or CAS message (SPZ-8400)
    - ASC 302A AM1: corrected wiring installed by ASC 302A; adjusted man-hours for ASC 302A
    - ASC 302A AM2: added copilot indicator light (SPZ-8000); modified wiring to eliminate nuisance messages and indications; revised test procedures
- ASC 323: installs a geared hand control wheel for nose wheel steering in aircraft S/Ns 1000-1230. The geared control wheel is production installed

in aircraft S/N 1231 and subsequent. The two to one ratio wheel provides a less sensitive input for nose wheel steering, allowing smoother turns, by increasing the control wheel rotation required for steering commands.

- ASC 434: modifies the nose wheel steering disconnect panel by removing the existing NOSE WHL STR SEL CONTROL panel (immediately aft of the control wheel), and adding a Nose Wheel Steering PEDAL DISC switch panel (forward of the existing PWR STEER panel). This modification will allow the pilot to select normal steering (handwheel and rudder pedals) or handwheel only. In addition to the RUD STRG OFF light(s) on the instrument panel, a RUDDER STRG OFF message is displayed on the EICAS (SPZ-8400 equipped airplanes). This ASC is applicable to aircraft S/Ns 1000-1444, and is production installed in aircraft S/N 1445 and subsequent.

2. Controls and Indications:

A. Circuit Breakers (CBs)

Nosewheel steering is protected by the following circuit breakers:

Circuit Breaker Name	CB Panel	Location	Power Source
STEER BY WIRE #1	CPO	C-12	Essential 28V DC Bus
STEER BY WIRE #2	CPO	D-12	Right Main 28V DC Bus

B. Caution (Amber) CAS Messages

Caution CAS messages associated with nosewheel steering are:

CAS Message	Cause or Meaning
STEER BY WIRE FAIL	Both steering channels have failed: steering not available. System is in shimmy damping mode. With ASC 176 incorporated, a switch position / system status miscompare is detected

C. Advisory (Blue) CAS Messages

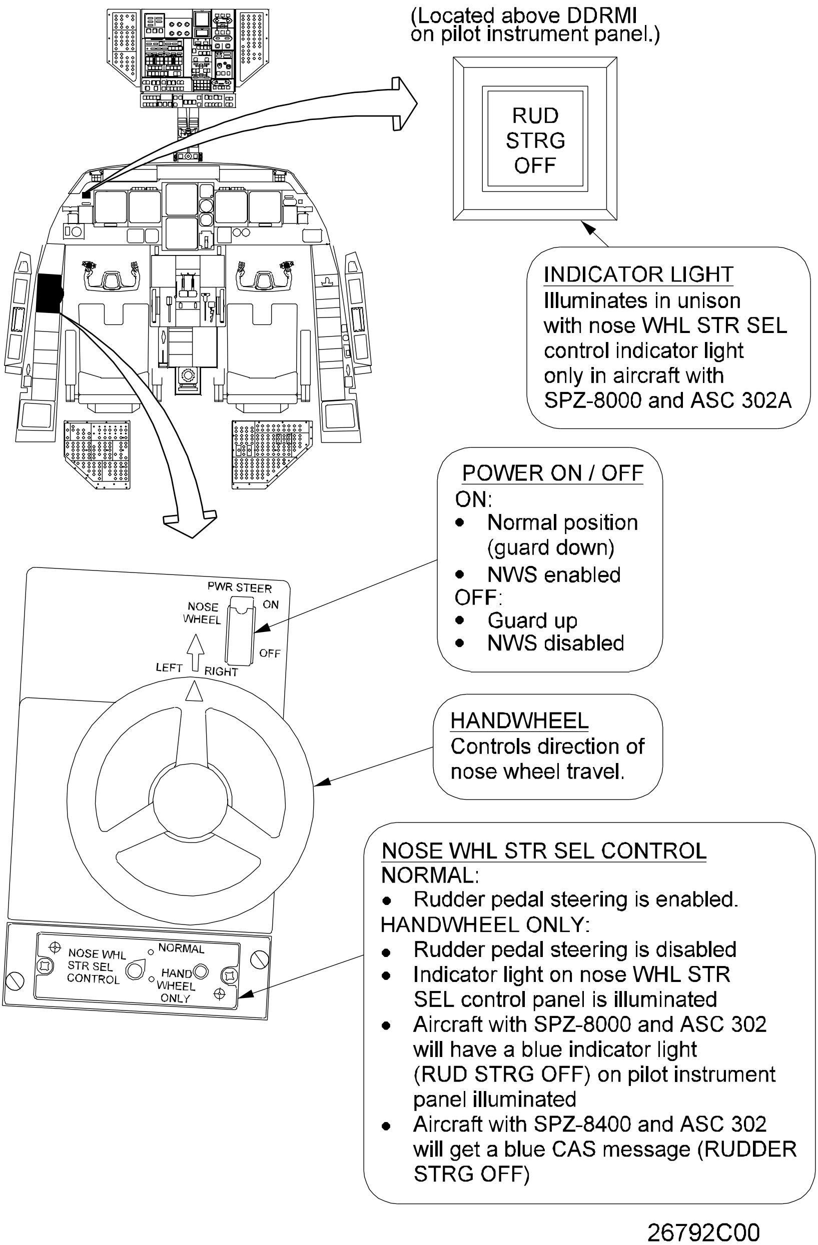
Advisory CAS messages associated with the nosewheel steering system are:

CAS Message	Cause or Meaning
RUDDER STRG OFF	(With ASC 302A) HAND WHEEL ONLY mode is selected on NOSE WHEEL STEERING control panel

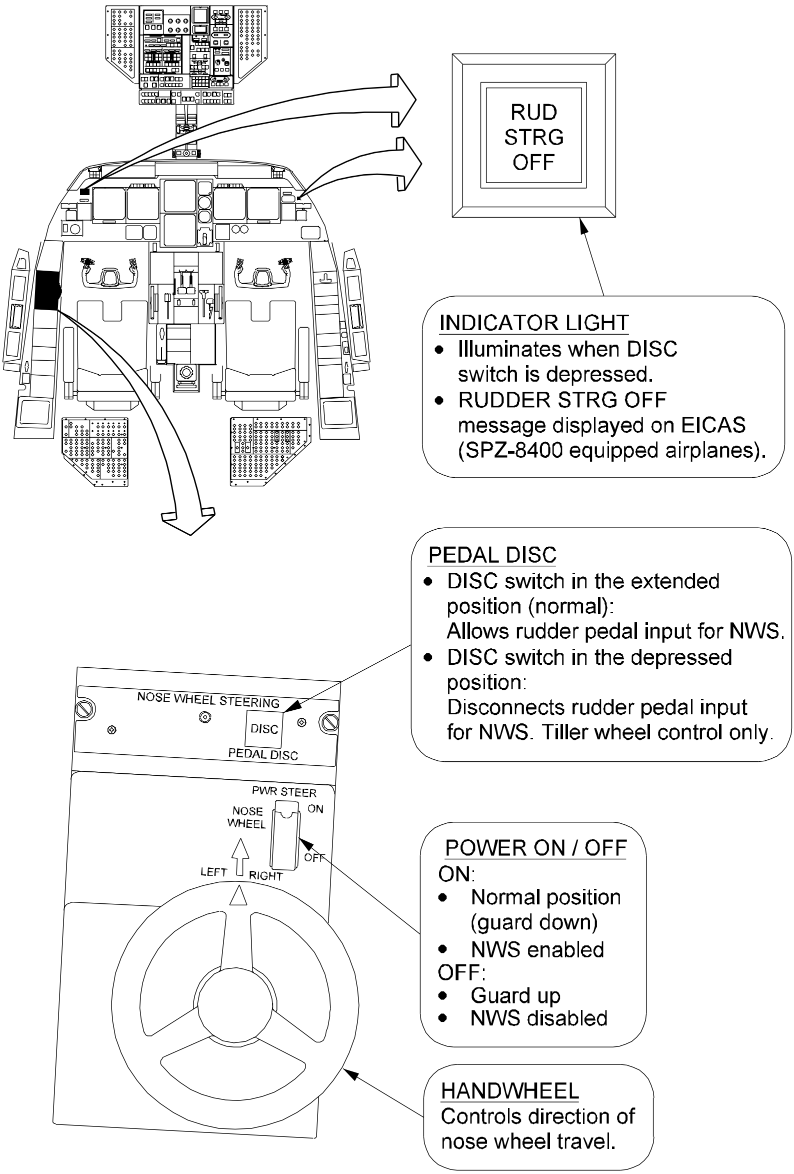
D. RUD STRG OFF Indicator (S/N with SPZ-8000)

A blue RUD STRG OFF indicator on the pilot instrument panel (ASC 302A) and on the copilot instrument panel (ASC 302A AM2) will illuminate if one of the following is accomplished:

- Airplanes not having ASC 434: Selecting HAND WHEEL ONLY on NOSE WHL STR SEL CONTROL panel (located on the pilot side console).
- Airplanes having ASC 434: Depressing PEDAL DISC switch on NOSE WHEEL STEERING panel (located on the pilot side console).



Nose Wheel Steering Controls and Components (Airplanes 1000 - 1444 Not Having ASC 434)
Figure 20 (Sheet 1 of 2)



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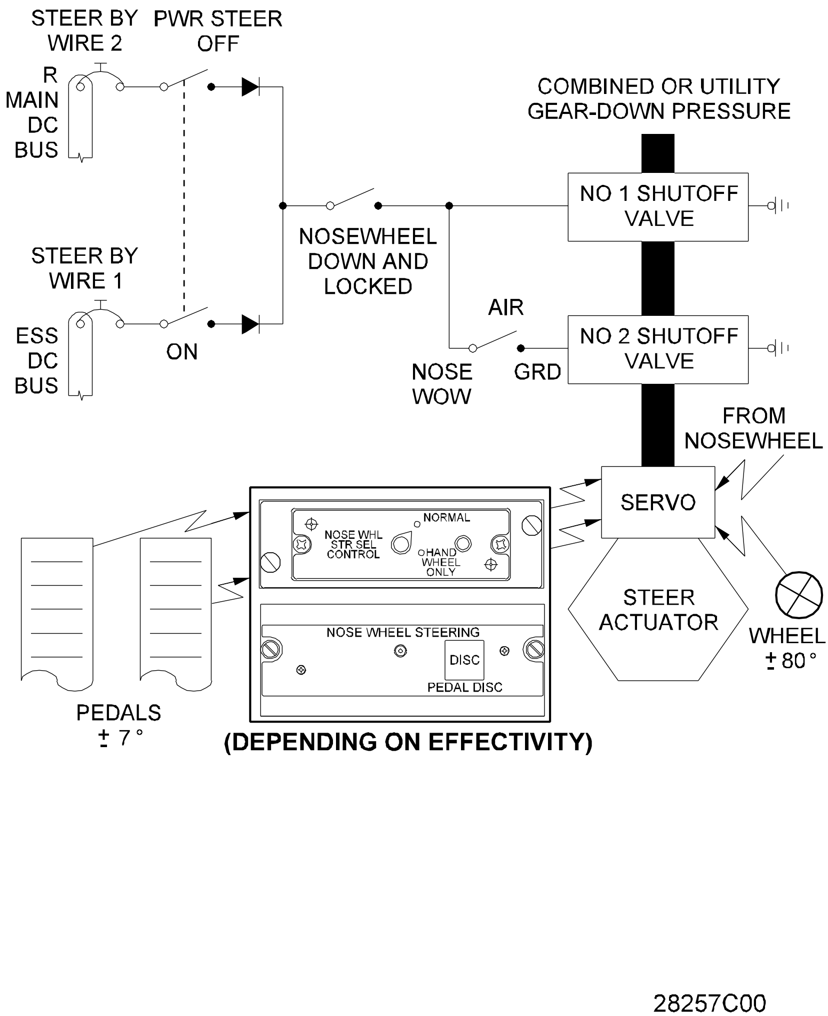
Nose Wheel Steering Controls and Components (Airplanes 1000 - 1444 Having ASC 434; Airplanes 1445 and Subsequent))
Figure 20 (Sheet 2 of 2)



Nose Wheel Steering Hydraulic Schematic
Figure 21

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2A-33-10: General

Lighting systems installed on the Gulfstream IV provide illumination for flight deck operations, external illumination for landing, taxi guidance and airplane recognition. Lighting for passenger convenience and emergency egress is also provided, as is illumination for airplane servicing and loading / unloading.

The lighting system is divided into following subsystems:

- 2A-33-20: Flight Compartment Lighting System
- 2A-33-30: Passenger Compartment Lighted Panels and Signs System
- 2A-33-40: Cargo and Service Compartment Lighting System
- 2A-33-50: Exterior Lighting System
- 2A-33-60: Emergency Lighting System

2A-33-20: Flight Compartment Lighting System

1. General Description:

Flight compartment lighting provides illumination of instruments, switches and controls, general illumination of flight deck areas to facilitate crew functions, and two (2) single point controls for full bright illumination of all installed lights. Lighting levels are controlled with switches located on the following installations:

- Pilot Cockpit Lighting Control Panel (Forward)
- Pilot Cockpit Lighting Control Panel (Aft)
- Copilot Cockpit Lighting Control Panel (Forward)
- Copilot Cockpit Lighting Control Panel (Aft)
- Dome Light with Integrated Switch
- Pilot and Copilot Yoke-Mounted Map Lights

Electrical power for cockpit lighting is 28V and 5V DC from the Main, Essential or Emergency DC buses. The GIV airplane is manufactured with a basic lighting configuration, as described in this section. Numerous optional lighting features are available for installation during the airplane completion process. Not all optional lighting features are covered in this section due to the scope of the available configurations, however the most commonly installed options are mentioned where appropriate.

2. Description of Subsystems, Units and Components:

A. Pilot Cockpit Lighting Control Panel (Forward, Airplanes 1000 through 1456):

(See Figure 1.)

The forward pilot lighting control panel has four (4) rotary light control switches and three (3) light control / test pushbuttons. The rotary light control switches are rheostats that enable selection of desired brightness levels from off to full bright. The two forward-most switches, L GLRSHLD FLT PNL and OVHD PNL are concentrically mounted dual controls. The upper or center control adjusts the brightness of switches and warning lights. The lower or outer part of the knob adjusts the brightness of edge lighting, instruments and digital displays. The two aft mounted rotary

switches, FLOOD L CONSOLE and FLOOD PED & OVHD, are single rheostat controls, adjusting the brightness of the dedicated flood lights from off to full bright.

(1) L GLRSHLD FLT PNL controls lighting on the following installations:

- Pilot Display Controller
- Flight Guidance Panel
- Yaw Damp / Mach Trim Panel
- Pilot HF Radio Control Panel
- Pilot Radar Control Panel
- Pilot Audio Control Panel (Baker Box)
- Nose Wheel Steering Panel
- Inertial System Display Unit
- Pilot Oxygen Control Panel

(2) OVHD PNL controls lighting on the following installations:

- Cockpit Overhead Panel Edge Lights
- Electric Power Monitor Panel (EPMP)
- APU - EGT / RPM Indicators
- Cabin Pressure Indicators
- ANTI-ICE indicators
- CABIN / CKPT Temperature Indicators
- BLEED AIR Indicators
- Cabin Pressurization Indicators
- STANDBY ELECTRICAL POWER Panel

(3) FLOOD L CONSOLE:

Controls brightness of florescent floodlight under left waterfall panel illuminating the pilot side console.

(4) FLOOD PED & OVHD:

Controls brightness of three floodlights: one mounted aft of the overhead console that shines down to illuminate the center pedestal, and two mounted overhead on either side of the cockpit entrance shining forward to illuminate each side of the overhead panels.

(5) WARN LT TEST switch:

Pressing the TEST switch illuminates all of the display lights on the overhead, forward instrument panels, center and side consoles as long as the switch is depressed.

(6) WARN LT BRT / DIM switch:

Pressing WARN LT BRT / DIM switch to the DIM selection enables the L GLRSHLD FLT PNL rotary selector to set the illumination level of the warning lights. Pressing the switch to the BRT selection places all warning lights at full bright regardless of the illumination level selected with the L GLRSHLD FLT PNL rotary switch.

(7) FLOOD ORIDE switch:

Selecting the FLOOD ORIDE switch to ON sets all cockpit lighting to full brightness, except the cockpit dome and map lights that have dedicated lighting controls.

B. Pilot Cockpit Lighting Control Panel (Forward, Airplane 1457 and Subsequent):

(See Figure 2.)

The forward pilot lighting control panel has eight (8) rotary light control switches and five (5) light control / test pushbuttons. The rotary light control switches are rheostats that enable selection of desired brightness levels from off to full bright. The two forward-most switches, L GLRSHLD FLT PNL and OVHD PNL, are concentrically mounted dual controls. The upper or center control adjusts the brightness of switches and warning lights. The lower or outer part of the knob adjusts the brightness of edge lighting, instruments and digital displays. The six remaining rotary switches (STBY INSTR, PED & OVHD FLOOD, CONT WHEEL MAP, L CONSOLE FLOOD, L CONSOLE PANEL and OVHD MAP), are single rheostat controls, adjusting the brightness of the dedicated flood lights from off to full bright.

(1) L GLRSHLD FLT PNL controls lighting on the following installations:

- Pilot Display Controller
- Flight Guidance Panel
- Yaw Damp / Mach Trim Panel
- Pilot HF Radio Control Panel
- Pilot Radar Control Panel
- Pilot Audio Control Panel (Baker Box)
- Nose Wheel Steering Panel
- Inertial System Display Unit
- Pilot Oxygen Control Panel

(2) OVHD PNL controls lighting on the following installations:

- Cockpit Overhead Panel Edge Lights
- Electric Power Monitor Panel (EPMP)
- APU - EGT / RPM Indicators
- Cabin Pressure Indicators
- ANTI-ICE indicators
- CABIN / CKPT Temperature Indicators
- BLEED AIR Indicators
- Cabin Pressurization Indicators
- STANDBY ELECTRICAL POWER Panel

(3) STBY INSTR:

Controls illumination intensity of the fluorescent floodlight from off to full bright.

(4) PED & OVHD FLOOD:

Controls brightness of three floodlights: one mounted aft of the overhead console that shines down to illuminate the center pedestal, and two mounted overhead on either side of the cockpit

entrance shining forward to illuminate each side of the overhead panels.

(5) CONT WHEEL MAP:

Controls the brightness of the left (pilot) control wheel map light.

(6) L CONSOLE FLOOD:

Controls intensity of the fluorescent floodlight under the waterfall panel.

(7) L CONSOLE PANEL:

Controls intensity of the instruments, displays, indicators and edge lighting on the left console panel.

(8) OVHD MAP:

Controls intensity of the pilot overhead map light.

(9) WARN LT TEST switch:

Pressing the TEST switch illuminates all of the display lights on the overhead, forward instrument panels, center and side consoles as long as the switch is depressed.

(10) WARN LT BRT / DIM switch:

Pressing WARN LT BRT / DIM switch to the DIM selection enables the L GLRSHLD FLT PNL rotary selector to set the illumination level of the warning lights. Pressing the switch to the BRT selection places all warning lights at full bright regardless of the illumination level selected with the L GLRSHLD FLT PNL rotary switch.

(11) PNL LTS:

Illuminates the pilot and copilot instrument panels with floodlights when the OFF legend is not illuminated.

(12) CB:

Illuminates the circuit breaker panels behind the pilot and copilot with floodlights when selected ON.

(13) FLOOD ORIDE switch:

Selecting the FLOOD ORIDE switch to ON sets all cockpit lighting to full brightness, except the cockpit dome and map lights that have dedicated lighting controls.

C. Copilot Cockpit Lighting Control Panel (Forward, Airplanes 1000 through 1456):

(See Figure 3.)

The forward copilot lighting control panel has four (4) rotary light control switches and three (3) light control / test pushbuttons. The panel is almost identical to that on the pilot side of the cockpit. The rotary light control switches are rheostats that enable selection of desired brightness levels from off to full bright. The two inboard switches, R GLRSHLD FLT PNL and PED PNL are concentrically mounted dual controls. The upper or center control adjusts the brightness of switches and warning lights. The lower or outer part of the knob adjusts the brightness of edge lighting, instruments and digital displays. The two outboard mounted rotary switches, STBY INSTR and FLOOD R CONSOLE, are single rheostat controls, adjusting the brightness of the dedicated flood lights from off to full bright.

(1) R GLRSHLD FLT PNL controls lighting on the following installations:

- Flight Guidance Panel
- Copilot Display Controller
- CABIN PRESSURE CONTROL
- Copilot Audio Control Panel (Baker Box)
- Copilot Radar Control Panel
- Copilot HF Radio Control Panel
- CVR Control Panel
- Copilot Oxygen Control Panel
- Passenger Oxygen Control Panel

(2) PED PNL rotary light control switch sets the brightness of the following installations:

- ADF No. 1
- ADF No. 2
- ATC No. 1
- ATC No. 2
- COM No. 1
- COM No. 2
- NAV No. 1
- NAV No. 2
- CDU No. 1
- CDU No. 2
- Display Brightness Controller
- IRS Mode Select Panel
- Marker Beacon Panel
- Autopilot Manual Control Panel

(3) FLOOD R CONSOLE rotary lighting selector:

Controls brightness of florescent floodlight under right waterfall panel illuminating the pilot side console.

(4) FLOOD ORIDE switch:

Selecting the FLOOD ORIDE switch to ON sets all cockpit lighting to full brightness, except the cockpit dome and map lights that have dedicated lighting controls.

(5) WARN LT BRT / DIM switch:

Pressing WARN LT BRT / DIM switch to the DIM selection enables the L GLRSHLD FLT PNL rotary selector to set the illumination level of the warning lights. Pressing the switch to the BRT selection places all warning lights at full bright regardless of the illumination level selected with the L GLRSHLD FLT PNL rotary switch.

(6) WARN LT TEST switch:

Pressing the TEST switch illuminates all of the warning lights on the overhead, forward instrument panels, center and side consoles as long as the switch is depressed.

D. Copilot Cockpit Lighting Control Panel (Forward, Airplane 1457 and Subsequent):

(See Figure 4.)

The forward copilot lighting control panel has seven (7) rotary light control switches and five (5) light control / test pushbuttons. The rotary light control switches are rheostats that enable selection of desired brightness levels from off to full bright. The two forward-most switches, PED PNL and R GLRSHLD FLT PNL, are concentrically mounted dual controls. The upper or center control adjusts the brightness of switches and warning lights. The lower or outer part of the knob adjusts the brightness of edge lighting, instruments and digital displays. The five remaining rotary switches (CONT WHEEL MAP, AFT PED PANEL, OVHD MAP, R CONSOLE PANEL and R CONSOLE FLOOD), are single rheostat controls, adjusting the brightness of the dedicated flood lights from off to full bright.

(1) PED PNL rotary light control switch sets the brightness of the following installations:

- ADF No. 1
- ADF No. 2
- ATC No. 1
- ATC No. 2
- COM No. 1
- COM No. 2
- NAV No. 1
- NAV No. 2
- CDU No. 1
- CDU No. 2
- Display Brightness Controller
- IRS Mode Select Panel
- Marker Beacon Panel
- Autopilot Manual Control Panel

(2) R GLRSHLD FLT PNL controls lighting on the following installations:

- Flight Guidance Panel
- Copilot Display Controller
- CABIN PRESSURE CONTROL
- Copilot Audio Control Panel (Baker Box)
- Copilot Radar Control Panel
- Copilot HF Radio Control Panel
- CVR Control Panel
- Copilot Oxygen Control Panel
- Passenger Oxygen Control Panel

(3) CONT WHEEL MAP:

Controls the brightness of the left (pilot) control wheel map light.

(4) AFT PED PANEL:

Controls intensity of instruments, displays, indicators and edge lighting on the aft pedestal panel.

(5) OVHD MAP:

Controls intensity of the pilot overhead map light.

(6) R CONSOLE PANEL:

Controls intensity of the instruments, displays, indicators and edge lighting on the right console panel.

(7) R CONSOLE FLOOD:

Controls intensity of the fluorescent floodlight under the waterfall panel.

(8) FLOOD ORIDE switch:

Selecting the FLOOD ORIDE switch to ON sets all cockpit lighting to full brightness, except the cockpit dome and map lights that have dedicated lighting controls.

(9) CB switch:

Illuminates the circuit breaker panels behind the pilot and copilot with floodlights when selected ON.

(10) PNL LTS switch:

Illuminates the pilot and copilot instrument panels with floodlights when the OFF legend is not illuminated.

(11) WARN LT BRT / DIM switch:

Pressing WARN LT BRT / DIM switch to the DIM selection enables the L GLRSHLD FLT PNL rotary selector to set the illumination level of the warning lights. Pressing the switch to the BRT selection places all warning lights at full bright regardless of the illumination level selected with the L GLRSHLD FLT PNL rotary switch.

(12) WARN LT TEST switch:

Pressing the TEST switch illuminates all of the display lights on the overhead, forward instrument panels, center and side consoles as long as the switch is depressed.

(13) PED & OVHD FLOOD:

Controls brightness of three floodlights: one mounted aft of the overhead console that shines down to illuminate the center pedestal, and two mounted overhead on either side of the cockpit entrance shining forward to illuminate each side of the overhead panels.

(14) WARN LT TEST switch:

Pressing the TEST switch illuminates all of the display lights on the overhead, forward instrument panels, center and side consoles as long as the switch is depressed.

(15) WARN LT BRT / DIM switch:

Pressing WARN LT BRT / DIM switch to the DIM selection enables the R GLRSHLD FLT PNL rotary selector to set the illumination level of the warning lights. Pressing the switch to the BRT selection places all warning lights at full bright regardless of the illumination level selected with the R GLRSHLD FLT PNL rotary switch.

E. Copilot Cockpit Lighting Control Panel (Aft, Airplanes 1000 through 1456 Only):

(See Figure 5.)

The aft Pilot Lighting Control Panel contains a single L CONSOLE rotary selector which controls the lighting levels of the instruments, displays, indicators and edge lighting on the left console panel.

The aft Copilot Lighting Control Panel has three (3) rotary rheostat controls for pedestal and console lighting:

(1) AFT PED PNL rotary selector controls illumination intensity of the instruments, displays, indicators and edge lighting on the aft pedestal panel.

(2) AFT PED DIG rotary selector controls illumination intensity of the digital displays in the pedestal panel.

(3) R CONSOLE rotary selector controls illumination intensity of the instruments, displays, indicators and edge lighting on the right console panel.

F. Dome Light:

A cockpit dome light is mounted in the flight compartment headliner aft of the overhead console. The dome light provides general area illumination for the flight compartment, and is controlled by a dedicated switch adjacent to the light.

A second dome light is often installed as optional equipment, placed in the headliner just aft of the cockpit in the vestibule area, providing general area illumination for a jumpseat occupant.

G. Map Lights:

Each pilot control yoke has an integrated map light in the control column located below the control wheel. The map lights are individually controlled by a switch on the aft side of each inboard arm of the yoke.

Additional map lights are frequently installed adjacent to the gasper air outlets above the cockpit side windows. A map light may also be fitted into the aft cockpit headliner, near the dome light, for use by a jumpseat occupant.

3. Controls And Indications:

A. Circuit Breakers (CBs):

The flight compartment lighting system is protected by the following circuit breakers:

Circuit Breaker	CB Panel	Location	Power Source
L FLT PNL / GLRSHLD	P	A-7	ESS DC
L CONSOLE FLOOD	P	A-10	L MAIN DC
L CONSOLE PWR SUPPLY	P	D-1	L MAIN DC
R FLT PNL / GLRSHLD	P	B-7	R MAIN DC
R CONSOLE FLOOD	P	B-10	R MAIN DC
R CONSOLE PWR SUPPLY	P	E-1	L MAIN DC
PEDESTAL	P	A-8	L MAIN DC
AFT PED PWR SUPPLY #1	P	E-2	L MAIN DC

Circuit Breaker	CB Panel	Location	Power Source
AFT PED PWR SUPPLY #2	P	E-3	L MAIN DC
OVHD / PED FLOOD LTS	P	A-9	L MAIN DC
OVHD PNL PRI	P	B-9	ESS DC
FLOOD LTS OVERRIDE	P	A-6	ESS DC
DOME LT	P	B-6	ESS DC
WARN LTS TEST	P	G-1	ESS DC
STBY INSTR	P	B-8	EMER DC

4. Limitations:

There are no operational limitations for the flight compartment lighting system as of this revision.

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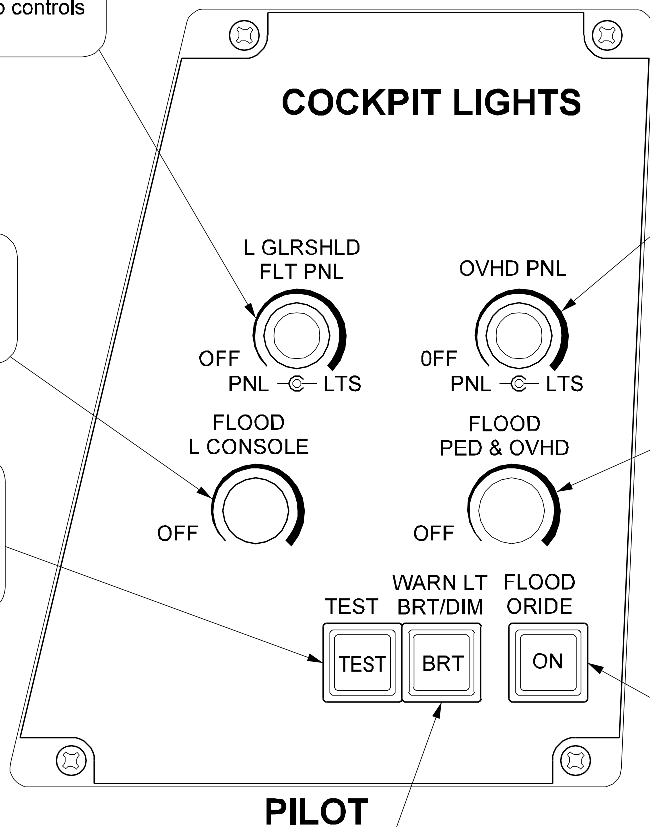
L GLRSHLD FLT PNL
Controls illumination intensity of instruments and displays on pilot instrument panel and left half of glareshield. Inner knob controls illumination of switches and warning lights while outer knob controls panel edge lighting and digital displays.

OVHD PNL
Controls illumination of instruments, displays, indicators and edge lighting on overhead panel. Inner knob controls switches and warning lights while outer knob controls panel edge lighting, instrument lighting and digital displays.

FLOOD L CONSOLE
Controls illumination intensity of fluorescent floodlight under waterfall panel.

FLOOD PED & OVHD
Controls intensity of incandescent floodlights placed in overhead.

TEST
Pressing switch causes all annunciator lights on the cockpit overhead panel, center pedestal and side consoles to illuminate.

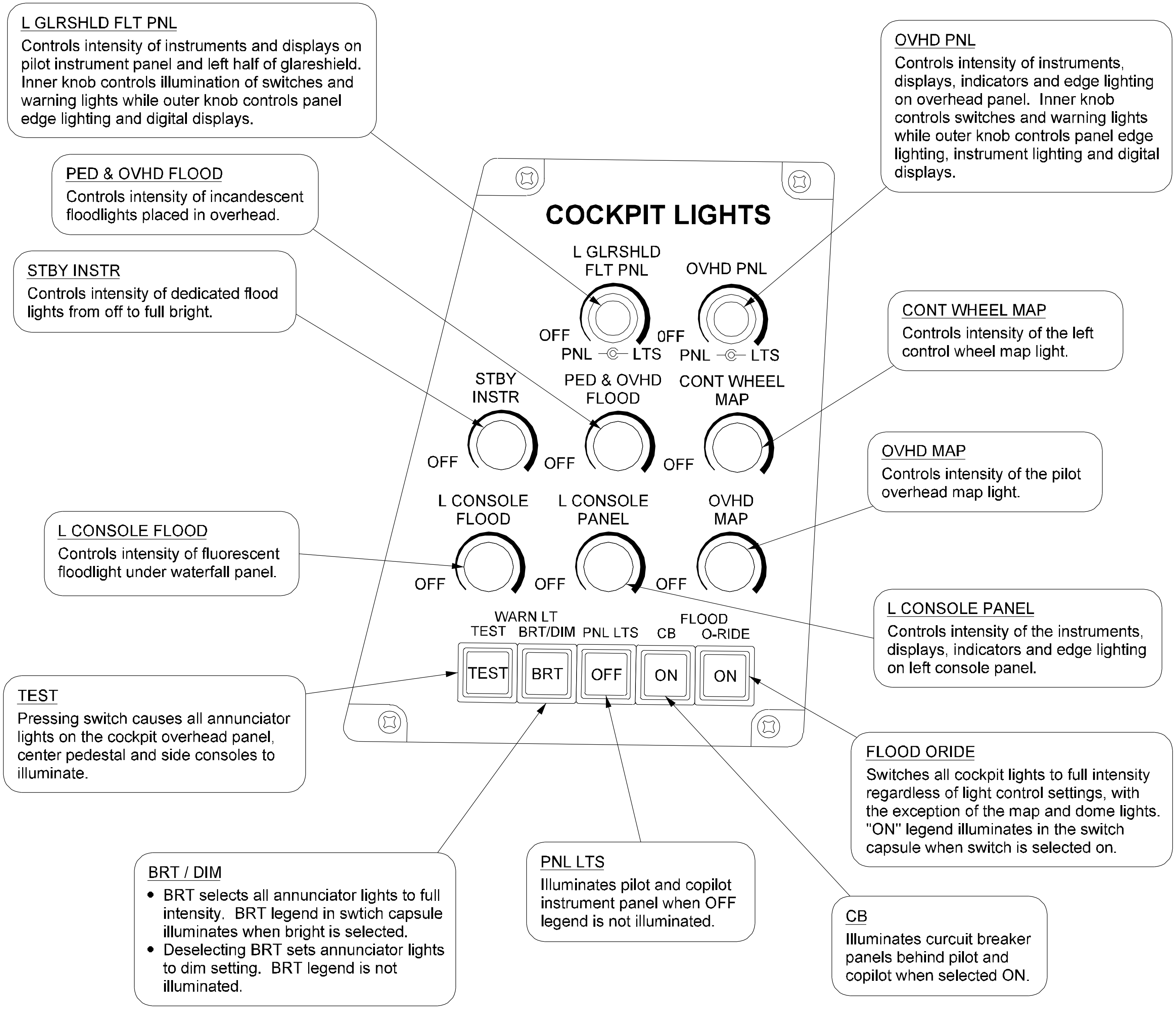


FLOOD ORIDE
Switches all cockpit lights to full intensity regardless of light control settings, with the exception of the map and dome lights. "ON" legend illuminates in the switch capsule when switch is selected on.

BRT / DIM
- BRT selects all annunciator lights to full intensity. BRT legend in swtich capsule illuminates when bright is selected.
- Deselecting BRT sets annunciator lights to dim setting. BRT legend is not illuminated.

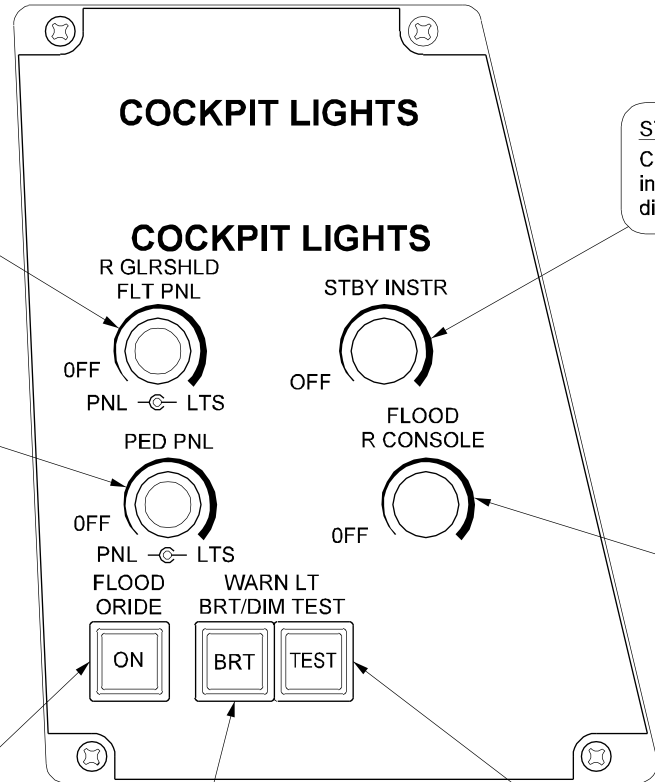
26840C01

Pilot Side Console Light Controls (Forward, Airplanes 1000 through 1456)
Figure 1



39574C00

Pilot Side Console Light Controls (Forward, Airplane 1457 and Subsequent)
Figure 2



COPILOT

R GLRSHLD FLT PNL
Controls intensity of instruments and displays on copilot instrument panel and right half of glareshield. Inner knob controls switches and warning lights while outer knob controls panel edge lighting and digital displays.

STBY INSTR
Controls illumination intensity of standby instruments, flight critical instruments and displays.

PED PNL
Controls intensity of instruments and displays on pedestal and panel edge lighting. Inner knob controls switches and warning lights while outer knob controls panel edge lighting, instrument lighting and digital displays.

FLOOD R CONSOLE
Controls intensity of fluorescent floodlight under waterfall panel.

FLOOD ORIDE
Selects cockpit lights to full intensity regardless of light control settings, with exception of the map and dome lights. "ON" legend illuminates in the switch capsule when switch is selected on.

TEST
Pressing switch causes all annunciator lights on the cockpit overhead panel, center pedestal and side consoles to illuminate.

BRT / DIM
- BRT selects all annunciator lights to full intensity. BRT legend in swtich capsule illuminates when bright is selected.
- Deselecting BRT sets annunciator lights to dim setting. BRT legend is not illuminated.

26842C01

Copilot Side Console Light Controls (Forward, Airplanes 1000 through 1456)
Figure 3

PED PNL
Controls intensity of instruments and displays on pedestal and panel edge lighting. Inner knob controls switches and warning lights while outer knob controls panel edge lighting, instrument lighting and digital displays.

R GLRSHLD FLT PNL
Controls intensity of instruments and displays on copilot instrument panel and right half of glareshield. Inner knob controls illumination of switches and warning lights while outer knob controls panel edge lighting and digital displays.

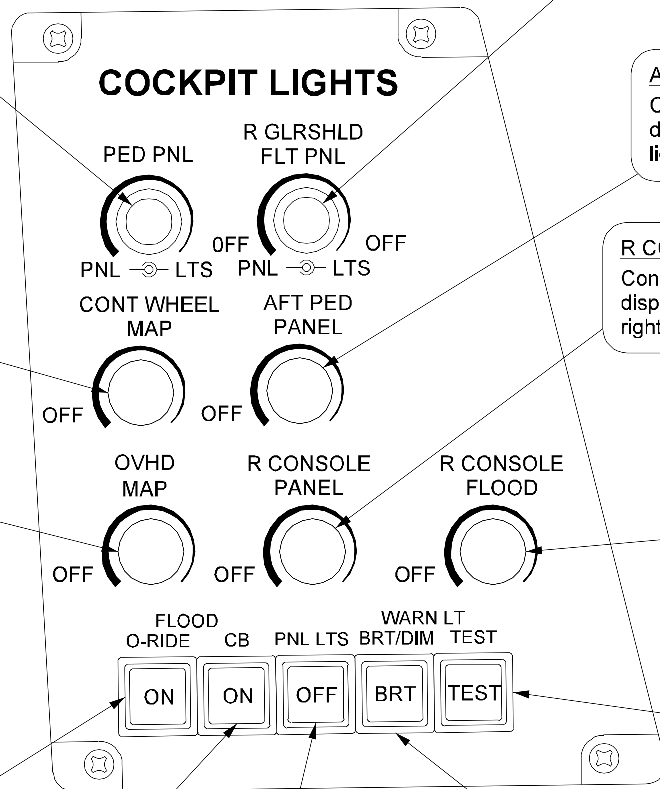
AFT PED PANEL
Controls intensity of instrruments, displays, indicators and edge lighting on aft pedestal panel.

CONT WHEEL MAP
Controls intensity of the right control wheel map light.

R CONSOLE PANEL
Controls lighting levels of the instruments, displays, indicators and edge lighting on right console panel.

OVHD MAP
Controls intensity of the copilot overhead map light.

R CONSOLE FLOOD
Controls intensity of fluorescent floodlight under waterfall panel.



FLOOD ORIDE
Switches all cockpit lights to full intensity regardless of light control settings, with the exception of the map and dome lights. "ON" legend illuminates in the switch capsule when switch is selected on.

TEST
Pressing switch causes all annunciator lights on the cockpit overhead panel, center pedestal and side consoles to illuminate.

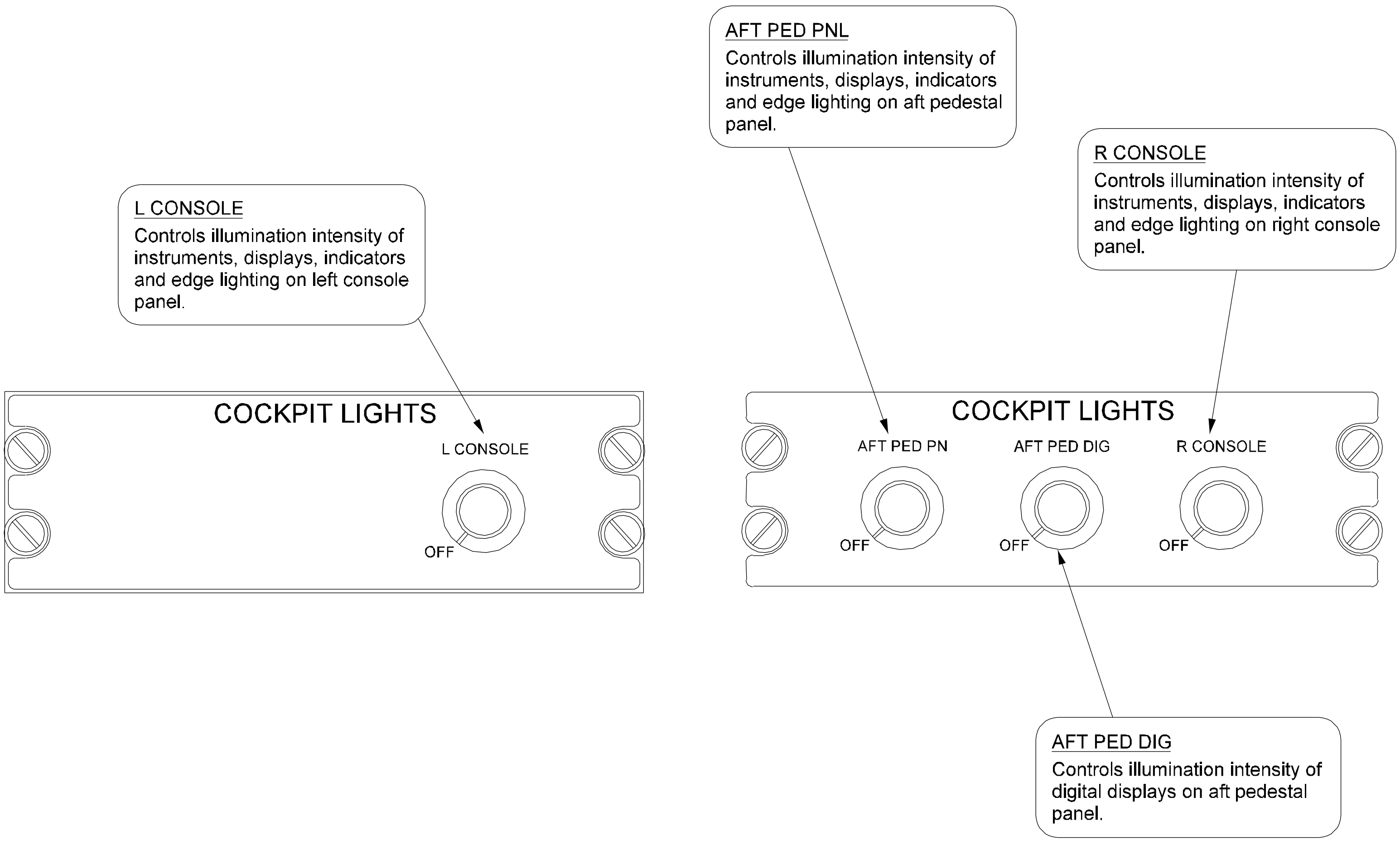
CB
Illuminates curcuit breaker panels behind pilot and copilot when selected ON.

PNL LTS
Illuminates pilot and copilot instrument panel when OFF legend is not illuminated.

BRT / DIM
- BRT selects all annunciator lights to full intensity. BRT legend in swtich capsule illuminates when bright is selected.
- Deselecting BRT sets annunciator lights to dim setting. BRT legend is not illuminated.

39575C00

Copilot Side Console Light Controls (Forward, Airplane 1457 and Subsequent)
Figure 4



Pilot and Copilot Side Console Light Controls (Aft, Airplanes 1000 through 1456 Only)
Figure 5

2A-33-30: Passenger Compartment Lighted Panels And Signs System

1. General Description:

The basic GIV airplane is equipped with the basic circuitry and switch controls for installing FASTEN SEAT BELT and NO SMOKING ordinance signs in the passenger compartment. Each airplane interior is customized with various seating arrangements, convenience compartments (lavatories and galleys) and bulkheads. The placement of the NO SMOKING and FASTEN SEAT BELT signs is dependent upon the configuration of the passenger compartment, thus only basic wiring provisions are made for the ordinance signs.

The placement of emergency exit signs is likewise limited by the configuration of the passenger compartment because signs must be placed in highly visible locations, with view unobstructed by cabin components. For basic airplane certification, only the required EXIT signs are installed: over the main entrance door and at the overwing exits. Other emergency exit signs are installed after completion of the cabin interior, and include multiple installations pointing out the nearest exit. Many airplanes are equipped with other emergency exit signage options, including wiring selected passenger compartment reading and table lights into the emergency lighting system to provide cabin illumination in critical situations, and / or installing an emergency floor path lighting system to provide exit guidance for an evacuation in instances where smoke might obscure visibility in the cabin.

This system description is limited to only the basic production installed passenger compartment lighting:

- No Smoking / Fasten Seat Belt Signs
- Emergency Exit Lights

For information regarding optionally installed ordinance signs and emergency lighting, see the Completion Center Maintenance Handbook for the particular airplane.

2. Description of Subsystems, Units and Components:

A. No Smoking / Fasten Seat Belt Signs and Controls:

Basic wiring provisions are installed for NO SMOKING / FASTEN SEAT BELT ordinance signs. The provisions include a five (5) amp circuit breaker, labelled SIGN LTS, installed in position D-5 on the pilot aft circuit breaker panel, and control switches for the signs on the lower forward section of the cockpit overhead panel. See the illustration in Figure 9. The control switches are in the panel section labelled PASS WARN, and titled ST BLT for the FASTEN SEAT BELT switch and NO SMK/ST BLT for the combined NO SMOKING / FASTEN SEAT BELT switch.

Ordinance signs controlled by the cockpit switches are placed throughout the passenger compartment in accordance with customer designated interior layouts. In many airplanes, a RETURN TO SEAT sign is installed in the lavatories, and illuminates when the FASTEN SEAT BELT sign is selected ON. On most airplanes, the NO SMOKING / FASTEN SEAT BELT switches are integrated into the cabin audio system, often with dedicated speakers and audio power supplies. For these airplanes, when either of the switches are selected ON of OFF, an audio signal sounds in the cabin.

B. **Emergency Exit Lights:**

Self contained battery operated emergency exit lights are installed above the main entrance door and at the emergency exit windows on the left and right side of the airplane. Additional emergency exit signs are installed in accordance with customer preferences upon completion of the airplane interior, and in conformity to FAA regulations. The emergency exit signs are powered by two (2) “D” cell size batteries, and illuminate in response to the mode selected on the switch incorporated into the sign. The sign switch may be selected to AUTO or ON. In the AUTO mode, the sign will automatically illuminate if subjected to a force of one and one-half (1½) G’s for more than one thirtieth (1/30) of a second. If illuminated in the AUTO mode, the sign may be reset by cycling the switch to ON and back to AUTO. If the switch is selected to ON, the sign will illuminate. The emergency exit signs are detachable from the sign mounts and may be used as portable illumination devices with the switch selected to ON.

NOTE:

The “D” cell batteries used in these lights should be inspected and/or replaced during periodic maintenance to insure full battery life.

3. **Circuit Breakers:**

Circuit Breaker Name	CB Panel	Location	Power Source
SIGN LTS	P	D-5	LEFT MAIN DC

4. **Limitations:**

There are no limitations established for this system as of this revision.

2A-33-40: Cargo and Service Compartment Lighting System

1. **General Description:**

The GIV has lighting installed in the baggage compartment for loading / unloading and illumination for inspection and maintenance of components in the tail compartment. The lighting installation includes the following subsystems and components:

- Baggage Compartment Lighting
- Tail Compartment and Service Panel Lighting

2. **Description of Subsystems, Units and Components:**

A. **Baggage Compartment Lights:**

Baggage compartment lighting is installed in accordance with customer preferences during final outfitting. A typical installation consists of two fluorescent lights mounted on the baggage compartment overhead, controlled by a switch mounted in a panel on the forward baggage compartment bulkhead. Power for the circuits is typically from the Main DC bus via a BAGGAGE LTS circuit breaker, usually located on the cabin circuit breaker panel.

B. Tail Compartment Lights:

Tail compartment utility lights provide illumination for maintenance and inspection of equipment installed in the tail compartment.

(1) Tail Compartment Utility Lights:

Two incandescent dome lights are mounted on the tail compartment overhead, one forward and one aft. The lights are controlled by either of two switches. One switch, located in the tail compartment on the left side near the entrance at the top of the ladder, switches the lights on only when the tail compartment access door is open. This arrangement prevents leaving the lights on unnecessarily. A second switch, located on the panel on the forward bulkhead of the baggage compartment (shown in Figure 6), controls the dome lights regardless of access door position. This allows the flight crew to view the interior of the tail compartment during flight through the inspection window in the center of the pressure dome bulkhead.

NOTE:

With the tail compartment access door open and the light switch in the tail compartment selected on, the lights will illuminate regardless of the position of the switch in the baggage compartment.

In addition to the incandescent dome lights, airplanes 1495 and subsequent have wiring provisions for three florescent light fixtures that can be installed as an option during the completion process. The optional fixtures are typically installed on the compartment overhead in locations requiring additional illumination. Control of the fluorescent fixtures is with the same pair of switches used for the incandescent dome lights, with the exception that power to the fluorescent lights is routed through the access door relay, limiting the fluorescent lights to ground operation only with the access door open. (The incandescent lights remain available for compartment inspection during flight.)

Both the incandescent and optional fluorescent light fixtures are powered by 28V DC from the Essential DC bus.

3. Controls and Indications:

(See Figure 6.)

A. Circuit Breakers (CBs):

Cargo and service compartment lighting systems are protected by the following CBs:

Circuit Breaker Name	CB Panel	Location	Power Source
UTILITY LTS	P	E-9	ESS 28V DC

4. Limitations:

A. Flight Manual Limitations:

There are no limitations established for this system as of this revision.

B. System Notes:

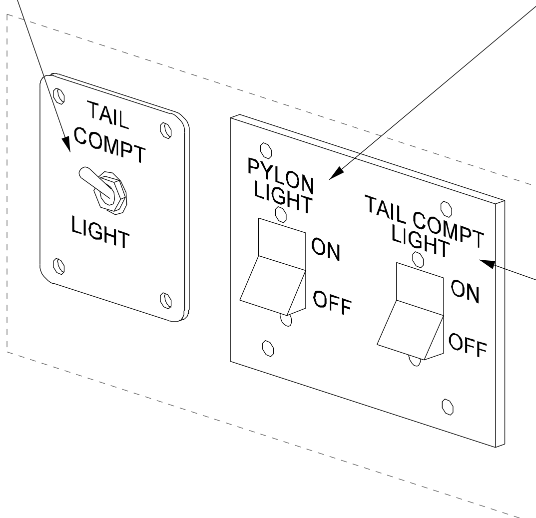
There is no cockpit indication when baggage compartment lights or tail compartment utility lights are illuminated.

TAIL COMPARTMENT LIGHT SWITCH
(S/N 1000 - 1279)
ON:
- With tail compartment door open (on the ground) turns on incandescent dome and fluorescent lights (if installed).
- With tail compartment door closed (on the ground or in flight) turns on incandescent dome lights only.

OFF:
- Turns off incandescent dome and fluorescent lights (if installed).

PYLON LIGHT SWITCH
(S/N 1280 and subsequent)
ON:
- Turns on pylon mounted ramp lights in lower aft fairing of left and right engine pylons and fuselage light above cargo door.

OFF:
- Turns off all three lights.



TAIL COMPARTMENT LIGHT SWITCH
(S/N 1280 and subsequent)
ON:
- With tail compartment door open (on the ground) turns on incandescent dome and fluorescent lights (if installed).
- With tail compartment door closed (on the ground or in flight) turns on incandescent dome lights only.

OFF:
- Turns off incandescent dome and fluorescent lights (if installed).

29601C00

Baggage and Tail
Compartment Lighting
Switch Panels
Figure 6

2A-33-50: Exterior Lighting System

1. General Description:

Lights installed on the GIV airplane exterior provide illumination for landing and taxiing, airplane recognition and collision avoidance, and inspection of airplane wing surfaces for ice formation during flight. Service compartment lighting is integrated into panels / access doors to facilitate refueling and resupply. The following lights are installed as standard equipment on the GIV:

- Anti-Collision Lights
- Strobe Lights
- Navigation Lights
- Ice Inspection Lights
- Landing Lights
- Pulse Lights
- Taxi Lights
- Wingtip Floodlights
- Wheel Well Lights
- Ramp Lights
- Logo Lights
- Refueling Panel Lights

The location of exterior lights is shown in Figure 7.

2. Description of Subsystems, Units and Components:

A. Anti-Collision Lights:

Two red anticollision lights are installed on the airplane. The top anticollision light is a red strobe light consisting of a Xenon flashtube with integrated power supply and a reflector, mounted on the upper center line at Fuselage Station 308. The bottom anticollision light is either an oscillating light assembly consisting of a motor, two bulbs, noise filter and red lens (airplanes not having ASC 10) , or a red strobe light similar to the top anticollision light (airplanes having ASC 10) mounted on the lower center line at Fuselage Station 465.

Both top and bottom lights are controlled by the BCN switchlight on the bottom section of the pilot overhead panel shown in Figure 8. When the BCN switchlight is selected ON, 28V DC power from the Right Main DC bus opens the ROTATING LTS BEACON (airplanes not having ASC 10) or BEACON (airplanes having ASC 10) relay. The open relay routes 115V AC power from the Right Main AC bus to the top anticollision strobe light (and to the bottom anticollision strobe light for airplanes having ASC 10). For airplanes not having ASC 10, the ROTATING LTS BEACON relay routes 28V DC power from the Right Main DC bus to the bottom anticollision light.

For all airplanes, the bottom anticollision light (or strobe) will come on whenever the OUTSIDE BATT switch on the service panel at the rear of the airplane is turned on. The flashing anticollision light (strobe) is a reminder to turn the battery switch off when no longer needed. Power for this function of the anticollision light (strobe) is 28V DC from the Essential DC bus for airplanes not having ASC 10, or 115V AC from the Essential AC bus for airplanes having ASC 10. Additionally, for SN 1034 and SN 1156 and

subsequent, operation of the EXTERNAL BATT switch in the service panel near the nose of the airplane will cause the bottom anticollision light (strobe) to illuminate.

B. Strobe Lights:

White strobe lights are installed in the tip of the tail cone and at each wingtip.

The strobe lights are controlled by the STROBE switch on the cockpit overhead panel depicted in Figure 8. When the switch is selected ON, 28V DC power from the Right Main DC bus opens the STROBE LT CONT relay that in turn routes 115V AC power from the Right Main AC bus to the three strobe lights.

C. Navigation Lights:

A red navigation light is installed in the left (port) wingtip, a green navigation light in the right (starboard) wing tip, and two white navigation lights in the aft tail cone (SN 1000-1349) or aft horizontal stabilizer bullet (SN 1350 and subsequent). On airplanes SN 1000-1155 having ASC 14, and SN 1156 and subsequent, each wingtip installation is equipped with dual lights. Regulations require that a navigation light at each position be operable for dispatch, thus with dual lights, failure of one light will not deter flight operations.

The navigation lights are controlled by the NAV switch on the bottom of the pilot overhead panel. See Figure 8. When the switch is selected to ON, 28V DC power from the Left Main DC bus opens the NAV LTS relay. When this relay is opened, power is routed from the 28V AC ϕA NAV / INSP LTS transformer bus (stepped down from the Left Main 115V AC bus) via the NAV LTS circuit breaker to the navigation lights.

D. Ice Inspection Lights:

Ice inspection lights are installed on either side of the fuselage and aimed to provide illumination of the wing leading edges. On airplanes with the cargo door installation, the right wing ice inspection light is positioned on the lower aft area of the cargo door. The two lights are controlled by the ICE switchlight on the pilot overhead panel shown in Figure 8. The lights are powered by the 28V AC ϕA NAV / INSP transformer bus (stepped down from the Left Main 115V AC bus) via the WING INSP LTS circuit breaker.

E. Landing Lights:

Landing lights are installed on each wing in the leading edge at the wing root fairing. The lights are mounted behind removable clear lens installations that conform to the curvature of the wing. Each light is individually controlled with switches labeled L LDG and R LDG on the overhead EXTERIOR LTS panel. See Figure 8.

Each light has a separate circuit breaker and dedicated relay powered from a corresponding bus: Right Main 115V AC bus ϕA for the right light, Left Main 115V AC bus ϕB for the left light.

F. Pulse Lights:

Provisions for the Precise Flight Pulselight System are installed on airplane SNs 1437 through 1454. The provisions consist of the control switch and necessary wiring. The pulselight system is completely installed on airplanes 1455 and subsequent.The pulselight control switch is located on

the COP in the EXTERIOR LTS section, and is labeled PULSE.

G. Taxi Lights:

A taxi light assembly, composed of three lights, is mounted on the steerable portion of the nose landing gear shock strut. The three lights are identical and interchangeable, but each mounted differently to illuminate a wide area ahead of the airplane (left, forward, and right). The taxi light control switch, labeled TAXI (shown in Figure 8), is located in the EXTERIOR LTS section of the overhead panel. The taxi lights are powered by 28V DC from the Left Main DC bus, through a TAXI LTS CONTR relay, with each of the three taxi lights having a dedicated circuit breaker (L TAXI LT PWR, CTR TAXI LT PWR, and R TAXI LT PWR).

NOTE:

On airplanes having ASC 131, the taxi light circuitry is modified to ensure that the taxi lights extinguish whenever the landing gear is retracted. For these airplanes, the ground for the TAXI LIGHTS CONTR relay is routed through the nose wheel downlock switch circuit. On airplanes not having ASC 131, the taxi lights do not automatically extinguish when the nose gear is retracted, and must be turned off when not in use.

H. Wingtip Taxi Floodlights:

On airplanes SN 1000-1279 having ASC 1, and SN 1280 and subsequent, wingtip floodlights are installed at the end of each wing, supplementing the taxi lights, and providing increased assurance of wingtip clearance during ground operations in low visibility. The wingtip floodlights are mounted on the underside of each wing near the winglet edge and positioned to illuminate the area beneath the lights.

For airplanes having ASC 1, the wingtip floodlights are controlled by the W/TIP switch (see Figure 8) on the EXTERIOR LTS section of the overhead panel. With the switch ON, power from the Main #2 28V AC ϕB transformer bus power (stepped down from 115V AC) via the WING TIP TAXI LTS circuit breaker opens the wingtip floodlights relay, powering the left and right wingtip floodlights.

For airplanes SN 1280 and subsequent, the wingtip taxi lights consist of a dual light installation on each wingtip. One of the lights is directed forward of the wingtip, and the other is directed down, illuminating the area beneath the wingtip. Control of the dual wingtip light installations is with the TAXI light switch in the EXTERIOR LIGHTS section of the cockpit overhead panel. (For SN 1280 and subsequent, the TAXI light switch controls both the nose wheel taxi lights and the wingtip taxi lights). The lights are powered by 28V DC from the Left Main DC bus through the WING TIP FLD LTS circuit breaker.

I. Wheel Well Lights:

A utility light in each wheel well provides illumination for maintenance and servicing. The lights are interchangeable and are powered by 28V DC from the Essential DC bus. On airplanes not having ASC 1, the wheel well lights are controlled by the W WELL switch on the EXTERIOR LTS section of the

overhead panel. On airplanes having ASC 1, the wheel well lights are controlled by a WHEEL WELL LIGHTS switch on the pilot side console. In addition to these locations, Airplanes 1455 and subsequent also utilize an externally-mounted wheel well lights switch under the forward external switch panel access door. Control switches for all three configurations are shown in Figure 8. When the switch (either) is selected ON, power is supplied from the 28V Essential DC bus through the WHEEL WELL LTS circuit breaker, energizing the wheel well lights relay and powering the nose, left and right main wheel well lights.

J. Ramp Lights:

On airplanes SN 1280 and subsequent, ramp lights are installed to illuminate the areas around the baggage door and beneath the engine pylons to facilitate loading and servicing. (The light installation is an optional feature for airplanes produced prior to SN 1280). The installation consists of three identical lights, one installed on the underside of the aft pylon fairings of each engine, and one in the underside of the left engine pylon above the cargo door. The ramp lights are controlled by the PYLON LIGHT switch on the panel in the baggage compartment. See Figure 6. The lights are powered by 28V DC from the Right Main DC bus through the PYLON LTS circuit breaker on the pilot circuit breaker panel.

K. Logo Lights:

Two logo lights are installed as production standard equipment beginning with airplane SN 1280. The lights are mounted in the underside of the horizontal stabilizer on each side and canted inward to illuminate the left and right sides of the vertical stabilizer. The lights are powered by 115V AC ϕC from the Right Main AC bus through the LOGO LTS circuit breaker. The lights are controlled by the LOGO switchlight on the cockpit overhead panel, shown in Figure 8.

NOTE:

On airplanes prior to 1280, only the LOGO switchlight is installed in the overhead panel. Completion of the circuitry and light installation is at the option of the operator.

L. Service Panel (Fueling) Lighting:

A dome-type light is installed in the refueling panel as standard equipment on airplanes SN 1034 and 1156 and subsequent. This feature may be installed as an option on other airplanes, as well as other service panel light installations at the aft lavatory service panel, oxygen service panel, water service panel and on the forward external electrical access panel. The refueling panel light (and other service panel lights if installed) operates from the 28V DC Essential DC bus in order that it may be powered by the airplane batteries only. A proximity switch installed in the panel door opens when the door is unlatched, completing the circuit to power the light. Integrated into the circuit is a SERVICE DOORS annunciation on the cockpit EICAS display whenever the panel is open.

3. Controls and Indications:

(See Figure 8.)

A. **Circuit Breakers (CBs):**

The exterior lighting system is protected by the following CBs:

Circuit Breaker Name	CB Panel	Location	Power Source
NAV / INSP LTS XFMR	P	C-6 **(1)** C-8 **(2)**	L MAIN 115V AC φA
TAXI LTS XFMR	P	D-6 **(1)** F-10 **(2)**	115V AC φB
L LDG LT CONT	P	E-6 **(1)** D-7 **(2)**	L MAIN 28V DC
R LDG LT CONT	P	F-6 **(1)** F-7 **(2)**	R MAIN 28V DC
L LDG LT PWR	P	C-7	L MAIN 115V AC φA
R LDG LT PWR	P	D-7 **(1)** E-7 **(2)**	R MAIN 115V AC φB
STROBE LTS CONT	P	E-7 **(1)** F-8 **(2)**	R MAIN 28V DC
BOT A/C LT	P	F-7 **(1)** D-6 **(2)**	R MAIN 28V DC
NAV LTS	P	C-8 **(1)** E-8 **(2)**	L MAIN 28V AC φA
NAV / INSP LTS CONT	P	D-8	L MAIN 28V DC
STROBE LTS	P	E-8 **(1)** F-9 **(2)**	R MAIN 115V AC φA
TOP A/C LT	P	F-8 **(1)** C-6 **(2)**	R MAIN 115V AC φC
WING INSP LTS	P	C-9 **(1)** E-9 **(2)**	L MAIN 28V AC φA
TAXI LTS CONT	P	D-9 **(1)** C-9 **(2)**	L MAIN 28V DC
UTILITY LTS	P	E-9 **(1)** D-9 **(2)**	ESS 28V DC
BOT A/C LT GND OPER	P	F-9 **(1)** E-6 **(2)**	ESS 28V DC
L TAXI LT PWR	P	C-10	R MAIN 28V AC φB
CTR TAXI LT PWR	P	D-10	R MAIN 28V AC φB
R TAXI LT PWR	P	E-10	R MAIN 28V AC φB
WHEEL WELL LTS	P	F-10 **(1)** F-6 **(2)**	ESS 28V DC
BCN LTS CONT	P	F-7 **(3)** D-6 **(4)**	R MAIN 28V DC
BCN LTS	P	F-8 **(3)** C-6 **(4)**	R MAIN 115V AC φC
BOT BCN GND OPER	P	F-9 **(3)** E-6 **(4)**	ESS 115V AC
LOGO LTS	P	C-5 **(5)**	R MAIN 115V AC φC
PYLON LTS	P	D-5 **(5)**	R MAIN 28V DC
WING TIP FLD LTS	P	E-5 **(5)**	L MAIN 28V DC
WING TIP TAXI LTS	P	K-14 **(6)**	R MAIN 28V AC φB

NOTE(S):

(1) SN 1000, 1002-1017

(2) SN 1001, 1018 and subs

(3) SN 1000, 1002-1017 having ASC 10

(4) SN 1001, 1018 and subs having ASC 10

(5) SN 1280 and subs

(6) SN 1000-1279 having ASC 1

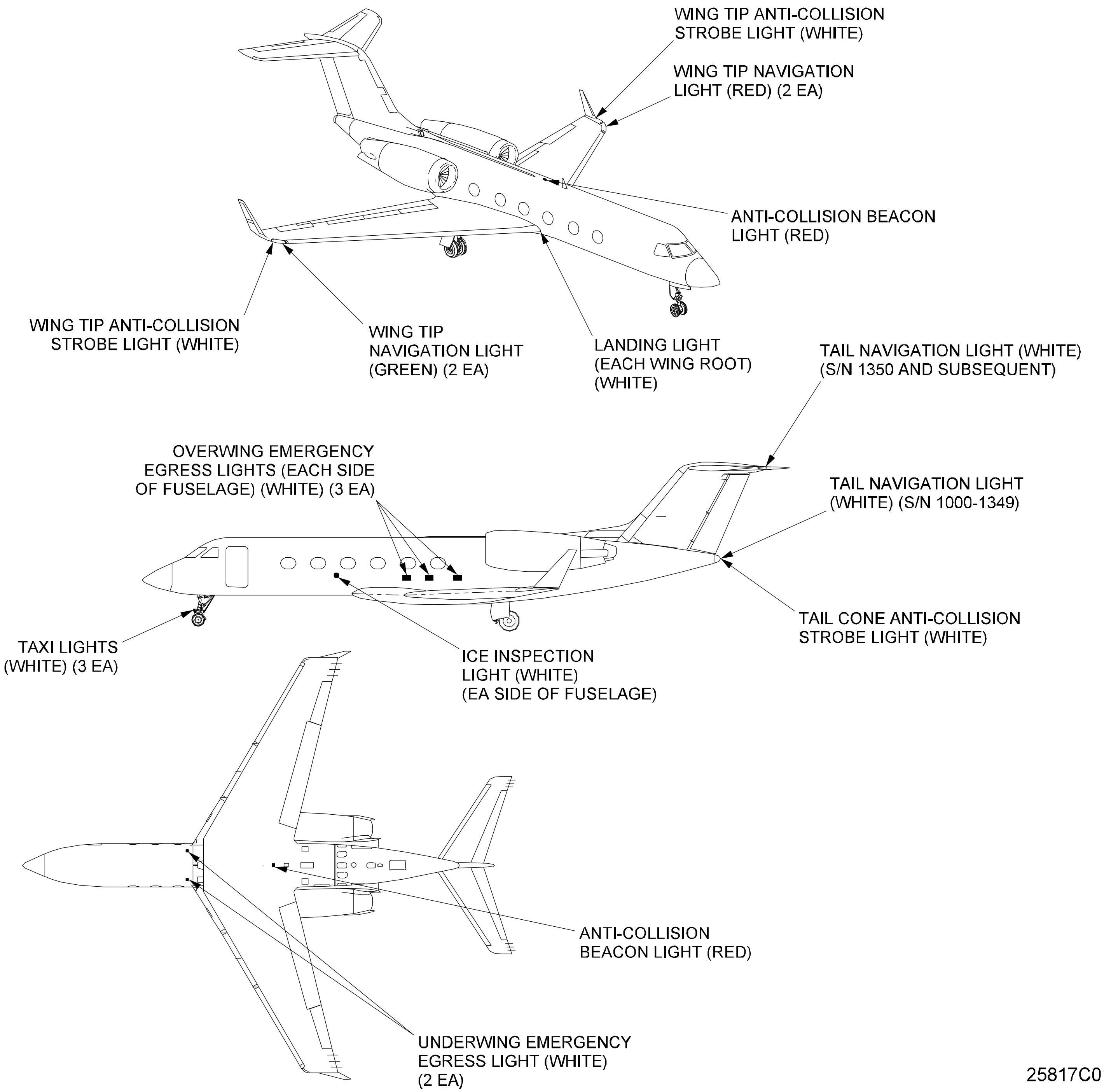
B. Advisory (Blue) CAS Messages:

CAS Message	Cause or Meaning
SERVICE DOORS	One or both of the following service doors is open: • Tail compartment access door • Refuel compartment access door (SN 1156 and subs)

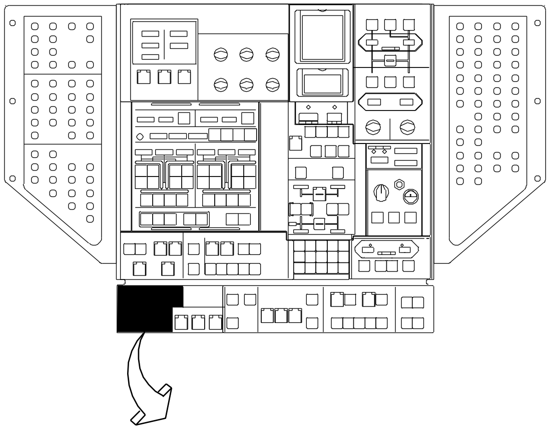
4. Limitations:

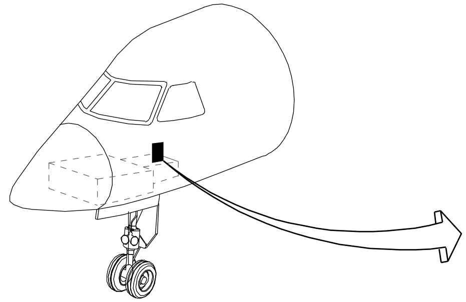
A. Flight Manual Limitations:

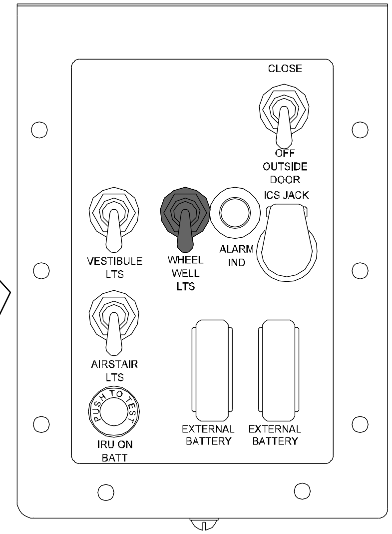
Ground operation of landing lights is limited to five (5) minutes.



Exterior Lights Location
Figure 7

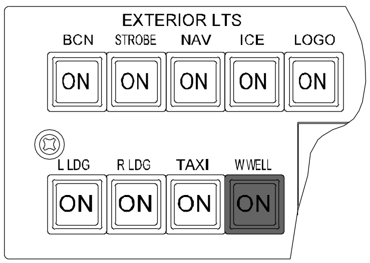




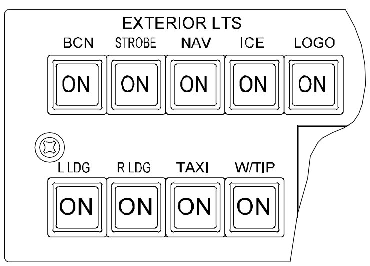


AIRPLANES 1455 AND SUBSEQUENT

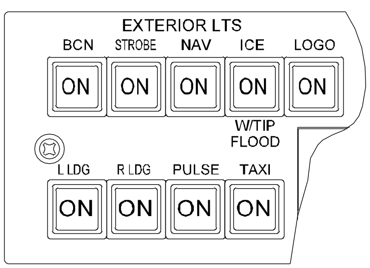
NOSE WHEEL SWITCH PANEL



AIRPLANES 1000 - 1279 NOT HAVING ASC 1 AND AIRPLANES 1280 - 1454

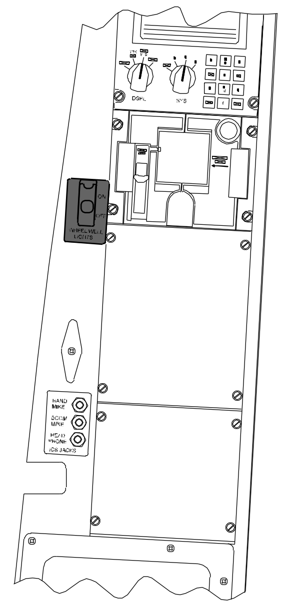


AIRPLANES 1000 - 1279 HAVING ASC 1



AIRPLANES 1455 AND SUBSEQUENT

NOTES: GRAY SHADED AREAS INDICATE LOCATION OF WHEEL WELL LTS SWITCH BASED ON EFFECTIVITY.



AIRPLANES 1000 - 1279 HAVING ASC 1

PILOT SIDE CONSOLE

29604C01

Cockpit EXTERIOR LTS Panel / Nose Wheel Switch Panel
Figure 8

2A-33-60: Emergency Lighting System

1. **General Description:**

The external emergency lighting system facilitates safe evacuation of the airplane during an emergency in darkness or low visibility. The system is composed of the following elements:

- Overwing Exit Lights
- Underwing Exit Lights
- Main Entrance Door Emergency Lighting
- Emergency Lighting Battery Packs
- Emergency Lighting Control Switches

2. **Description of Subsystems, Units and Components:**

A. **Overwing Exit Lights:**

There are three exit lights on each side of the airplane, installed in the fuselage below the emergency exit windows. A light is located at the forward, mid and aft wing root area to illuminate the top of the wing surface as an aid in airplane evacuation.

B. **Underwing Exit Lights:**

An exit light positioned forward of the leading edge of the wing on each side of the airplane illuminates the area beneath the wing. The light facilitates judging the distance from the wing surface to the area below for airplane occupants evacuating using the emergency exit windows.

C. **Main Entrance Door Emergency Lighting:**

Automatic activation of the main entrance door emergency lighting is available on SN 1467 and subsequent through the use of a second set of emergency batteries. If the main entrance door is open and the EMERG POWER ARM switchlight (cockpit overhead panel) has been previously been selected to ARM, activation of the emergency batteries will allow emergency batteries No. 3 and 4 to automatically power the interior emergency lighting and main entrance door emergency lighting.

D. **Emergency Lighting Power (Airplanes 1000 Through 1466):**

Two emergency batteries, one forward and one aft, make up the emergency power system. The forward emergency battery is located in the right radio rack. The aft emergency battery is located in the tail compartment at fuselage station 680. The batteries are continually charged by the Essential DC bus. Each of the batteries powers three emergency buses: the forward battery powers 1A, 1B and 1C, while the aft battery powers 2A, 2B and 2C. Basic electrical equipment and instruments necessary for flight are on the emergency buses: communication and navigation radios, clocks, landing gear and flap position indicators, etc. (For a complete discussion of the emergency power system, see the electrical section of this manual.) All of the external emergency exit lights are wired to both batteries to ensure that the exit lights will be powered in the event of a single battery failure or damage to either the front or rear of the airplane.

E. Emergency Lighting Power (Airplane 1467 and Subsequent):

Four emergency battery units, two forward and two aft, make up the emergency power system. The forward emergency batteries (No. 1 and 3) are located in the right radio rack. The aft emergency batteries (No. 2 and 4) are located in the tail compartment at fuselage station 680. The batteries are continually charged by the Essential DC bus.

In the event of Essential DC power loss, batteries 1 and 2 will automatically power the Emergency buses, wing leading edge evacuation lighting and external overwing evacuation lighting. Batteries 3 and 4 will automatically power the interior emergency lighting and main entrance door emergency lighting (if the door is open).

Each set of batteries powers three emergency buses: the forward batteries power 1A, 1B and 1C, while the aft batteries power 2A, 2B and 2C. Basic electrical equipment and instruments necessary for flight are on the emergency buses: communication and navigation radios, clocks, landing gear and flap position indicators, etc. (For a complete discussion of the emergency power system, see the electrical section of this manual.) All of the external emergency exit lights are wired to both sets of batteries to ensure that the exit lights will be powered in the event of a single battery failure or damage to either the front or rear of the airplane.

F. Emergency Power Control Switches:

(See Figure 9 and Figure 10.)

The forward and aft emergency batteries are controlled by three guarded momentary action switches located in the EMERGENCY POWER section of the cockpit overhead panel. The switches are labeled ON, ARM, and OFF. Selections made with the switches result in the following operations:

(1) Switch ON:

NOTE:

Asterisk (*****) items below apply to SN 1467 and subsequent.

- Forward and aft emergency lighting batteries are activated
- Overwing and underwing exit lights illuminate
- Legend “1 ON / 2 ON / 3 ON***** / 4 ON*****” is illuminated in switch capsule
- A blue E BATT 1-2-3*****-4***** DISCH message is displayed on the Crew Alerting System (CAS)

(2) Switch ARMED:

When ARM is selected, all legends in the emergency power control switches are extinguished and the system monitors the Essential DC bus. If Essential DC bus power falls below 20V or is lost, the emergency power system is activated.

(3) Switch OFF:

When OFF is selected and Essential DC bus voltage is above 20V, the emergency power system is disabled.

NOTE:

The emergency power system can only be selected OFF if the Essential DC bus is above 20V. To prevent emergency battery depletion, it is important to select the system OFF prior to selecting the left and right main airplane batteries OFF.

(4) Impact Switch:

Two impact switches, one adjacent to each battery, will activate the emergency batteries if a force of 2.5 G's is detected. Battery activation by the impact switches is independent of emergency power switch position (system may be selected to OFF or ARM).

3. Controls and Indications:

A. Circuit Breakers:

There are no circuit breakers for the external emergency exit lights. The lights are protected by fuses on each battery.

B. Advisory (Blue) CAS Messages:

CAS Message	Cause or Meaning
E BATT 1-2-3*****-4***** FAIL	Indicated Emergency Battery has failed.
E BATT 1-2-3*****-4*****DISCH	Indicated Emergency Battery is discharging.

NOTE(S):

Asterisk ***** items in above table apply to SN 1467 and subs.

NOTE:

If the emergency power system is activated and an emergency power system fault message (E BATT FAIL) is displayed on CAS, the battery pack may have an overcurrent trip condition due to the charging cycle. Attempt to restore normal operation by selecting the switch ON (if switch is in ARMED position). If the overcurrent fault clears, the battery will reset and the fault message on CAS will extinguish.

4. Limitations:

There are no limitations established at the time of this revision.

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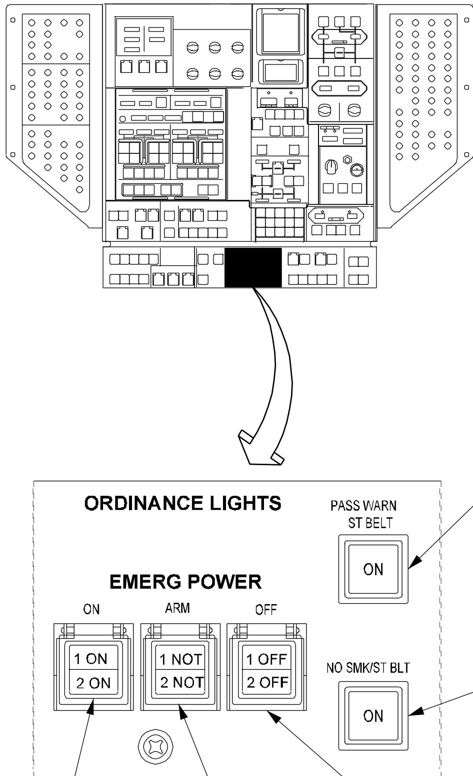

ST BELT SWITCHLIGHT:

- ON: Turns on FASTEN SEAT BELT (and if installed, RETURN TO SEAT) signs in the cabin. ON legend illuminates when on.
- OFF: Turns off signs and ON legend in switchlight.

NO SMK / ST BLT SWITCHLIGHT:

- ON: Turns NO SMOKING and FASTEN SEAT BELT (and if installed RETURN TO SEAT) signs in cabin. ON legend illuminates when on.
- OFF: Turns off signs and ON legend in switchlight.

OFF

- Depressing switch selects emergency batteries OFF. essential DC must be powered for switch to function.
- OFF selection is verified by switchlight 1 OFF / 2 OFF amber legends illuminated.

ARM

- Depressing switch arms emergency batteries.
- All switchlight legends are extinguished.
- If essential DC bus is lost, emergency batteries will automatically power emergency buses.

ON

- Depressing switch selects emergency batteries ON.
- Amber 1 ON / 2 ON switch legends illuminate.
- Amber 1 NOT / 2 NOT switch legends extinguish.

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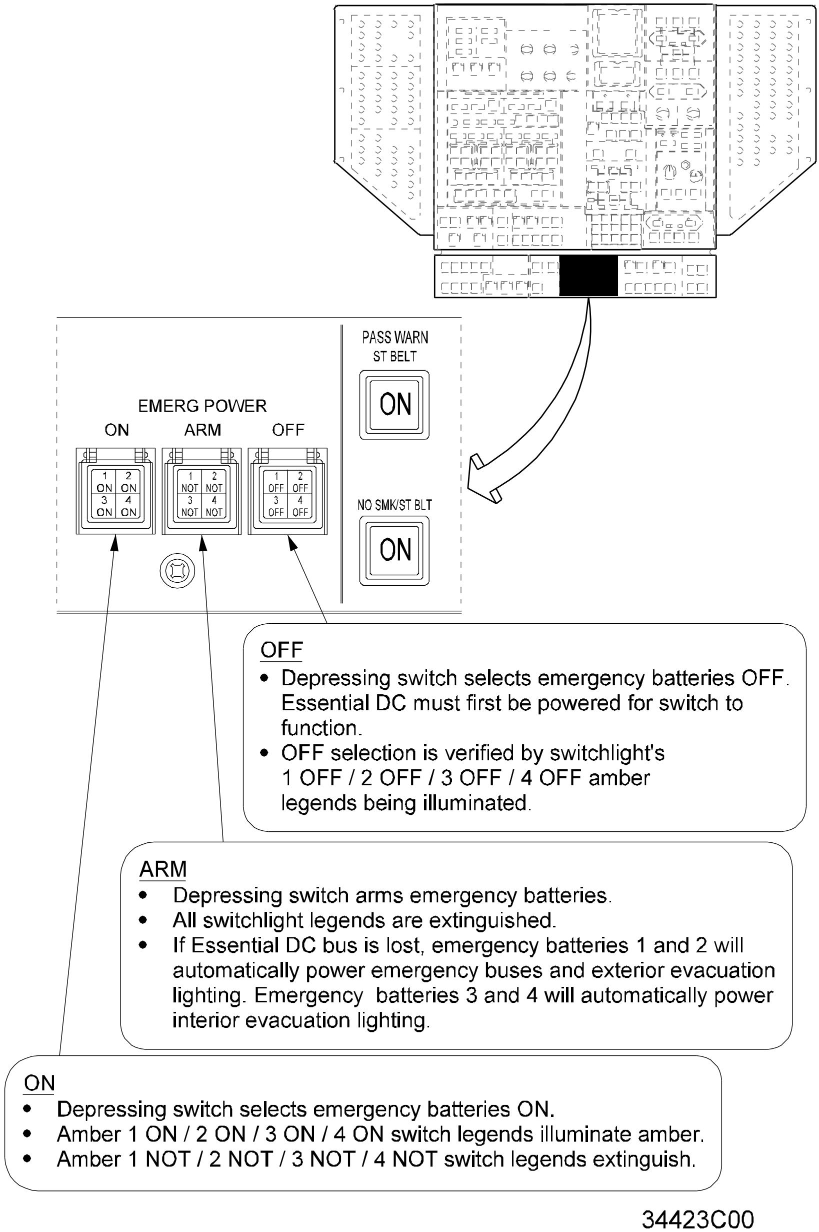
ORDINANCE LIGHTS
Panel (SN 1000 - 1466)
Figure 9



ORDINANCE LIGHTS Panel (SN 1467 & Subs)
Figure 10

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2A-34-10: General

The navigation system provides the flight crew with indications of position in three dimensions and vector information for management of the flight environment, supplemented with aural and visual alerts to prevent Controlled Flight into Terrain (CFIT) and collision with other airborne traffic.

Flight environment data, aircraft attitude and direction are integrated with the SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System (DAFCS) and the Integrated Automatic Tuning System (Collins RTU-4200 Series Radio Tuning Unit (RTU)). The DAFCS is described in Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV. The Integrated Automatic Tuning System (Collins RTU-4200 Series Radio Tuning Unit (RTU)) is described in Section 2A-23-40, Integrated Automatic Tuning System and Collins' RTU-4200 Series Pilot's Guide. This section details the sensor systems used to determine airspeed and altitude, the integration functions of the Digital Air Data Computers that supply sensor data to the DAFCS and the onboard systems that provide alerts and warnings to prevent hazardous flight conditions.

The Enhanced Ground Proximity Warning System (EGPWS) is a terrain awareness and warning system incorporating alerting and display functions. The system uses aircraft geographic position, altitude, climb and descent rate, and a terrain database to determine potential conflicts between the aircraft flight path and terrain, and provide aural alerts and visual depictions (in conjunction with the DAFCS) of hazardous terrain clearance. (Visual cues generated by the TERRAIN DISPLAY feature of the EGPWS are shown and described in Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV.

The Traffic Collision Avoidance System / Aircraft Collision Avoidance System (TCAS / ACAS) uses transponder signal information to detect and plot the tracks of other airborne traffic and formulate flight guidance for maneuvers to avoid potential collisions.

The navigation system is divided into the following subsystems:

- 2A-34-20: Flight Environment Data System
- 2A-34-30: Attitude and Direction System
- 2A-34-40: Radio Altimeter System
- 2A-34-50: Enhanced Ground Proximity Warning System (EGPWS)
- 2A-34-60: Traffic / Aircraft Alert and Collision Avoidance System (TCAS / ACAS)

2A-34-20: Flight Environment Data System

1. **General Description:**

   The flight environment data systems incorporate navigational sensors that sample environmental conditions to determine airplane attitude, speed and direction.

   The systems include the following components:

   - Pitot-Static System
   - Standby Airspeed Indicator
   - Standby Altimeter
   - Static Air Temperature / Total Air Temperature (SAT / TAT) System
   - Digital Air Data Computer System

2. Description of Subsystems, Units and Components:

A. Pitot-Static System:

The pitot-static system samples the atmospheric environment to provide indications of airplane airspeed and altitude. (See the illustration in Figure 3 for a depiction of the system.) Three pitot tubes are mounted on the exterior of the airplane and aligned forward to sense dynamic air pressure generated proportional to airplane velocity. As the airplane moves through the atmosphere, more air molecules are encountered than if the airplane were stationary. The increase is proportional to speed, and is sensed as pressure. The hollow pitot tube heads, positioned into the airplane slipstream, direct the increased pressure into the pitot system for measurement by the airspeed indicators. The metal heads of the tubes are heated electrically to prevent blockage by frozen precipitation. Internally, the pitot tubes are plumbed with nylon tubing. The increased dynamic air pressure in pitot tubes is compared with static air pressure sensed by other tubes attached to static ports mounted flush on the airplane fuselage. (The static ports are essentially vents, since they are not aligned into the slipstream.) By measuring the difference between static air pressure and the increased dynamic air pressure, an accurate determination of airspeed can be made.

There are four static air pressure tubes and ports: three are paired with and plumbed to the associated pitot tube (pilot to DADC #1, copilot to DADC #2, and standby to the standby airspeed and altimeter). The fourth static tube and port is connected to the cabin pressure indicator and controller. Cabin pressure is measured and controlled within structural limits by comparing the air pressure within the airplane to the atmospheric pressure outside the cabin (for the cabin differential pressure). Unlike the pitot tubes, the static tubes have dual connections, with each tube plumbed to a static port on either side of the airplane. Dual ports are necessary in order to obtain an accurate static pressure sampling. If a static tube were not connected to a port on either side of the airplane, any untrimmed flight condition around the vertical axis, such as a skid or slip, would induce an increase in the sampled static pressure due to the port being slightly aligned into the airplane slipstream.

NOTE:

On some airplanes the customer has chosen to install an optional secondary cabin pressure indicator. For those airplanes, a switch on the aft portion of the copilot side console, labelled NON-ESSENTIAL PITOT/STATIC CONTROL, is used to isolate the secondary cabin pressure indicator from the primary indicator. When the secondary indicator is isolated with the switch, an independent verification of cabin altitude is available.

B. Standby Airspeed Indicator:

The standby airspeed indicator provides an emergency source of airspeed information in the event of failure or malfunction of the primary flight displays. The standby indicator displays current airspeed, maximum airspeed for altitude (V_{MO}) and Mach number. On airplanes Serial Number

(SN) 1000 to 1059 (except SN 1001 and 1034) not having Aircraft Service Change (ASC) 66, airspeed is indicated with a white pointer that rotates clockwise around the face of the instrument proportional to airplane speed and V_{MO} by a “barber pole” striped pointer. Pitot/static connections provide the input for airspeed information. Although the indicator has a drum-type readout of Mach number in the upper center of the instrument, the ARINC 429 connections necessary to derive Mach speed information from the DADC are not installed. The OFF flag covering Mach readout will be displayed at all times on the instrument, indicating that this feature is not available.

On airplanes SN 1060 and subsequent, and prior airplanes having ASC 66, the standby airspeed indicator is not powered (except for lighting circuits) and obtains speed information directly from the pitot/static system. On these airplanes, airspeed is indicated by a rotating white pointer, Mach number is read on an additional scale positioned on the outside perimeter of the airspeed scale and V_{MO} / M_{MO} is denoted with a red band on both the airspeed and Mach number scales. A knob on the lower face of the instrument is provided to set a movable airspeed reference bug (V_{REF}). See the illustration in Figure 1.

On all airplanes, the indicator is illuminated internally using 5V DC power from normal cockpit lighting circuits.

C. Standby Altimeter:

The standby altimeter provides an indication of airplane pressure altitude in event of primary flight display failure or malfunction. Altitude is indicated in twenty (20) foot increments within a range of one thousand (1,000) feet by a pointer that moves in front of the circular scale on the face of the instrument, making one revolution for every 1,000 feet. A drum-type counter in the center of the instrument provides graphic indications of altitude in hundred (100) and 1,000 foot increments. The local barometric pressure in inches of mercury (Hg) or millibars (Mb) is set with the knob on the lower corner of the instrument and displayed in windows in the instrument face. See the illustration in Figure 1.

Like the standby airspeed indicator, the ARINC 429 connections necessary to derive altitude information from the DADCs are not installed. A yellow flag, labeled PNUE, is displayed at all times on the instrument, indicating that this feature is not available.

The standby altimeter is powered by 28V DC from the Essential DC bus, and uses only pitot/static input for altitude determination. It is internally illuminated and has an electrically driven vibrator to smooth pointer movement. If electric power is lost, a VIB flag is displayed on the instrument and while indications will be valid, pointer movement may be intermittent, requiring occasional manual tapping on the face of the instrument during climb and descent.

D. Static Air Temperature / Total Air Temperature (SAT / TAT) System:

A heated Total Air Temperature probe is installed on the lower right forward fuselage to provide actual air temperature to the DADCs. See Figure 2 for a depiction of the probe. On the ground with electrical power and bleed air available, the probe is aspirated by bleed air through a dedicated line controlled by a solenoid operated valve that routes air into the probe housing, venting the air through holes in the probe to induce outside air

flow over the temperature sensing element. Air flow through the probe gives a more accurate reading of ambient conditions and avoids temperature increases associated with the heating effects of sunlight. Bleed air aspiration is controlled by the nutcracker (squat) switch system and requires 28V DC from the Right Main DC bus. The TAT probe is electrically heated when the airplane is airborne to prevent icing.

E. Digital Air Data Computer System (DADC):

Dual Honeywell AZ-810 Digital Air Data Computers (DADCs) are installed in the forward avionics racks. The DADCs are connected to the pitot/static system, with the pilot side pitot/static probe and ports connected to DADC #1 and the copilot side pitot/static sensors connected to DADC #2. Both DADCs are connected to the TAT probe for temperature data. The DADCs use the pneumatic and temperature data to compute correct airspeed, altitude, vertical speed, static source error correction (SSEC) and to send a signal to the audible tone generator initiating the overspeed “cricket” warning when airspeed reaches V_{MO}/M_{MO}. Other inputs used by the DADCs are Angle of Attack (AOA), flap handle position, barometric setting and pre-selected altitude. The DADCs formulate digital signals for elements of the airplane avionics system. The DADCs communicate data to the following:

- Electronic Display System
- Transponders
- Flight Recorder
- Flight Guidance Computer
- Fault Warning Computer
- Inertial Reference System
- Flight Management System
- Cabin Pressurization System
- Stall Barrier System
- Engine Pressure Ratio (EPR) Sensor
- Traffic / Aircraft Collision Avoidance System (TCAS / ACAS)

In order to maintain a continual crosscheck of system accuracy, it is recommended that the pilot select DADC #1 as a data source, and the copilot select DADC #2. Data source selection is made on the SENSOR page of the Display Controller. In this configuration, each pilot has an independent source for altitude, airspeed, AOA, vertical speed, SAT and TAS on their respective PFDs and navigation displays. If both pilots select the same DADC as data source, an amber annunciation is displayed on both PFDs. (If the pilot PFD is selected to DADC 2, both PFDs would display DADC 2 in amber in the upper right of the PFDs.)

Data source selection for the guidance panel altitude pre-select is a function of the PFD command button (PFD-CMD). When the button indicates L, the data source is DADC #1, when R is displayed, data comes from DADC #2.

Most subsystems with dual installations normally source DADC data according to a standard coupling arrangement: the left (L), #1 or pilot side sourced to the #1 DADC, and the right (R), #2 or copilot side connected to

the #2 DADC. For some, but not all installations, the data source is selectable to provide redundancy in case of DADC failure. Transponders (ATC) #1 and #2 are normally sourced to their respective DADC, but may be selected to the alternate DADC. Angle of Attack (AOA) indicators are referenced to the DADC selected to the respective PFD. EPR sensors are normally paired with engine #1 to DADC #1, engine #2 to DADC #2, but failure of a DADC will prompt an automatic switch to the remaining DADC for EPR computation. The DADC source for cabin pressure control is selectable on the cockpit overhead panel.

DADC Failure Modes (Flagged, Unflagged)

(1) A “flagged” DADC failure is one where the failure is readily apparent, because of the blue DADC 1 (or 2) FAIL advisory CAS message, and red “X’s” through all four air data scales (airspeed, altitude, AOA, and vertical speed) of the PFD using the failed DADC as its air data source, as selected on the display controllers. Other confirmation of failure is as follows:

- Transponder indications
- AOA indexer failure
- Automatic cabin pressurization control problems and faulty guidance panel indications (if operating on the failed DADC)
- EICAS message indicating that EPR is receiving pressure information from the opposite DADC (EPR 1 - DADC 2, or EPR 2 -DADC 1)

The solution is to select the opposite (good) DADC as the air data source, on display controllers, guidance panel, cabin pressure control panel, etc.

(2) An "unflagged" DADC failure will produce a blue DADC MISCOMPARE advisory CAS message, and the failure may not be readily apparent. The autopilot and yaw damper will disconnect and will not re-engage until the faulty DADC has been identified and isolated by pulling its circuit breaker. Pitch trim will remain operative. The flight crew may be able to identify the faulty DADC by looking for an amber IAS and / or ALT comparator warning annunciation to left of the horizon in each PFD. The IAS indication means a split of 20 or more knots exists between air data systems; the ALT indication means a split of 200 feet or more exists between systems. To determine which system is correct requires reference to an independent data source, in this case standby flight instrumentation. Since the standby flight instruments show large errors because they are uncorrected for static source error, it is recommended that standby altimeter be set so as to read the same as the cruising flight level. Once stable cruise speed is attained, the settable airspeed bug should be aligned with the standby airspeed pointer. Thus, reference can be made to the standby altimeter and airspeed indications, as "voters" in helping to determine which DADC outputs are more nearly correct. Then, check the other DADC outputs, the pressurization system, AOA indexers and transponders, for indications of faulty operation. If observation leads to a determination of which DADC is faulty, select the "good" DADC to both PFDs, guidance panel, transponders, and the cabin pressurization system. Then isolate the faulty DADC by pulling its circuit breaker, and after at least a one minute wait, re-engage the autopilot.

3. Controls and Indications:

(See Figure 1.)

NOTE:

A description of the SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System (DAFCS) can be found in Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV. A description of the Integrated Automatic Tuning System (Collins RTU-4200 Series Radio Tuning Unit (RTU)) can be found in Section 2A-23-40, Integrated Automatic Tuning System and Collins' RTU-4200 Series Pilot's Guide.

A. Circuit Breakers (CBs):

Circuit Breaker Name	CB Panel	Location	Power Source
TOTAL TEMP PROBE HTR	CP	L-10	MAIN 115V AC φB
TOTAL TEMP VALVE	CP	M-10	R MAIN 28V DC
L PITOT HT PWR	CP	L-11	ESS 115V AC φA
R PITOT HT PWR	CP	M-11	MAIN 115V AC φA
L PITOT HT CONT	CP	L-12	ESS 28V DC
R PITOT HT CONT	CP	M-12	MAIN 28V DC

Circuit Breaker Name	CB Panel	Location	Power Source
STBY PITOT HT CONT	CP	L-13	ESS 28V DC
STBY PITOT HT PWR	CP	M-13	ESS 115V AC ϕA
AOA PRB HTR #1	CP	L-14	ESS 28V DC
AOA PRB HTR #2	CP	M-14	MAIN 28V DC
DADC #1	CP	F-3	ESS 28V DC
DADC #2	CP	G-3	MAIN 28V DC
STBY AIR SPD IND	CP	H-4 **(1)**	EMER 28V DC
DUAL MODE ALTM VIB	CP	H-3 **(1)**	EMER 28V DC
STBY ALTM VIB	CP	H-3 **(2)**	EMER 28V DC
DUAL MODE ALTM	CP	H-4 **(3)**	ESS 28V DC
STBY ALTM VIB	CP	H-4 **(4)**	EMER 28V DC

NOTE(S):

(1) SN 1000 -1059 (except 1001 and 1034) not having ASC 66

(2) SN 1000, 1002-1122 (except 1034) having ASC 66, SN 1001, and 1060 - 1167

(3) SN 1000, 1002 - 1059 (except 1034) not having ASC 66

(4) SN 1034, 1168 and subs

B. Caution (Amber) CAS Messages:

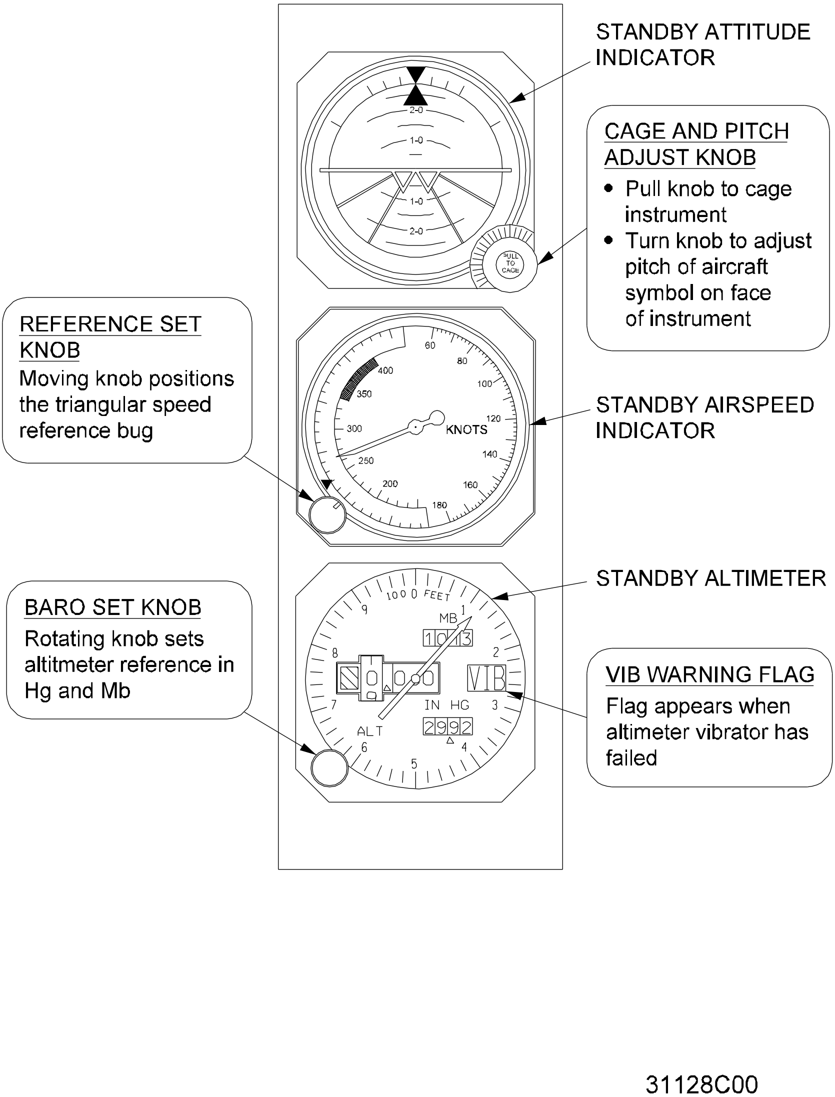
CAS Message	Cause or Meaning
AOA HEAT 1-2 FAIL	Angle of attack probe heater failed
L-R PITOT HT FAIL	Indicated pitot tube heater elements not energized
SSEC DISABLED	Static source error correction to DADC has been disabled
STBY PITOT HT FAIL	Standby pitot heater elements not energized
TAT PROBE HT FAIL	TAT probe heater has failed

C. Advisory (Blue) CAS Messages:

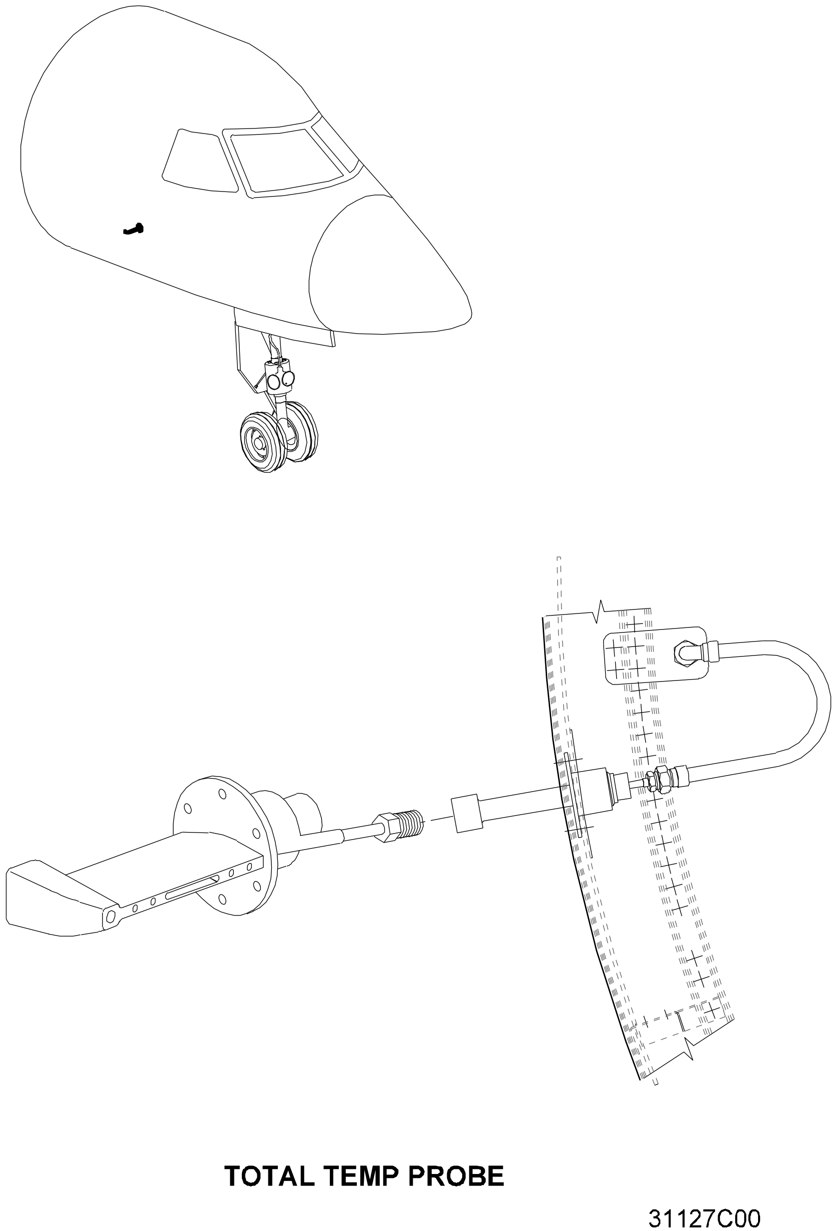
CAS Message	Cause or Meaning
DADC 1-2 FAIL	A DADC has failed
DADC MISCOMPARE	The priority FGC has detected an unflagged miscompare between DADC 1 and DADC 2
EPR 1 - DADC 2 EPR 2 - DADC 1	DADC malfunction has caused remaining DADC to supply information to both EPR systems

4. Limitations:

There are no limitations to this system at the time of this revision.



Standby Flight Instruments
Figure 1



SAT / TAT Probe
Figure 2

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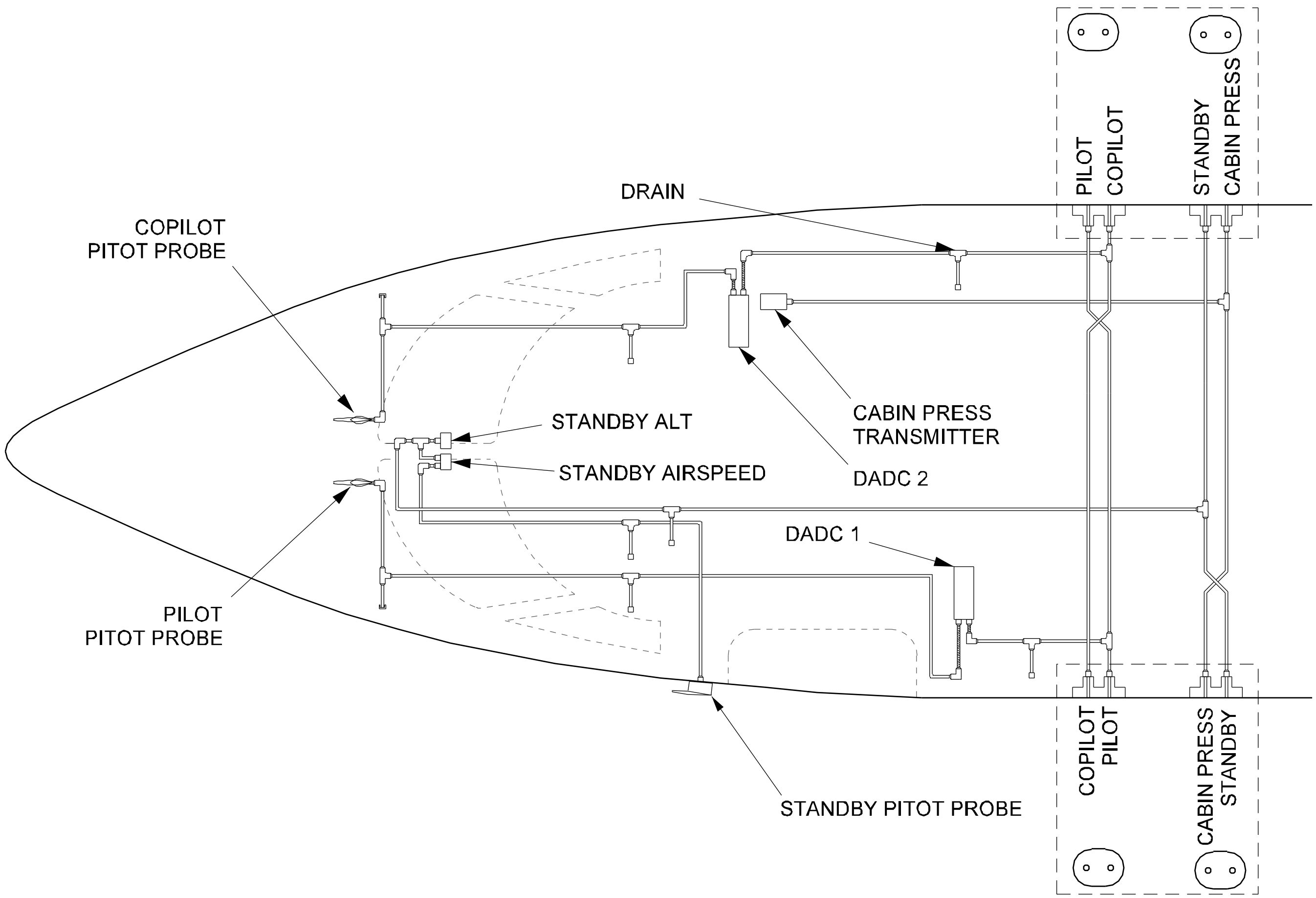

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Pitot / Static System
Schematic
Figure 3

2A-34-30: Attitude and Direction System

1. General Description

The standby attitude and direction systems provide references for steering the airplane in the desired direction using basic instruments that remain powered during instances of failure / malfunction of the Primary Flight Displays (PFDs) and / or the Flight Management System (FMS). The following subsystems, units and components are included:

- Standby Attitude Indicator
- Standby Directional System

2. Description of Subsystems, Units and Components:

A. Standby Attitude Indicator:

The standby attitude indicator (shown in Figure 1) is a gyroscopically driven artificial horizon that provides a backup indication of airplane pitch and roll in the event of a failure or malfunction of PFDs, FMS or IRS systems. The indicator contains a sphere divided into hemispheres painted blue and brown (or black) representing sky and earth. Markings on the hemispheres denote pitch attitude in five degree (5°) increments from level up or down to eighty degrees (80°). When the airplane is initially powered, the indicator will power-up and self-erect once the internal gyroscope reaches operating speed. To immediately erect the indicator, use the cage knob on the face of the instrument, orienting the sphere upright and leveling the horizon (where the sky and earth representations meet). The airplane symbol on the face of the instrument is adjusted to match the airplane pitch attitude by rotating the cage knob.

At the top of the instrument is a triangular pointer that indicates airplane bank angle against the semicircular scale surrounding the artificial horizon. The scale is marked in ten degree (10°) increments up to thirty degrees (30°) of bank, and additional marks at forty-five degrees (45°) (for Serial Number (SN) 1330 and subsequent only), sixty degrees (60°) and ninety degrees (90°).

The standby attitude indicator is powered by the Emergency DC bus. If power to the instrument is interrupted, a red warning flag is displayed on the face of the instrument (the flag is also displayed when the indicator is caged). If Emergency DC bus power to the indicator is lost (battery depleted - warning flag displayed) the rate of spin of the gyro may be sufficient to supply attitude information for several minutes before becoming unreliable.

On airplanes SN 1330 and subsequent, the standby attitude indicator has pointers for glideslope and localizer signals, providing guidance for an ILS approach. ILS information is supplied by VHF NAV #1 over an ARINC 429 data bus. A knob on the lower left corner of the attitude indicator is used to select signal input to the glideslope and localizer pointers. The knob has three (3) positions: OFF, ILS and B/C (for a back course approach). Both the glideslope and localizer pointers have a warning flag to indicate power off or lack of signal reception.

B. Digital Bearing and Distance Indicator (DBDI):

The dual DBDI (pilot and copilot) indicators provide a standby display of heading, bearing, and distance to a selected navigation source

independent of the Primary Flight Display (PFD). The indicators have dual power supplies, each powered by the Essential or Emergency DC bus, with only basic heading information available when operating on Emergency DC bus power. The DBDI is illustrated in Figure 4.

During normal operation with all electrical sources available, the DBDIs are powered through the Essential DC bus and display the following information:

- Airplane heading beneath the lubber line at the top of the compass card, sourced from the IRS system in use (pilot on #1, copilot on #2).
- Pointer #1 (single shaft arrow) pointing to the currently tuned navigational radio transmitter (VOR / VORTAC or ADF) selected with the pointer switch on the lower left of the instrument. The pointer will be positioned to the three o'clock position and a flag displayed if no navigation radio is tuned or the signal is out of range for the transmitter type selected.
- Pointer #2 (double shaft arrow) pointing to the currently tuned navigational radio transmitter (VOR / VORTAC or ADF) selected with the pointer switch on the lower right of the instrument. The pointer will be positioned to the three o'clock and a flag displayed if no nave radio is tuned or the signal is out of range for the transmitter type selected.
- Digital DME distance readout is displayed in the window in the top of the instrument for both (1 and 2) currently tuned VORTACs (only dashes are displayed if no VORTAC is currently tuned). Distance is indicated in one-tenth mile (0.1) increments up to 99.9 nautical miles and one mile increments from 100 to 999 nautical miles.
- Digital display of the frequency of the currently tuned VORTAC if the NAV selector is placed in HOLD mode, with the frequency preceded by an H to indicate that HOLD has been selected. In normal NAV mode, only dashes are displayed in the frequency space.

A failure of the heading information source (IRS #1 and IRS #2) prompts the DBDI to automatically switch to an alternate heading source if one is installed (typically AHRS or IRS #3). The use of an alternate heading source is annunciated on the face of the instrument by the illumination of a green AHDG light in the upper right corner of the instrument.

A more severe failure that involves the loss of primary and alternate heading reference sources would automatically switch the DBDI to standby mode, annunciated by the illumination of the amber STBY light in the upper right of the indicator. In standby mode, the heading reference source is the dual flux detector installation in the airplane wing tips. The flux detectors act as magnetic compasses, sensing the horizontal component of the magnetic field surrounding the earth. The magnetic direction sensed by the flux detectors is corrected for errors (induced by the metallic structure of the airplane) by a magnetic compensator before it is sourced to the DBDI for use as the heading reference in standby mode.

Should there be a failure of the standby / flux detector system, a red STBY light illuminates in the upper right corner of the instrument. The red STBY will illuminate with a standby system failure even if the DBDI is operating in the normal (IRS) mode.

The DBDI indicators annunciate system degradation and/or subcomponent failure with the following displays:

- OFF flag in the upper left of the instrument indicates loss of instrument power
- HDG flag at the lubber line at the top of the instrument indicates an invalid compass signal
- Single shaft arrow flag on the left of the instrument indicates invalid signal to the #1 pointer
- Double shaft arrow flag on the right of the instrument indicates invalid signal to the #2 pointer

All of the mode annunciations and system / subcomponent failure annunciations may be tested on the ground during preflight with the ST/R switch (momentary) on the upper right of the heading display. This test feature is wired through the nutcracker (squat) switch system to operate in the ST or self-test mode on the ground and the R or reset mode in the air. Pressing the ST/R button in the air will reset the heading indicator, recapturing a primary or alternate heading source if one is available. Pressing the ST/R button during a ground preflight will illuminate the mode annunciations, display warning flags and indicate a DBDI malfunction by a five (5) second flashing red STBY light followed by continuous illumination of the light.

On airplanes having Aircraft Service Change (ASC) 217, DBDI MANUAL STBY switches are installed on the outboard side of the pilot and copilot forward instrument panels, one for each DBDI. The installation is shown in Figure 5. (A switch that enables manual selection of the standby mode is located on the face of the DBDI electronic module in the radio rack on airplanes not having ASC 217). This switch allows the crew to manually select the standby (STBY) heading mode in flight to crosscheck inertial (IRS) heading information, compensate for faulty IRS information, or to select a heading source during realignment of the IRSs.

3. Controls and Indications:

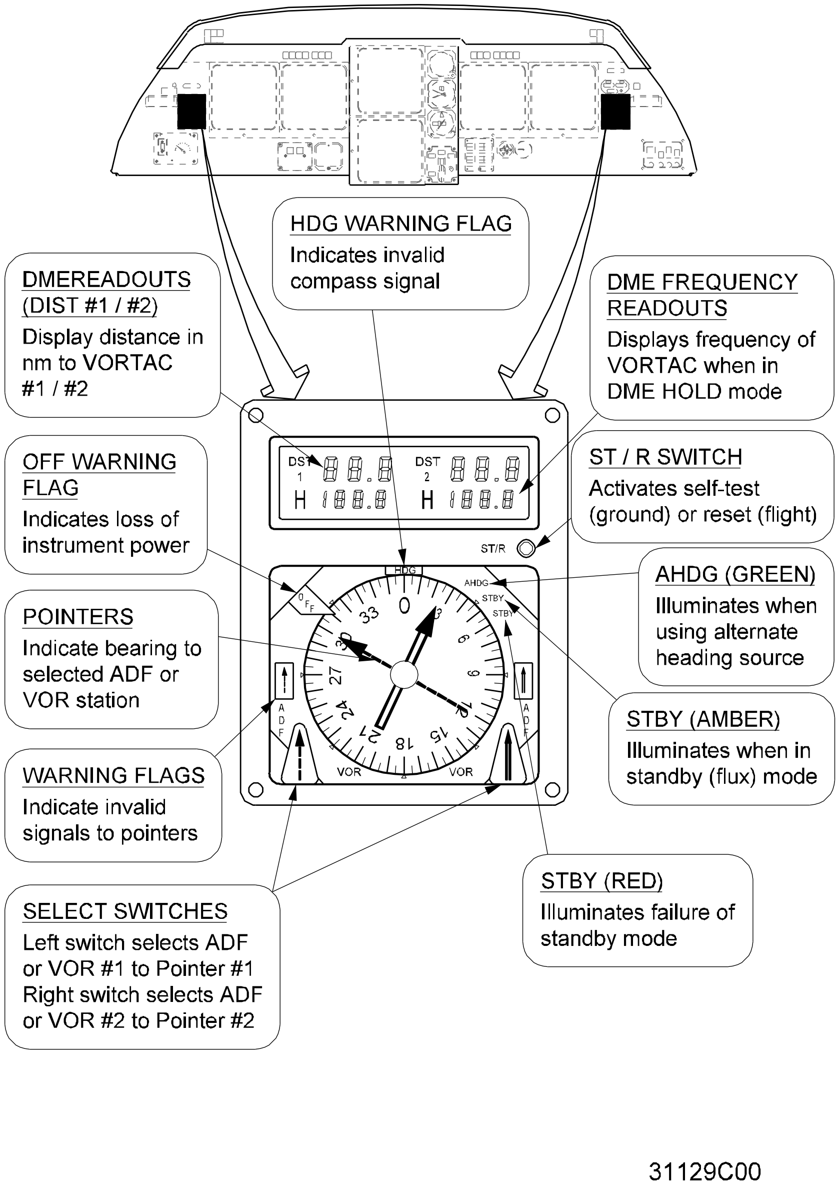
(See Figure 4 and Figure 5.)

A. Circuit Breakers (CBs):

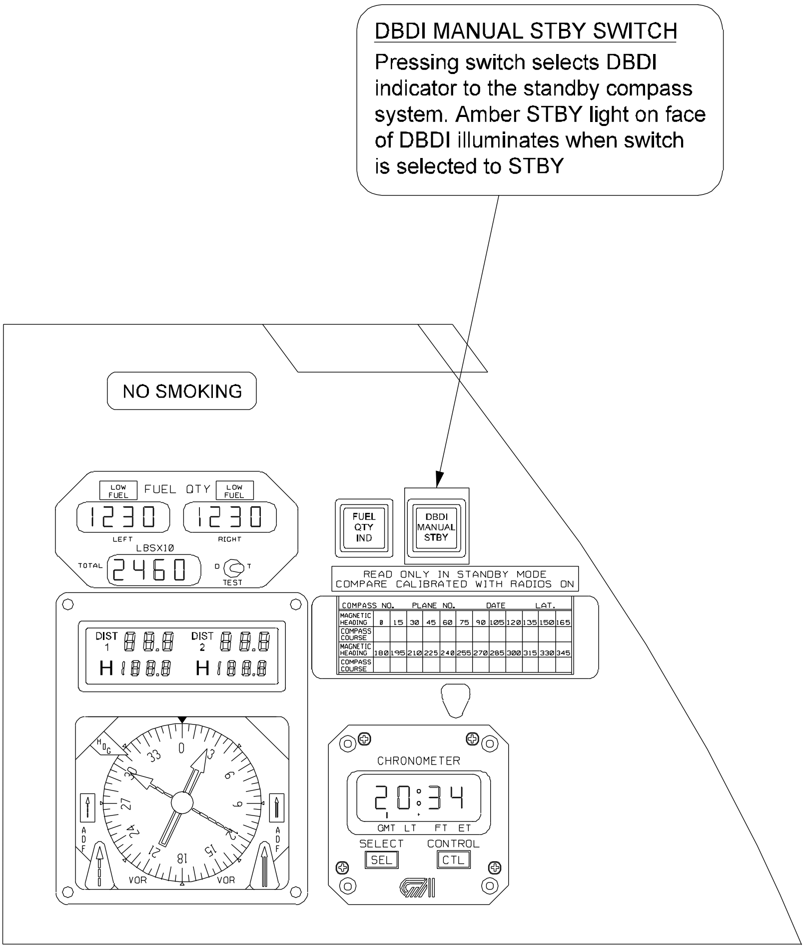
Circuit Breaker Name	CB Panel	Location	Power Source
STBY HORZN	CP	E-4	EMER 28V DC
DBDI #1 (SN 1212 & subs) (DDRMI #1 on SN 1000-1211)	CP	F-4	ESS 28V DC
DBDI #2 (SN 1212 & subs) (DDRMI #2 on SN 1000-1211)	CP	G-4	ESS 28V DC
E DBDI #1 (SN 1212 & subs) (E DDRMI #1 on SN 1000-1211)	CP	H-5	EMERG 28V DC 1C
E DBDI #2 (SN 1212 & subs) (E DDRMI #2 on SN 1000-1211)	CP	I-5	EMERG 28V DC 2B

4. Limitations:

There are no limitations for this system at the time of this revision.



Digital Bearing and Distance Indicator
Figure 4



31130C00

DBDI MANUAL STBY Switch Installation
Figure 5

2A-34-40: Radio Altimeter System

1. General Description:

The GIV is equipped with two independent AA-300 radio altimeters. Each has a dedicated Receiver / Transmitter (R/T) unit, transmitting antenna and receiving antenna. (Antenna location is shown in Figure 27.) There are no off / on controls (switches) or separate displays for the radio altimeters. The radio altimeters operate continuously when the airplane Main DC buses are powered, using high resolution, short pulse radar signals that are accurate over wide variations of terrain, target reflectivity, weather and airplane altitude.

The radio altimeters provide a continuous readout of airplane altitude above ground level in the operating range of zero to twenty-five hundred (0-2,500) feet. Accuracy of the altitude readout varies with height above ground, with readings between zero and one hundred (0 and 100) feet accurate to within three (3) feet, readings between one hundred and five hundred (100-500) feet have an accuracy within three percent (3%), and readings between five hundred and twenty-five hundred (500-2,500) feet accurate within four percent (4%). Above twenty-five hundred (2,500) feet, the radio altimeters continue to operate, however the radio altitude information is no longer displayed because of the increase in error margin at higher altitudes.

2. System Operation:

Radio altimeter data is sent to the Data Acquisition Units (DAUs) to be digitized and then forwarded over the Avionics Standard Communications Bus (ASCB) to the symbol generators for display on the Primary Flight Display (PFD). A discrete signal is also provided to the landing gear indication system to provide the altitude warning at twelve hundred (1,200) feet if the landing gear is not down and locked (see the discussion in Section 2A-32-30, Extension and Retraction System of this manual).

The PFD displays of radio altitude differ slightly between airplanes with SPZ-8000 and SPZ-8400 Digital Automatic Flight Control Systems (DAFCS). See Figure 6 and Figure 7 for illustrations of the radio altitude display formats.

A. SPZ-8000 Radio Altimeter Display:

The display of radio altitude on the PFD is selected using the SENSOR function on the Display Controller (DC). When SENSOR is selected, a menu appears on the screen of the DC with the available data display options. Pushing the appropriate Line Select Key (LSK) adjacent to the menu item will select that sensor data for display on the PFD. When Radio Altitude (RAD ALT) is displayed, it is shown in digital format just outside the lower right corner of the attitude display. The white digital readout is in ten (10) foot increments between twenty-five hundred (2,500) feet and two hundred (200) feet, and five (5) foot increments below two hundred (200) feet.

If the dual radio altimeters are paired on-side (pilot selected to radio altimeter #1 and copilot selected to radio altimeter #2) the source of the altimeter data is not shown. If the pilot and copilot have selected off-side radio altimeters (pilot to #2 and copilot to #1) the letters RA, and the numbers 1 or 2 as appropriate, are shown in white immediately to the right of the digital readout. If both pilot and copilot are selected to the same radio altimeter, the RA is shown in amber. (The pilot and copilot can select radio altimeter data source with options on the DC.)

A reference set option on the DC allows entering a radio altitude reference for use as a visual indication of Decision Height (DH) during precision approaches predicated on radio altitude data. Pushing the FLT REF button on the DC displays the menu containing RAD ALT. A value for radio altitude DH is entered with the rotary set knob and selected for display with the LSK. The selected DH is displayed on the PFD just above the readout of current radio altitude. As the airplane descends to the selected DH, a flashing white box is displayed around the current radio altitude when it is equal to the selected DH. The white box flashes for five (5) seconds after reaching DH, and remains displayed as long as airplane radio altitude is less than the DH.

When a valid ILS frequency is received for an approach and the airplane descends below two hundred (200) feet radio altitude, a yellow runway symbol is displayed on the PFD, rising from the bottom of the attitude display up to meet the airplane symbol on runway contact at landing. The symbol also moves laterally to indicate displacement from localizer centerline.

An additional reminder of airplane altitude is shown on the altitude tape on the right side of the PFD. As the airplane descends below six hundred (600) feet radio altitude, the color of the altitude tape changes to brown, indicating proximity the ground. The brown altitude tape extends from six hundred (600) feet down to zero (0) feet radio altitude.

NOTE:

This altitude reminder should not be confused with the radio altitude display. The brown portion of the altitude tape corresponds to airplane terrain clearance at a given height above mean sea level (with barometric altimeter set to QNH). For instance, if the airplane is landing on a runway with an elevation of one thousand twenty (1,020) feet MSL, the altitude tape color will change from blue to brown as the airplane descends through one thousand six hundred twenty (1,620) feet MSL.

If there is a system malfunction that causes a loss of valid radio altimeter data, the white digital radio altitude readout is replaced by amber dashes (“- - - - “). If the yellow runway symbol is displayed at the time of the failure, the symbol is removed from the PFD.

B. SPZ-8400 Radio Altitude Display:

The display of radio altitude on the PFD is selected using the SENSOR function on the Display Controller (DC). When the SENSOR button is selected, a menu appears on the screen of the DC with the available data display options. Pushing the appropriate Line Select Key (LSK) adjacent to the menu item will select that sensor data for display on the PFD. When radio altitude (RAD ALT) is displayed, it is shown in digital format at the bottom of the attitude display. The digital readout is in green, indicated in ten (10) foot increments between twenty-five hundred (2,500) feet and two hundred (200) feet, and five (5) foot increments below two hundred (200) feet.

If the dual radio altimeters are paired on-side (pilot selected to radio

altimeter #1 and copilot selected to radio altimeter #2) the source of the altimeter data is not shown. If the pilot and copilot have selected off-side radio altimeters (pilot to #2 and copilot to #1) the letters RA, and the numbers 1 or 2 as appropriate, are shown in white immediately to the right of the digital readout. If both pilot and copilot are selected to the same radio altimeter, the RA is shown in amber. (The pilot and copilot can select radio altimeter data source with options on the DC.)

A reference set option on the DC allows entering a radio altitude reference for use as a visual indication of Decision Height (DH) during precision approaches predicated on radio altitude data. A DH must be entered when the airplane is one hundred (100) feet or more above the desired DH. Pushing the FLT REF button on the DC displays the menu containing RAD ALT. A value for radio altitude DH is entered with the rotary set knob and selected for display with the LSK. The selected DH is displayed on the PFD outside the lower right corner of the attitude indicator. As the airplane descends to the selected DH, an empty box symbol is displayed in the upper right corner of the attitude display when the airplane is within one hundred (100) feet of the selected DH. When the airplane descends to the set DH, an amber DH annunciation is displayed in the box. The DH symbol flashes for five (5) seconds, then remains displayed whenever the airplane is below DH. The readout of actual radio altitude (displayed at the bottom of the attitude indicator) changes color from green to amber when the airplane is one hundred (100) feet or less above the selected DH.

When a valid ILS frequency is received for an approach and the airplane descends below two hundred (200) feet radio altitude, a yellow runway symbol is displayed on the PFD, rising from the bottom of the attitude display up to meet the airplane symbol on runway contact at landing. The symbol also moves laterally to indicate displacement from localizer centerline.

An additional reminder of airplane altitude is shown on the altitude tape on the right side of the PFD. As the airplane descends below six hundred (600) feet radio altitude, the color of the altitude tape changes to brown, indicating proximity the ground. The brown altitude tape extends from six hundred (600) feet down to zero (0) feet radio altitude. A yellow line appears at the top of the brown portion of the tape. The line flashes for the first ten (10) seconds after the brown tape is displayed or until the airplane descends to four hundred (400) feet radio altitude, whichever occurs first. The yellow line rides in front of the brown altitude tape display, indicating height above touchdown.

NOTE:

This altitude reminder should not be confused with the radio altitude display. The brown portion of the altitude tape corresponds to airplane terrain clearance at a given height above mean sea level (with barometric altimeter set to QNH). For instance, if the airplane is landing on a runway with an elevation of one thousand twenty (1,020) feet MSL, the altitude tape color will change from gray to brown as the airplane descends through one thousand six hundred twenty (1,620) feet MSL.

If there is a system malfunction that causes a loss of valid radio altimeter data, the white digital radio altitude readout (and DH setting if selected) is replaced by amber dashes ("- - - - "). If the yellow runway symbol is displayed at the time of the failure, the symbol is removed from the PFD.

3. Controls and Indications:

There are no separate controls or indicators for the radio altimeter system. All cockpit interface with the system is through the DCs and the EFIS (PFD) display system. When the airplane is on the ground with full electrical power, the normal radio altitude readout is -5 (minus five) feet ± 5 feet.

A radio altimeter self-test may be initiated (on-side test only: pilot tests RAD ALT #1, copilot tests RAD ALT #2) using the LSK labeled RAD ALT on the TEST menu of the DC. When the test is in progress, a radio altitude of one hundred (100) feet should be displayed on the PFD:

- Preflight: Entering a DH of 50 feet prior to performing the radio altimeter self-test will test the DH display on the PFD.
- In-Flight Test: Entering a DH of 200 feet prior to performing the radio altimeter self-test will test the DH display on the PFD.

NOTE:

The radio altimeter self-test is inhibited with the AFCS engaged. The self-test is also inhibited by the fault warning computer during certain flight director modes.

A. Circuit Breakers (CBs):

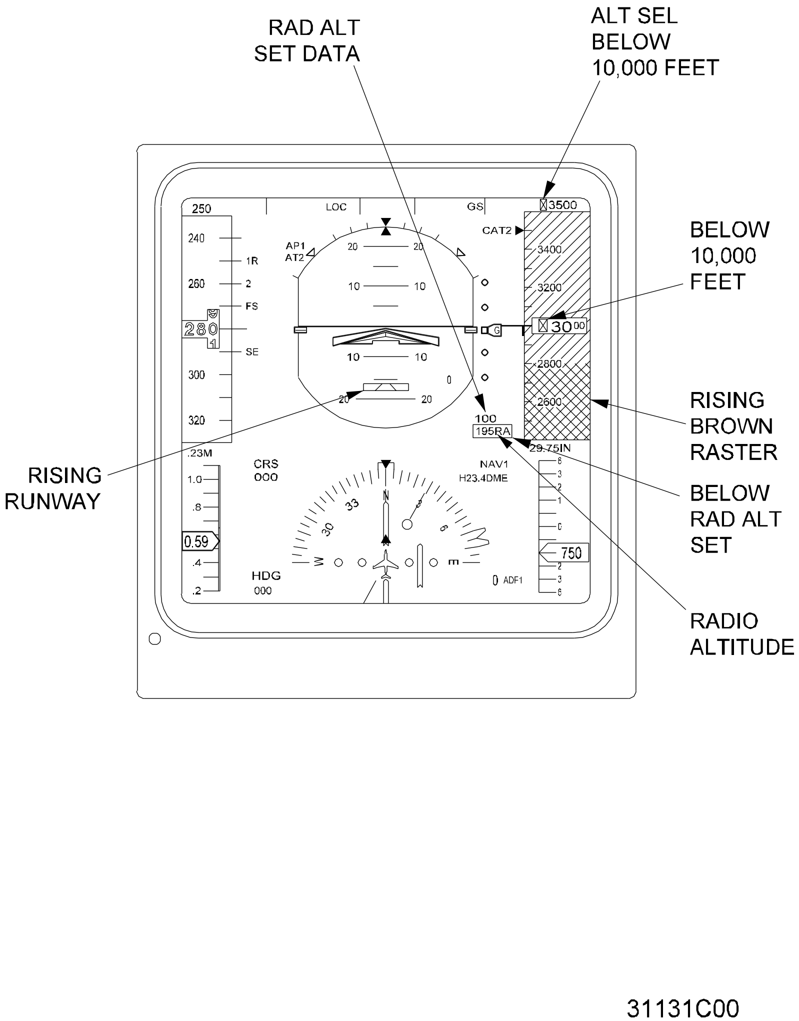
Circuit Breaker Name	CB Panel	Location	Power Source
RADIO ALT #1	CPO	I - 4	L MAIN 28V DC
RADIO ALT #2	CPO	J - 4	R MAIN 28V DC

B. Advisory (Blue) CAS Messages:

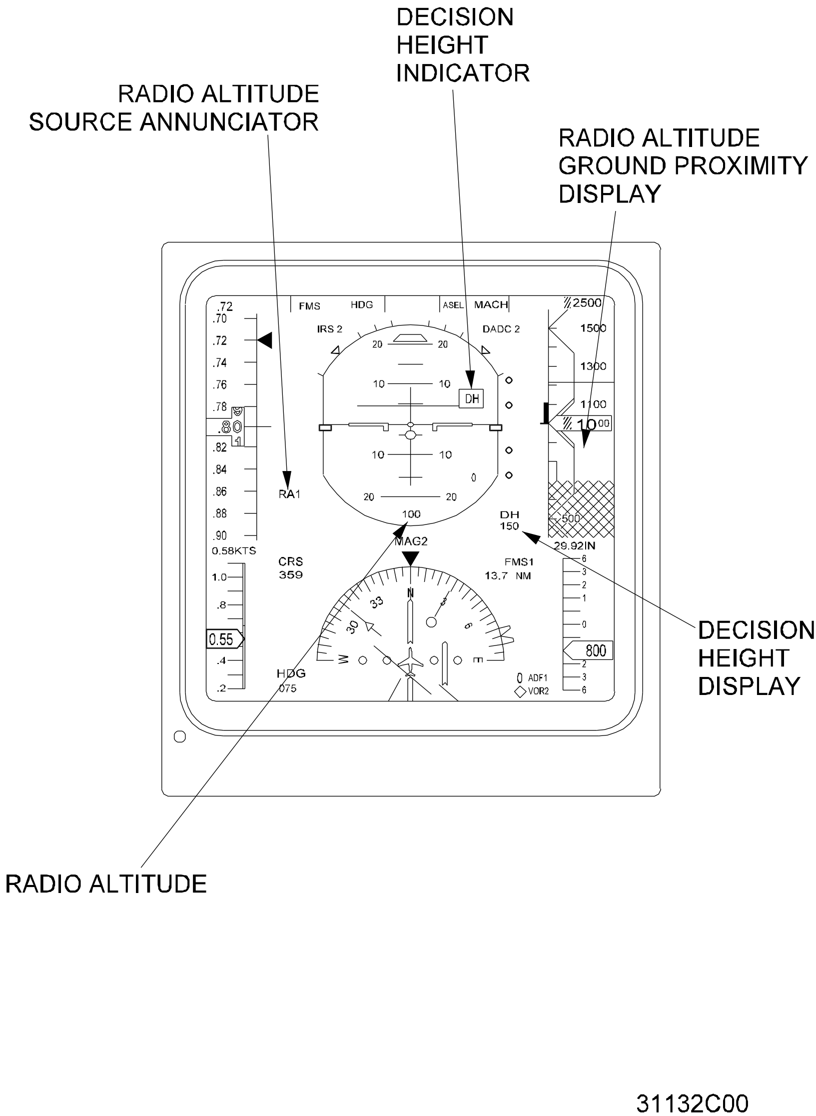
CAS Message	Cause or Meaning
RAD ALT 1 - 2 FAIL	Indicated radio altimeter(s) has failed

4. Limitations:

There are no limitations associated with this system at the time of this revision.



SPZ-8000 Radar Altitude Display
Figure 6



SPZ-8400 Radar Altitude Display
Figure 7

2A-34-50: Enhanced Ground Proximity Warning System (EGPWS)

1. General Description:

The Enhanced Ground Proximity Warning System (EGPWS) is installed during production beginning with airplanes Serial Number (SN) 1390 and subsequent. (The following system description is applicable to the production installed system only. Operators with systems installed by completion outfitters should consult the documentation supplied by the outfitter.) EGPWS provides aural and visual alerts to prevent Controlled Flight Into Terrain (CFIT). Alerts are generated in conditions of terrain clearance danger, severe windshear and excessive deviation below an Instrument Landing System (ILS) glideslope. EGPWS also provides aural notification of excessive bank angles and provides height above runway callouts, including approach minimums, during final approach.

2. Subsystems, Units and Components:

The EGPWS consists of a computer and geographical database interfaced with the SPZ-8400 DAFCS and airplane subsystems over ARINC 429 busses and discrete connections for data sensing and display presentation. See the system diagram in Figure 8. The system uses inputs from the DADCs, radio altimeter, FMS/GPS/IRS, angle of attack (AOA), landing gear and flap position, navigation data and a manually entered approach decision height. This information is integrated with the database in the computer to produce the aural and visual alert messages and the Terrain Awareness Display (TAD) graphic on the cockpit EFIS displays. (The TAD is usually selected to the NAV display, however, the Display Controller has an EGPWS option that allows selection of the TAD to the PFD, with the TAD shown on the HSI similar to the radar display.) Aural alerts and warnings are transmitted over cockpit speakers and through the cockpit interphone system, while alert and warning text messages are displayed on the Primary Flight Display (PFD). EGPWS aural alerts and warnings, except the windshear alert / warning can be inhibited with the GPWS VOICE O-RIDE switch on the O-RIDES panel on the center pedestal, shown in Figure 9 (location may vary).

The computer is located in the right electronic equipment rack and is powered from ϕC of the Left Main 115V AC bus and the Left Main 28V DC bus. The system operates in seven (7) distinct modes:

- Mode 1 - Excessive descent rate.
- Mode 2 - Excessive terrain closure rate
- Mode 3 - Altitude loss after takeoff
- Mode 4 - Unsafe terrain clearance
- Mode 5 - Excessive deviation below glideslope
- Mode 6 - Advisory callouts
- Mode 7 - Windshear alerting

The basic seven modes of operation are available in earlier model GPWS systems, with the alerts and warnings formulated using only airplane sensor data (airspeed, radio altitude, etc.). With the EGPWS, the ability to compare present airplane position, and predicted flight path vectors with data in the geographical database provides expanded warning envelopes and display options that increase situational awareness.

System enhancements improve the basic modes of operation and offer features not previously available. A Terrain Awareness Display (TAD) feature provides a graphic of terrain ahead of the airplane's current flight path. The TAD graphic is in

multicolor and can be selected for continuous view on the NAV display (or PFD in the HSI position). If not selected for continuous view, the TAD will automatically “pop up” on the NAV display when the EGPWS computer detects terrain conflicts. Terrain displayed is within the range selected. The display range is adjusted with the same control as the weather radar, however if the TAD is not selected for continuous display and subsequently “pops up” the default range is ten (10) nautical miles - the range may then be adjusted from the default value. On the TAD, the color red is used to indicate the highest and most hazardous terrain areas, yellow (in varied intensity) identifies less dangerous terrain at elevations equal to or higher than airplane altitude, and green (in varied intensity) denotes areas equal to or below airplane altitude.

NOTE:

The TERR INHIB switch on the O-RIDES panel on the center pedestal, shown in Figure 9, will prevent the “pop up” of the TAD. If TERR INHIB is selected, the basic EGPWS modes 1 - 7 will continue to provide terrain clearance / windshear alerts, but indications will be limited to aural alerts over speakers and interphone and visual text alerts on the PFD.

Features of the Terrain Awareness Display include the following:

- Obstacles are displayed on the TAD when the airplane’s flight path will conflict with any of the known obstacles in the EGPWS database. (The database does not include temporary man-made obstacles covered in NOTAMS).
- A Terrain Peaks feature enhances situational awareness with a digital readout of elevations of the highest and lowest terrain and additional color gradations to further define terrain.
- A Terrain Clearance Floor (TCF) feature alerts the crew of a premature descent based upon current airplane position relative to the nearest runway. The TCF feature is useful in non-precision approaches and is enabled with TAD.
- On airplanes with EGPWS software build -210 -210 (SN 1426 and subsequent), a Runway Field Clearance Floor (RFCF) feature based on airplane position and height above destination runway using a computed Geometric Altitude (GA) provides improved safety margins at locations where the runway is higher than surrounding terrain.
- Geometric Altitude is computed blend of altitude information including GPS data, and at lower altitudes enhanced with Radio Altitude, to reduce or eliminate altimeter errors induced by non standard atmospheric conditions or reference setting errors.
- A EGPWS Envelope Modulation feature is incorporated that compensates for terrain and obstacles at some airports that have historically generated nuisance alerts, or that have environmental characteristics that inhibit needed alerts.

3. Basic Mode Functions:

NOTE:

Basic modes 1 - 6 require radio altimeter information to function. If radio altimeter information is not available, the TAD will continue to provide terrain awareness using Geometric Altitude (GA) inputs (see the discussion of GA in the following text).

A. Mode 1: Excessive Descent Rate:

Excessive descent rate alerts and warnings are generated when the descent is too steep for the margin of altitude below the airplane. The alert / warning envelope upper boundary is approximately 2,500’ radio altitude, and within the envelope, the alert / warning logic is biased for the amount of recovery time for the hazardous condition. See Figure 10. At 2,500’ a descent of approximately 4500 feet per minute (FPM) would initiate a “SINK RATE, SINK RATE” aural alert annunciation, and a descent of approximately 7000 FPM would prompt a “WHOOP, WHOOP, PULL UP” aural warning and the text PULL UP in red is displayed on the PFD. At lower radio altitudes, corresponding lower rates of descent will initiate alerts and warnings, and the envelope margin between alerts and warnings narrows.

If a valid Instrument Landing System (ILS) front course signal has been tuned and received, and the airplane is descending to capture the glideslope from above, the margin of the SINK RATE alert envelope is desensitized to prevent unwanted alerts when the airplane is in a safe position to capture (or recapture) the glideslope.

B. Mode 2: Excessive Terrain Closure Rate:

Mode 2 is the inverse of Mode 1 in that this instance the airplane is in level flight, but is in danger of impacting rapidly rising terrain. Mode 2 is also based on radio altitude and the alert / warning envelope predicated upon closure rates and time remaining for evasive maneuvers. Mode 2 is split into two sub-modes with different parameters depending upon airplane configuration.

Sub-mode 2A is operable during climbout, cruise and initial approach (defined as flaps not in landing configuration and airplane not on ILS centerline). See the envelope shown in Figure 11. In these circumstances, if the airplane approaches rising terrain at a speed such that avoidance time is limited, initially an aural “TERRAIN, TERRAIN” alert message is prompted, and an amber TERRAIN text message is displayed on the PFD. If conditions deteriorate such that ground contact is imminent, an aural “WHOOP, WHOOP, PULL UP, PULL UP” warning is heard over cockpit speakers and headsets, and a red PULL UP text message is displayed on the PFD. The aural and text annunciations will continue until terrain clearance is sufficient that the warning envelope is cleared. If terrain clearance does not continue to increase, the TERRAIN aural and text alerts will continue. In all instances, the visual text alert will continue to be displayed on the PFD until the airplane has gained 300 feet of altitude, forty-five (45) seconds have elapsed, landing flaps have been selected or the flap override switch has been activated.

Sub-mode 2B is a desensitized alert and warning envelope that is automatically activated when landing flaps are selected (or flap override activated), or when within two (2) dots of the centerline of glideslope and localizer during an ILS approach. See Figure 12. This mode is also active during the first sixty (60) seconds after takeoff. The alerts and warnings in sub-mode 2B are the same as those in sub-mode 2A, with additional provisions that if the airplane enters the boundaries of the warning envelope without gear or flaps in the landing configuration, the aural “TERRAIN, TERRAIN” alert will sound with the accompanying text message on the PFD. Further penetration of the envelope will result in “PULL UP” aural warnings and text message until the airplane exits the envelope or the airplane configuration is corrected. If the airplane configuration is correct for landing (gear and flaps down), and a hazardous terrain closure rate exists, the “PULL UP” aural and text warnings are suppressed, and the “TERRAIN” aural and text alerts are prompted until the airplane exits the sub-mode 2B envelope.

C. Mode 3: Altitude Loss after Takeoff:

This mode provides an aural alert for any significant altitude loss after takeoff or when performing a go-around at an altitude of less than two hundred forty-five feet (245’) radio altitude with the gear and flaps not in the landing configuration. The alert envelope, shown in Figure 13, is predicated upon the amount of terrain clearance available below the airplane versus sink rate. Any significant loss of altitude prompts an aural alert of “DON’T SINK, DON’T SINK”. The aural alert is sounded twice only, unless there is a continued loss of altitude clearance.

D. Mode 4: Unsafe Terrain Clearance:

Mode 4 is subdivided into three sub-modes to address specific phases of flight, airplane configurations and airspeeds. The sub-modes 4A, 4B and 4C are active in circumstances similar to those that prompt alerts and warnings under Mode 2 and Mode 3, but provide increased situational awareness when hazardous conditions are not as immediate.

Sub-mode 4A is active during cruise and approach with gear and flaps up, with the alerting envelope predicated on speed and altitude. See Figure 14. (This envelope also provides additional protection against a gear-up landing). Flying in altitude and airspeeds from 1000 feet radio altitude at a speed of 250 knots down to an altitude of 500 feet and a speed of 190 knots prompts an aural “TOO LOW TERRAIN” alert over speakers and headsets and an amber text message TERRAIN on the PFD. If the airplane is still in the clean configuration below 500 feet and at less than 190 knots, the aural alert changes to “TOO LOW GEAR”. Either of these aural alerts is sounded only once, unless there is further decrease of altitude / airspeed of twenty percent (20%) or more.

Sub-mode 4B operates during cruise and approach with the landing gear down, but with the flaps not in landing configuration. See the envelope depicted in Figure 14. Below 1000 feet at 250 knots down to 245 feet at 159 knots with the flaps not fully extended prompts a “TOO LOW TERRAIN” aural alert and the display of an amber TERRAIN on the PFD. Below 245 feet and less than 159 knots, the aural alert changes to “TOO LOW FLAPS”. The aural alerts are sounded only once unless there is a further twenty percent (20%) degradation of clearance.

Sub-mode 4C operates during climbout toward rising terrain that produces a decrease in vertical clearance, but is not severe enough to prompt activation of mode 2. See the alert envelope shown in Figure 15. After takeoff or a go-around below 245 feet, and the gear and flaps not in landing configuration, the airplane must continue to gain terrain clearance at a rate that is equal to or exceeds 75% of the radio altitude averaged over the previous fifteen (15) seconds, with no decrease. This envelope is upwardly limited at 500 feet radio altitude at airspeeds less than 190 knots, and expands linearly to 1000 feet at 250 knots. If airplane climb does not meet the envelope gradients, an aural alert “TOO LOW TERRAIN” is heard and the TERRAIN text alert is displayed on the PFD.

E. Mode 5: Excessive Deviation Below Glideslope:

Mode 5 provides terrain clearance alerts during ILS approaches. The alerts are triggered at two different levels, depending upon how closely the airplane is aligned with the glideslope and the terrain clearance available below the airplane. The alerting envelope is shown in Figure 15. For glideslope alerts to be operative, the airplane must be within two (2) dots of localizer centerline, gear and flaps in the landing configuration and a valid front course ILS signal received. As the airplane descends below 1000 feet radio altitude on the localizer, any deviation below glideslope center that exceeds 1.3 dots prompts an aural “GLIDESLOPE” alert and illumination of the BELOW G/S lights below the cockpit glareshield. This aural alert is sounded at only half of the volume of normal aural alerts, and is called a “soft” alert. If the airplane deviates twenty percent (20%) further from the 1.3 dot displacement, the “soft” alert is repeated at increasingly faster rates.

The BELOW G/S light is a dual function switchlight installation. The top half of the switchlight is labelled BELOW G/S and illuminates amber when the airplane deviates outside of the glideslope alerting envelope. The bottom half of the light is labelled G/S INHIBIT. Pushing the switchlight will inhibit further glideslope alerts. When the inhibit function is selected by pushing the switchlight, the legend G/S INHIBIT illuminates blue. The inhibit switchlight may be used to cancel Mode 5 alerts at any time the airplane is below 2000 feet radio altitude. Once cancelled, Mode 5 alerting is reset when the airplane descends below 30 feet, climbs above 2000 feet or the ILS frequency is deselected then retuned. Mode 5 would then be available for a subsequent approach in the event of a go-around

As the airplane descends to 300 feet and lower during the ILS approach, a deviation from glideslope center of two (2) dots or more prompts aural alerts “GLIDESLOPE” at normal (louder) volume. The aural alert is sounded every three (3) seconds until the airplane returns to within 1.3 dots of glideslope center.

Both the 1.3 dot “soft” alert and the 2 dot normal aural alerts are desensitized below 150 feet radio altitude to allow for glideslope beam variations and to reduce the possibility of nuisance (unwarranted) alerts.

If the airplane is maneuvering at low altitude to capture the localizer for an ILS approach, the upper altitude limit of the glideslope deviation alert envelope is reduced to 500 feet radio altitude if the airplane is descending at less than 500 FPM.

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F. Mode 6: Advisory Callouts:

The mode 6 advisory callouts are aural notifications of specific altitudes and excessive bank angle. There are no visual text messages or displays associated with mode 6. The specific altitudes announced over the cockpit speakers and interphone system are selected upon installation of the EGPWS system, as are the optional instrument approach minimum altitude and excessive bank angle callouts. The following altitude callouts are most commonly selected, but others may be selected by individual operators. Different selections should be noted by system placards in the cockpit (if confirmation of selected altitudes is required, note the callouts during EGPWS self-test):

- “ONE THOUSAND”
- “FIVE HUNDRED”
- “FOUR HUNDRED”
- “THREE HUNDRED”
- “TWO HUNDRED”
- “ONE HUNDRED”
- “FIFTY”
- “FORTY”
- “THIRTY”
- “TWENTY”
- “TEN”

The above listed aural callouts are sounded by the EGPWS when the radio altitude associated with the callout is reached. In addition to these standardized callouts, aural notification of descent within one hundred feet (100’) of instrument approach minimum altitude and reaching approach minimum altitude will take place if the crew has manually entered the minimum altitude (MDA or DH). The callouts are “APPROACHING MINIMUMS” and “MINIMUMS” and are sounded only once during the approach. (Other aural notification options may be selected and programmed during system installation - monitor the callouts during self-test for verification). If altitude callouts are not desired, selecting the RAD ALT VOICE O/R switch on the O-RIDES panel (shown in Figure 9) will inhibit the annunciation of callouts.

An aural notification of excessive bank angle is a standard option (others are available for operator customizing). The business airplane bank angle limits are set at forty degrees (40°) above 150 feet radio altitude. See Figure 16. Below 150 feet, the bank angle limit is proportionally decreased with altitude, down to ten degrees (10°) at thirty (30) feet. The feature is inhibited below five (5) feet of altitude. If the airplane exceeds the bank angle limit for altitude, an aural notification of “BANK ANGLE, BANK ANGLE” is heard. If the bank limit is exceeded, the airplane must return to a bank angle of thirty degrees (30°) or less to reset the excessive bank callout.

G. Mode 7: Windshear Alerting:

If the airplane encounters environmental conditions often associated with a windshear at lower altitudes, aural and visual caution and warning alerts

are provided to the crew by mode 7 of the EGPWS. The operating envelope for the windshear alert is shown in Figure 17.

Windshear alerting is computed using inputs from air data sensors for pitot/ static information, and accelerometers for sensing forces on the airplane. The computer interfaces with the IRSs and DADCs for attitude and TAT information, and with discretes for flap position and nutcracker switch data. The computer generates windshear cautions and warnings that are communicated via ARINC 429 bus inputs to the Symbol Generators for display on the PFDs.

Mode 7 is operable between ten (10) and fifteen hundred (1500) feet radio altitude during takeoff, approach or go-around. The Windshear Computer detects suddenly changing headwinds and tailwinds, excessive updrafts and downdrafts, and other factors indicating an impending microburst. Moderate conditions of increasing headwind and updraft that do not immediately hazard the airplane result in the aural message “CAUTION, WINDSHEAR” over cockpit speakers and interphone, and the amber text message WINDSHEAR displayed on the PFD. More severe conditions (decreasing headwinds and downdrafts) prompt windshear warnings. The aural warning “WINDSHEAR, WINDSHEAR, WINDSHEAR” is sounded, and the red text warning WINDSHEAR is displayed on the PFD.

For both cautions and warnings, the aural message is repeated only once, but the text message on the PFD remains in view until the airplane exits the windshear conditions. The parameters that prompt the windshear cautions / warnings are adjusted as a function of available climb performance, flight path angle, airspeeds that significantly vary from normal takeoff / approach / go-around speeds, and unusual fluctuations in Static Air Temperature (SAT) often associated with microbursts.

4. Enhanced Mode Functions:

With the ability to compare accurate airplane position from FMS / IRS / GPS systems with the terrain database stored within the computer, EGPWS is able to improve the function of the seven basic modes of operation and provide the flight crew with additional features.

A. Envelope Modulation:

EGPWS modifies the alert envelopes of some basic modes at specific geographical locations where there are terrain features that are known to cause nuisance alerts or to inhibit needed alerts. The alert envelope for basic modes 4 (unsafe terrain clearance), 5 (excessive glideslope deviation) and 6 (advisory callouts) is expanded at some locations that are known to require additional terrain clearance, while at other locations modes 1 (excessive descent rate), 2 (excessive terrain closure rate) and 4 (unsafe terrain clearance) are desensitized to avoid nuisance alerts generated by known non-hazardous terrain.

B. Terrain Clearance Floor:

In conjunction with the Terrain Awareness Display (TAD), the Terrain Clearance Floor (TCF) function alerts the flight crew at any time the airplane descends below the TCF defined altitude regardless of airplane configuration. The alert envelope is depicted in Figure 18. A descent below the TCF will trigger the aural alert “TOO LOW TERRAIN” and the amber text message TERRAIN on the PFD. This feature is operational at all times

unless the TAD is inhibited. The floor is defined as seven hundred feet (700’) above terrain for all areas except within fifteen (15) miles of airport with a runway of 3500 feet or longer that is in the EGPWS database. As the airplane approaches a database airport, the floor drops to four hundred feet (400’) between twelve (12) miles and four (4) miles of the center of the runway. On airplanes with EGPWS software version -210 -210 and higher (SN 1426 and subsequent), the inner alert floor is lowered to two hundred forty-five feet (245’) and positioned closer to the center of the runway (typically 1/3 NM to 1 NM) due to the higher resolution of airplane position relative to the terrain database. For SN 1426 and subsequent, this software version also provides an identification logic that determines the most likely destination runway based on airplane position and navigation information.

C. Runway Clearance Floor:

The Runway Clearance Floor (RCF) is very similar to the TCF, and is available on -210 -210 software (SN 1426 and subsequent) equipped airplanes. The RCF uses a computed Geometric Altitude (see the description of Geometric Altitude in the following sections) in lieu of radio altitude. RCF provides improved terrain alerting in locations where the runway is located at a much higher altitude than the surrounding terrain, or where an approach to the runway transits a steep decrease in terrain clearance. If the airplane enters the RCF alert envelope, an aural “TOO LOW TERRAIN” alert is sounded and the amber TERRAIN text message is displayed on the PFD. The aural alert is not repeated unless there is a further twenty percent (20%) decrease in terrain clearance. The amber TERRAIN text message remains displayed on the PFD until the RCF alert envelope is exited.

D. Look Ahead Terrain Alerting:

The EGPWS is able to anticipate potential hazards to the airplane by using the terrain database and algorithms based on airplane position, flight path vertical component (climb or descent), and airplane track and speed relative to the terrain database. See Figure 19. The EGPWS projects a terrain alert envelope ahead of the airplane, above and below the projected flight path and laterally within ¼ mile and out to within ± three degrees (3°) of track (or more if the airplane is turning). If the system algorithms predict that the airplane will encounter hazardous terrain within sixty (60) seconds, an aural “TERRAIN, TERRAIN” caution is sounded and the amber TERRAIN text message is displayed on the PFD. The aural caution is repeated every seven (7) seconds while the airplane is in the caution alert envelope, and the text TERRAIN remains displayed until the airplane clears the terrain caution envelope. If the airplane is projected to encounter a terrain hazard within thirty (30) seconds, an aural “TERRAIN, TERRAIN” followed by a “PULL UP, PULL UP” warning is sounded, and a red text message PULL UP is displayed on the PFD. The aural warnings are repeated continuously and the red PULL UP is displayed until the airplane exits the terrain warning alert envelope.

E. Terrain Awareness Display (TAD):

The Terrain Awareness Display (TAD) is graphic representation of the terrain within two thousand feet (2000’) above or below the airplane, usually selected to the NAV / RADAR cockpit EFIS display by the EGPWS options on the Display Controller. (The TAD may be selected for view on

the PFD, in which case the display is similar to the weather radar display mode in HSI format.) The terrain display defined altitudes are shown in Figure 20. The TAD as it appears on cockpit EFIS displays is seen in Figure 21. This display will automatically appear on the NAV display if a terrain conflict is detected. The automatic display function is initiated by the EGPWS computer that switches the Display Controller to the TAD mode. If desired by the crew, the automatic TAD function may be inhibited with the TERR INHIB switch on the O-RIDES panel located on the center pedestal.

CAUTION

THE TERRAIN AWARENESS DISPLAY (TAD) IS INTENDED FOR USE ONLY AS AN ADVISORY OF POTENTIALLY THREATENING TERRAIN AHEAD. IN NO WAY SHOULD THE FLIGHT CREW USE THE DISPLAY FOR NAVIGATION OF THE AIRPLANE OR FOR GUIDANCE IN STEERING THE AIRPLANE CLEAR OF TERRAIN.

The TAD offers a plan view image of surrounding terrain in patterns of green, yellow and red in varying densities. Each specific color and intensity represents terrain and / or obstacles above, level with or below the airplane's altitude based upon airplane position relative to the geographic database. If the airplane is in an area not covered by the database (typically near the poles) the display is low density magenta. Terrain that is more than two thousand feet (2000') below the airplane is not displayed, nor is terrain within four hundred vertical feet (400') of the elevation of the nearest airport runway. See the table at the end of this topic for a full description of the significance of the colors and densities of the TAD function.

With the incorporation of the Peaks function, the TAD presents a digital readout of the elevations of the highest and lowest terrain and / or obstacle currently displayed. The numerical values of the readout are in hundreds of feet above mean sea level (MSL), thus a display of 125 equals (=) 12,500 feet MSL. The elevation values are displayed in the same color as the terrain of that elevation. The elevation of the highest terrain indicated in red, the lowest in green. If there is no significant variation in terrain elevation, for instance over flat terrain, only the highest elevation is indicated numerically. The numerical elevation values are no longer displayed when the airplane is five hundred feet (500') or less above the terrain (250' if the landing gear is extended).

When the image is initially presented, the ten (10) mile range is the default value - other range values must be manually selected. When potential terrain conflicts prompt caution or warning alerts, the terrain and / or obstacle is depicted in solid yellow for cautions and solid red for warnings. Additionally, during alerts the image scale of the area immediately surrounding the hazard is enlarged to better identify a small obstacle or terrain feature.

Color	Indication
Solid Red	Terrain / Obstacle Threat Area - Warning
Solid Yellow	Terrain / Obstacle Threat Area - Caution
50 % Red Fill	Terrain / Obstacle that is more than 2000 feet above airplane altitude
50 % Yellow Fill	Terrain / Obstacle that is between 1000 feet and 2000 feet above airplane altitude
25 % Yellow Fill	Terrain / Obstacle that is 500 feet (250 feet with landing gear extended) below to 1000 feet above airplane altitude
Solid Green (Peaks display)	Shown only when no Red or Yellow Terrain / Obstacles are within range of the display. The highest Terrain / Obstacle is not within 500 feet (250 feet with landing gear extended) of airplane altitude
50 % Green Fill	Terrain / Obstacle that is between 500 feet (250 feet with landing gear extended) and 1000 feet below airplane altitude
50 % Green Fill (Peaks display)	Terrain / Obstacle that is in the middle elevation band when there is no Red or Yellow Terrain / Obstacle within range on the display
16 % Green Fill	Terrain / Obstacle that is between 1000 feet an 2000 feet below airplane altitude
16 % Green Fill (Peaks display)	Terrain / Obstacle that is in the lower elevation band when there is no Red or Yellow Terrain / Obstacle within range on the display
Black	No significant Terrain / Obstacle
16 % Cyan Fill (Peaks display)	Water at mean sea level elevation (0 feet MSL)
Magenta Fill	Unknown terrain. No terrain data in the database for the magenta area shown

F. Geographic Altitude:

The alerts and displays generated by the EGPWS are most accurate when airplane altitude can be determined with a high degree of certainty. To obtain the highest accuracy in measuring airplane altitude, the EGPWS computes a Geographic Altitude (GA). GA is a blended altitude derived from all altimeter data sources available, and includes:

- Non-corrected standard altitude
- Runway calibrated altitude computed during takeoff
- GPS calibrated altitude
- Radio altitude calibrated during approach
- Barometric altitude, corrected for local conditions if available

For each of these readings, a Vertical Figure of Merit (VFOM) is determined in order to calculate the importance of the individual reading in blending the final GA computation. The final computed GA value is more accurate than the value of individual sensor readings and allows a more

precise determination of separation between the airplane and the terrain in the EGPWS database.

NOTE:

A terrain awareness display with degraded accuracy remains available if radio altitude information is lost. A GA is formulated from the available altitude data sources for computing the terrain display. A display generated without radio altitude should be used only as a general cue for terrain awareness, and should not be relied upon for navigational purposes.

5. Controls and Indications:

A. Circuit Breakers (CBs):

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
EGPWS AC	CPO	D-10	L MAIN 115V AV ϕC
EGPWS DC	CPO	E-10	L MAIN 28V DC

B. Advisory (Blue) CAS Messages:

CAS Message:	Cause of Meaning:
GPWS FAIL	Ground Proximity Warning System (GPWS) has failed.
WINDSHEAR FAIL	Failure of essential input data from one or more of the following: AOA, Stall Warning System or IRS accelerometers.
TERRAIN INHIBITED	Terrain inhibit switch selected ON.
TERRAIN NOT AVAIL	Airplane position cannot be determined due to failure / malfunction in GPS and / or IRS systems.

C. System Test:

The GPWS switch on the TEST panel, located on the cockpit center console (location varies) initiates an EGPWS system self-test. Prior to initiating a self-test, determine that the following conditions are met:

- Normal airplane power is available and EGPWS is ON
- No O-RIDE switches are selected (TERR INHIB, RAD ALT VOICE O/R or GPWS VOICE O/R)
- No GPWS inoperative annunciations are displayed on CAS

Pressing the GPWS TEST switch will result in the following indications if the system is functioning normally:

- CAS messages GPWS FAIL, WINDSHEAR FAIL, TERRAIN INHIBITED and TERRAIN NOT AVAIL displayed
- Amber caution light BELOW G/S on
- “GLIDESLOPE” annunciation over cockpit speakers and interphone
- Amber BELOW G/S light extinguishes
- Blue G/S INHIBIT light on
- Blue G/S INHIBIT light extinguishes

- Red PULL UP text warning displayed on PFD
- “PULL UP” aural annunciation over cockpit speakers and interphone
- Red PULL UP text warning clears
- Red WINDSHEAR text warning displayed on PFD
- “WINDSHEAR, WINDSHEAR, WINDSHEAR” aural annunciation over cockpit speakers and interphone
- Red WINDSHEAR text warning clears
- Amber WINDSHEAR text caution displayed on PFD
- Amber WINDSHEAR text caution clears
- Red PULL UP text warning displayed on PFD
- “TERRAIN, TERRAIN” aural warning followed by “PULL UP, PULL UP” aural warning annunciated over cockpit speakers and interphone
- Terrain test pattern shown on cockpit displays
- Red PULL UP text warning clears
- CAS messages GPWS FAIL, WINDSHEAR FAIL, TERRAIN INHIBITED and TERRAIN NOT AVAIL clear
- Terrain test pattern clears on cockpit displays

6. Limitations:

A. Flight Manual Limitations:

(1) Pilot’s Manuals:

The Honeywell Enhanced Ground Proximity Warning System Pilot’s Guide, Publication Number 060-4241-000, Revision D, dated March 2000 (or later approved revision appropriate to the software version below) shall be immediately available to the pilots for -208 -208 (SN 1390 through 1425) or -210 -210 (SN 1426 and subsequent).

(2) Clearance:

Pilots are authorized to deviate from their current Air Traffic Control (ATC) clearance to the extent necessary to comply with an EGPWS warning.

(3) Navigation:

Navigation is not to be predicated upon the use of the Terrain Display.

(4) Database:

The EGPWS database, displays and alerting algorithms currently account for man-made obstructions.

(5) Terrain Display:

The Terrain Display is intended to serve as a situational awareness tool only, and may not provide the accuracy and / or fidelity on which to solely base terrain avoidance maneuvering.

Terrain Display shall be selected OFF when within 15 NM of landing at an airport when:

- The airport has no published instrument approach procedure (-104 -104 software version only).

- The longest runway is less than 3500 ft in length.
- The airport is not in the EGPWS database.

(6) TAWS:

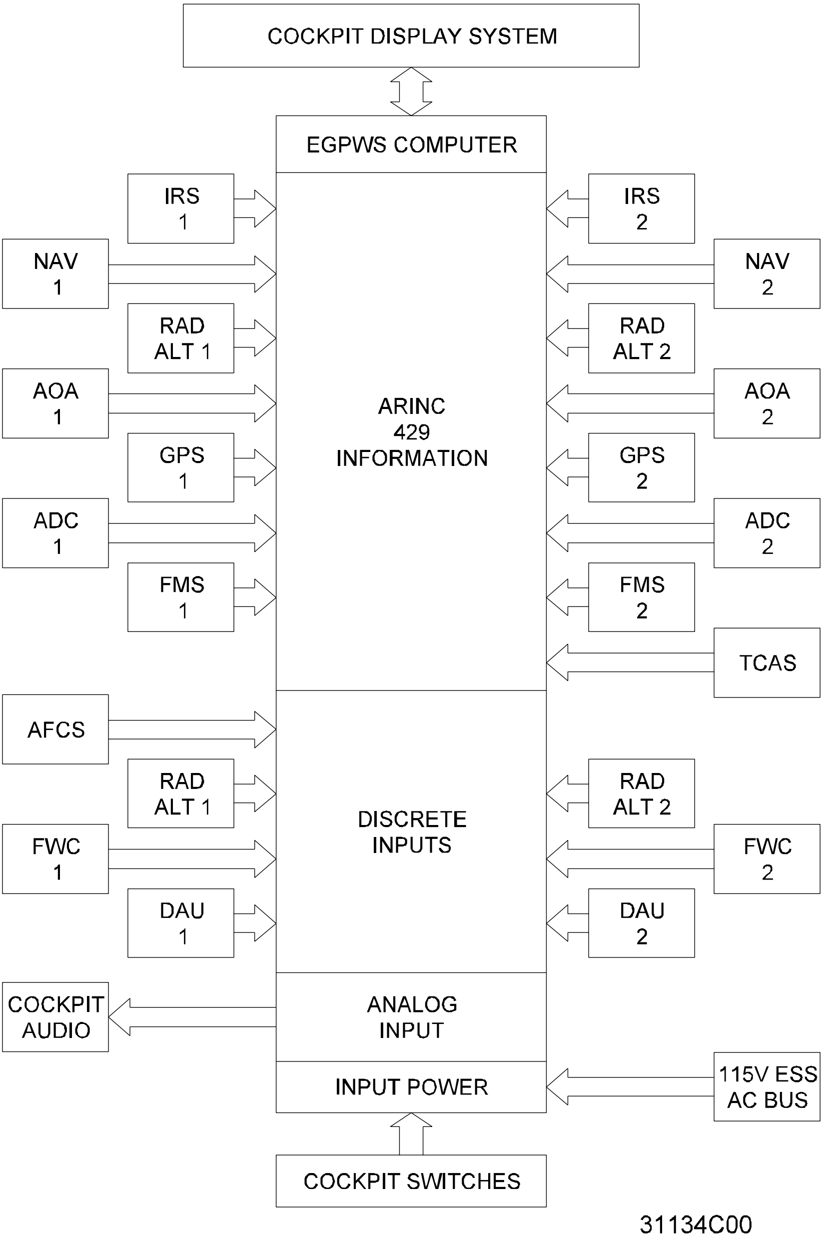
The production EGPWS installation meets the requirements for Class A TAWS as defined in Advisory Circular AC 25-23.

7. System Notes:

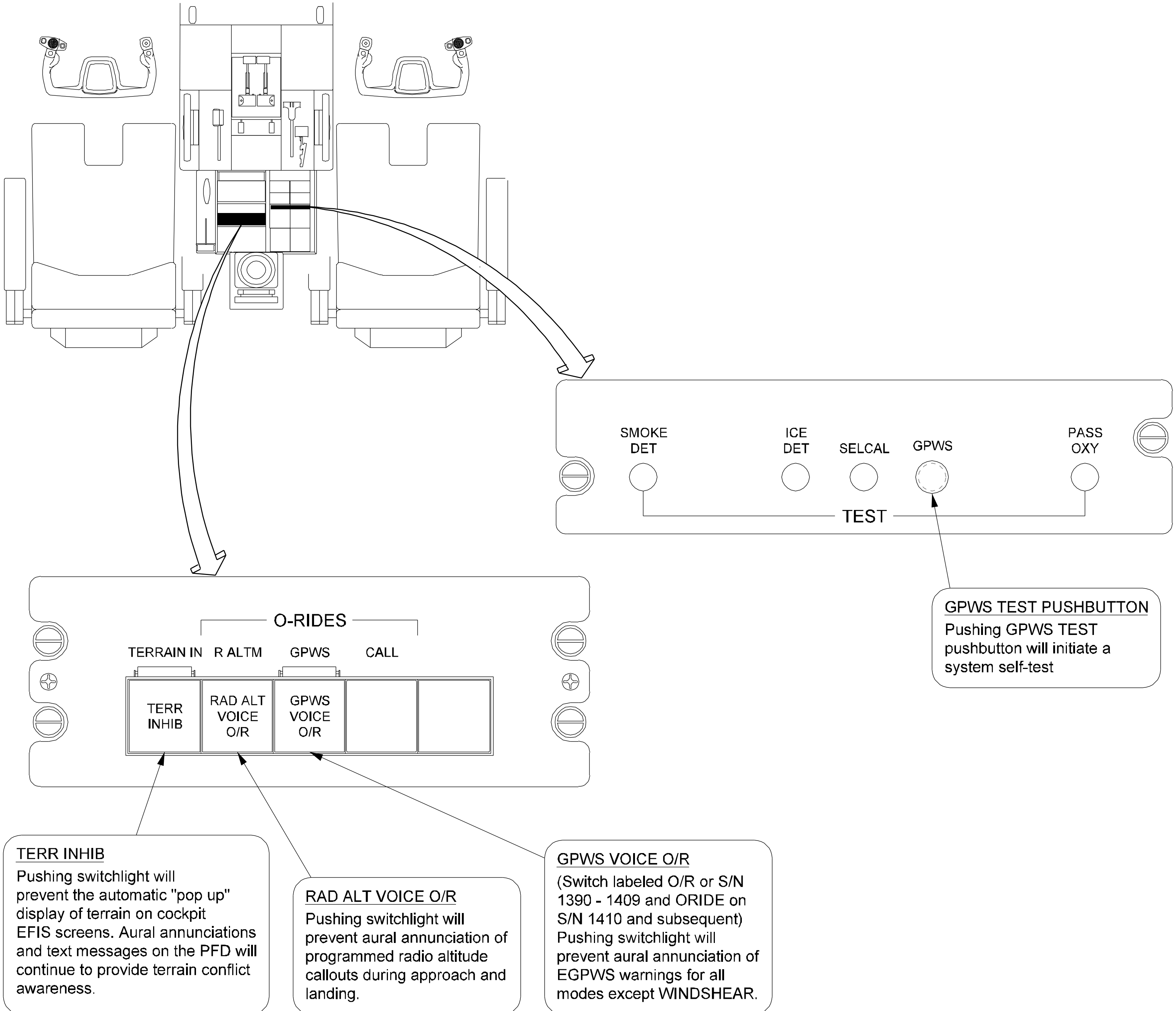
The EGPWS database consists of three (3) smaller databases:

- Terrain database that covers most of the earth
- Obstacles database that covers all charted obstacles in North America and slightly beyond
- Runway database that covers all runways at least 3,500 feet in length

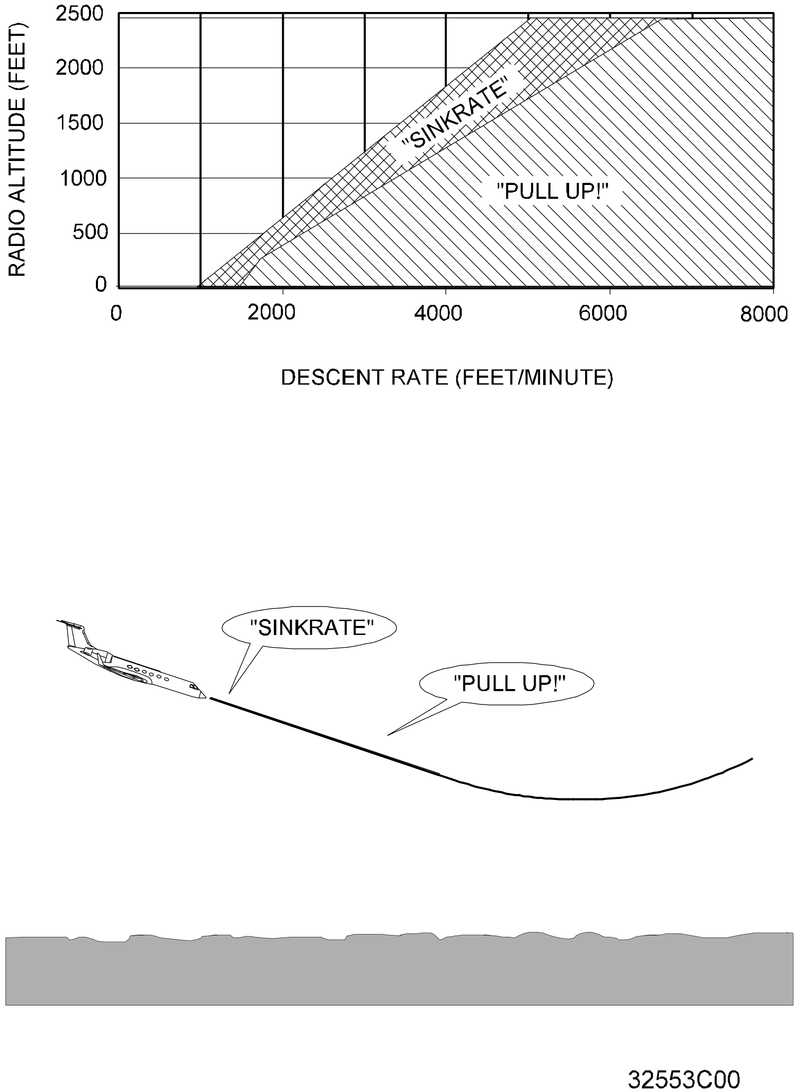
The database is updated when significant changes occur. The database updates are on a PCMCIA card available from Honeywell. The number of the latest database version is listed on the Honeywell website http://www.egpws.com/ or by calling the EGPWS hotline at (800) 813-2099.



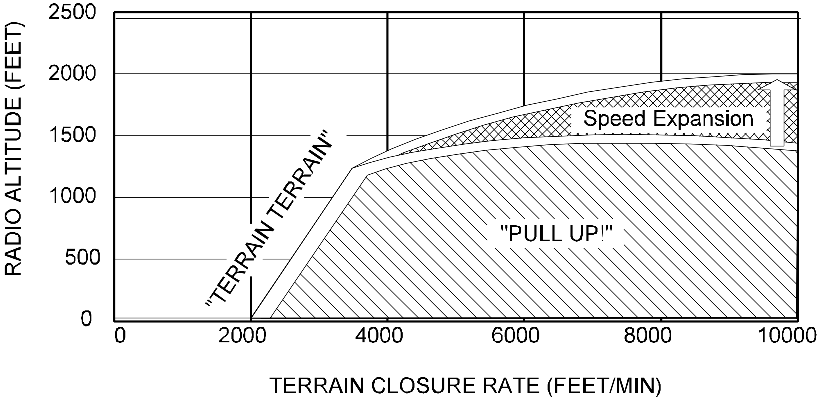
EGPWS System Diagram
Figure 8

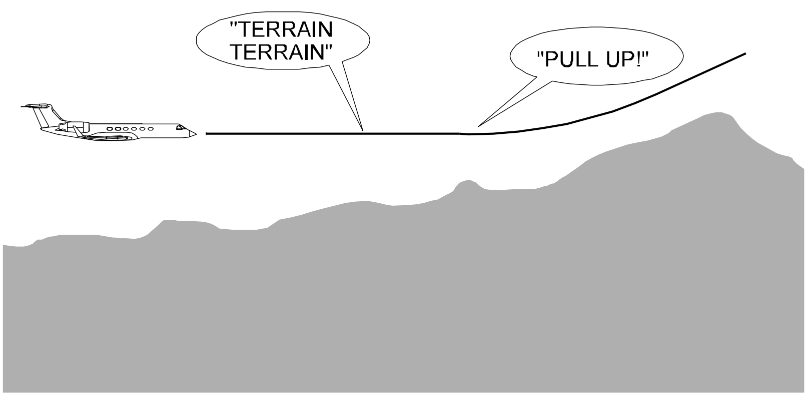


O-RIDES / TEST Panels
on Center Pedestal
Figure 9



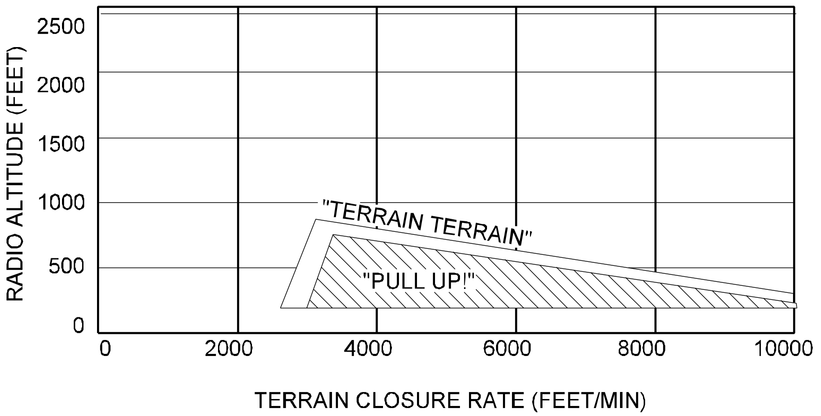
Excessive Descent Rate Envelope - Mode 1
Figure 10

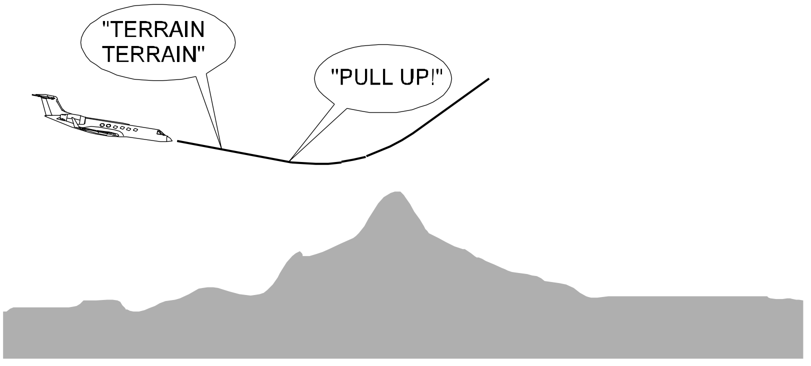




32554C00

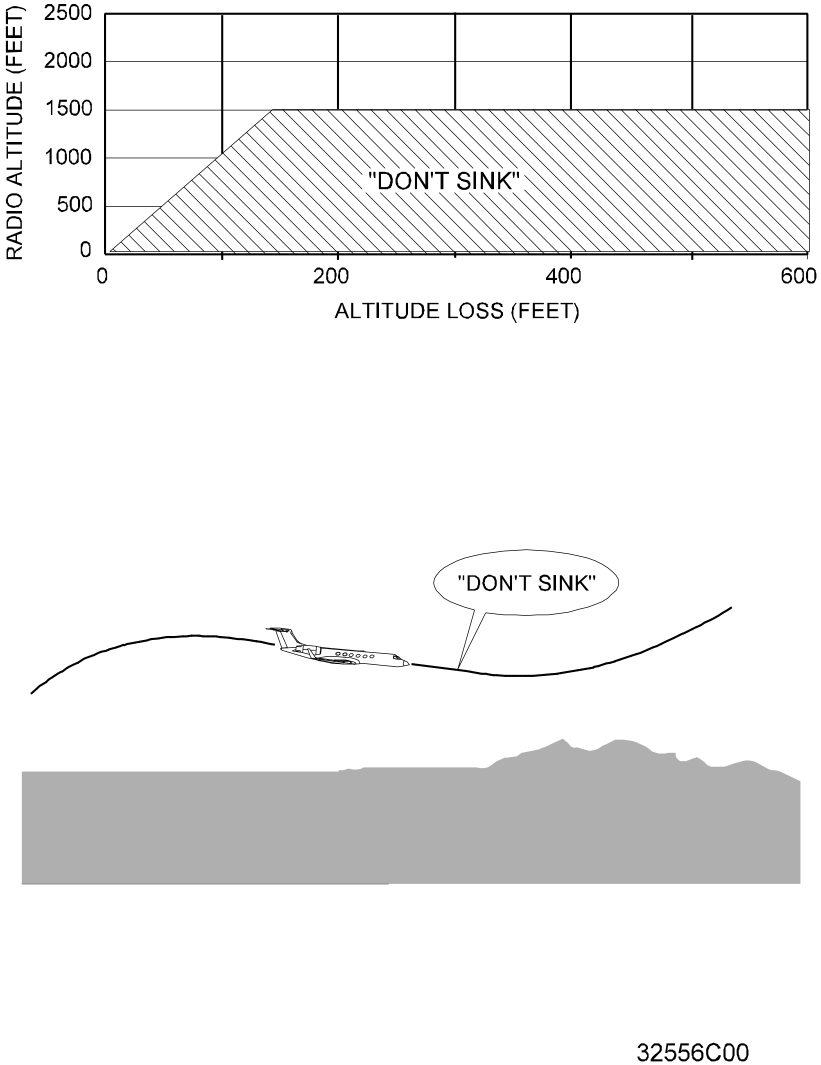
Excessive Terrain Closure Envelope - Mode 2A
Figure 11





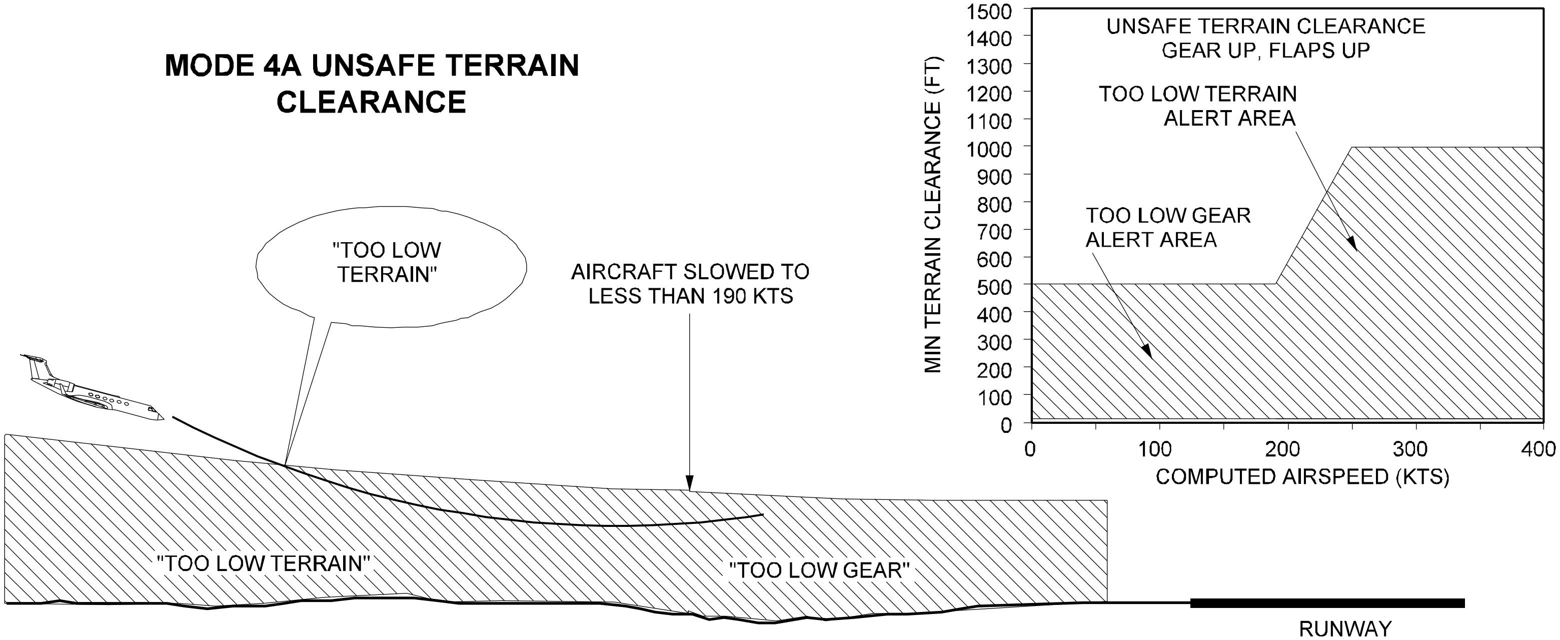
32555C00

Excessive Terrain Closure Envelope - Mode 2B
Figure 12

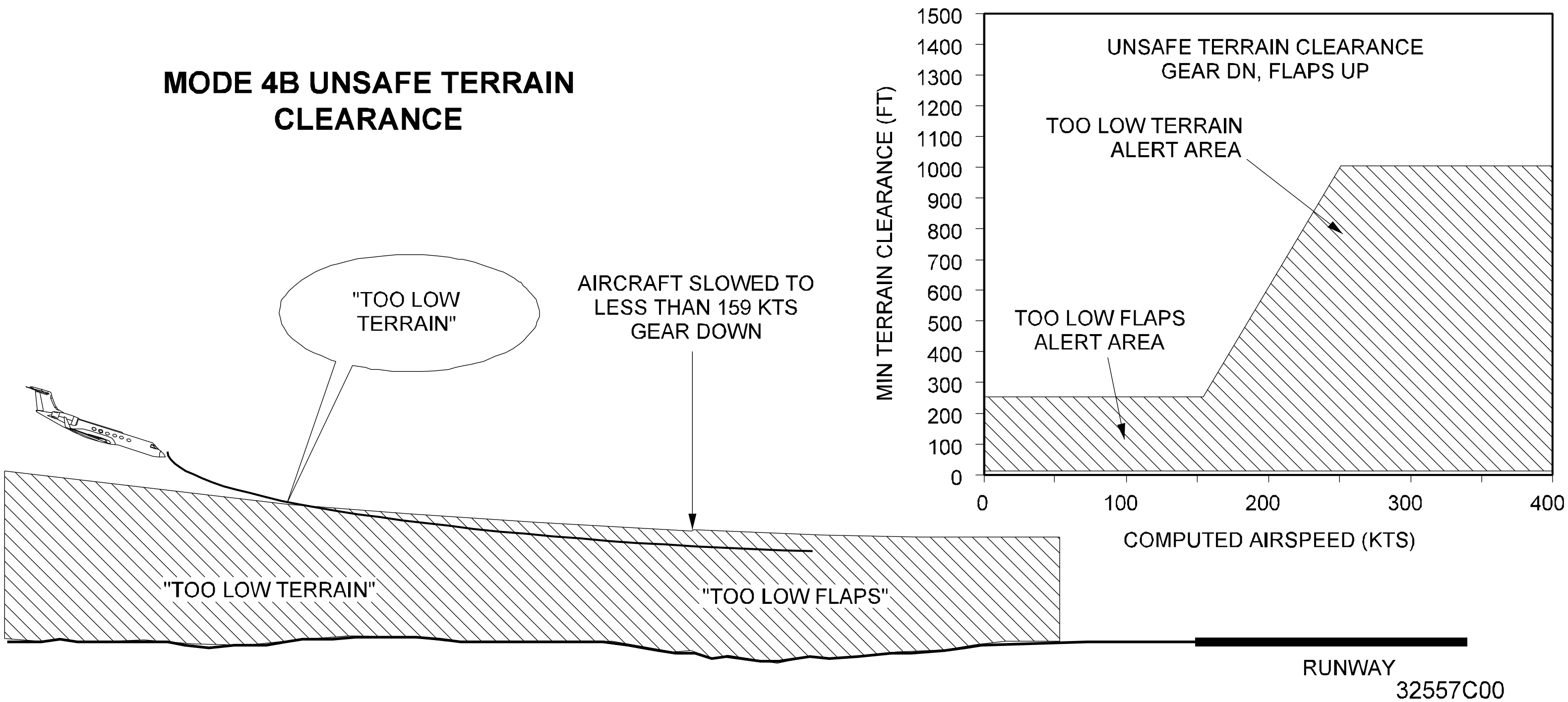


Altitude Loss After Takeoff Envelope - Mode 3
Figure 13

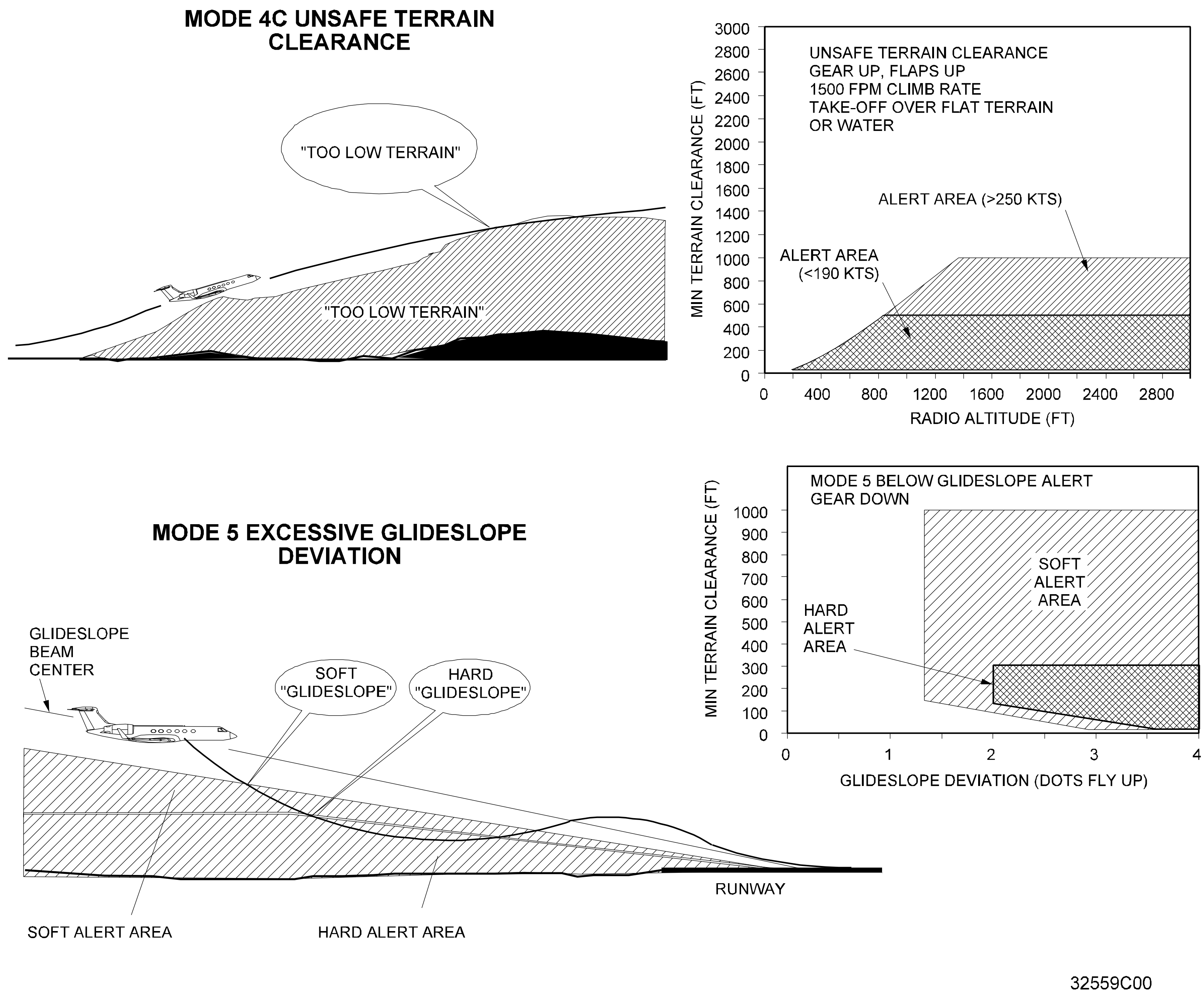
MODE 4A UNSAFE TERRAIN CLEARANCE



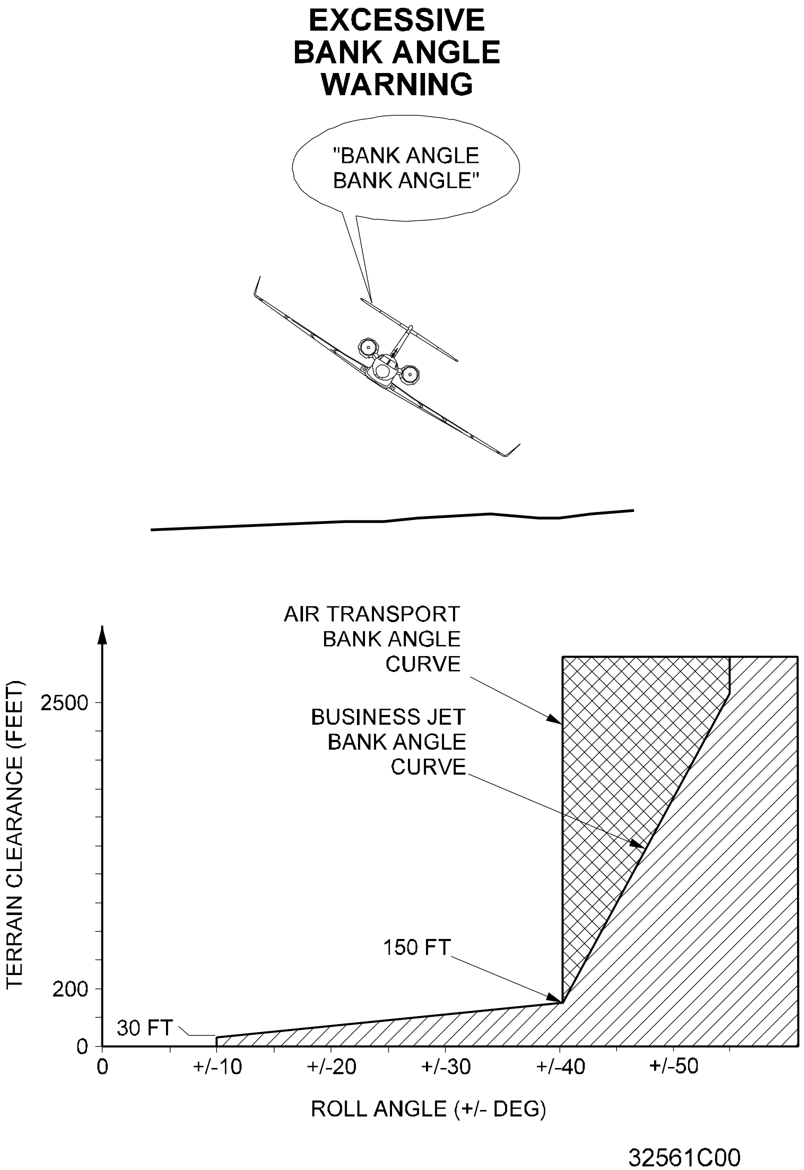
MODE 4B UNSAFE TERRAIN CLEARANCE



Unsafe Terrain Clearance Envelope - Mode 4A and Mode 4B
Figure 14



Unsafe Terrain Clearance
Envelope - Mode 4C and
Excessive Glide Slope
Deviation - Mode 5
Figure 15



Excessive Bank Angle Envelope - Mode 6
Figure 16

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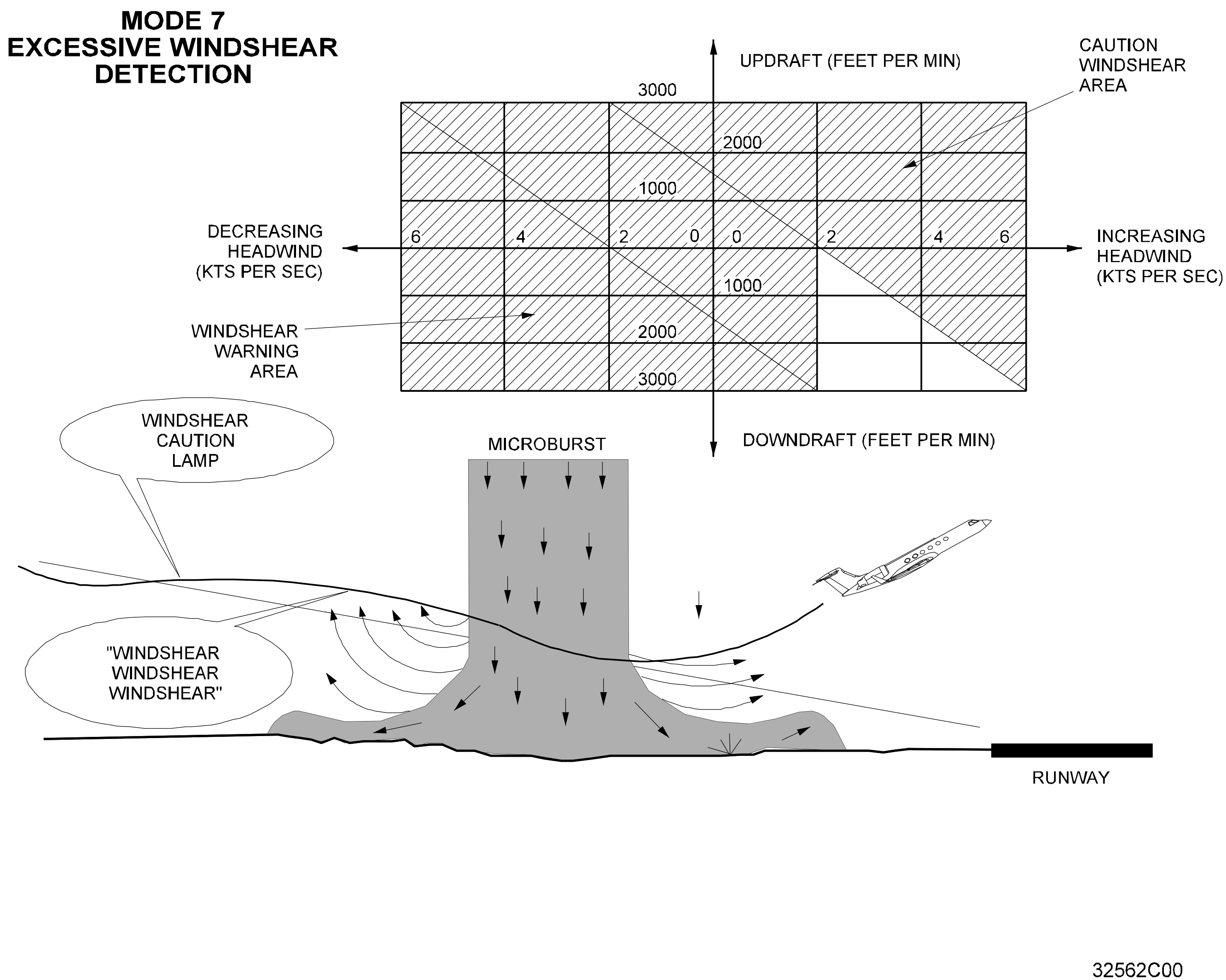
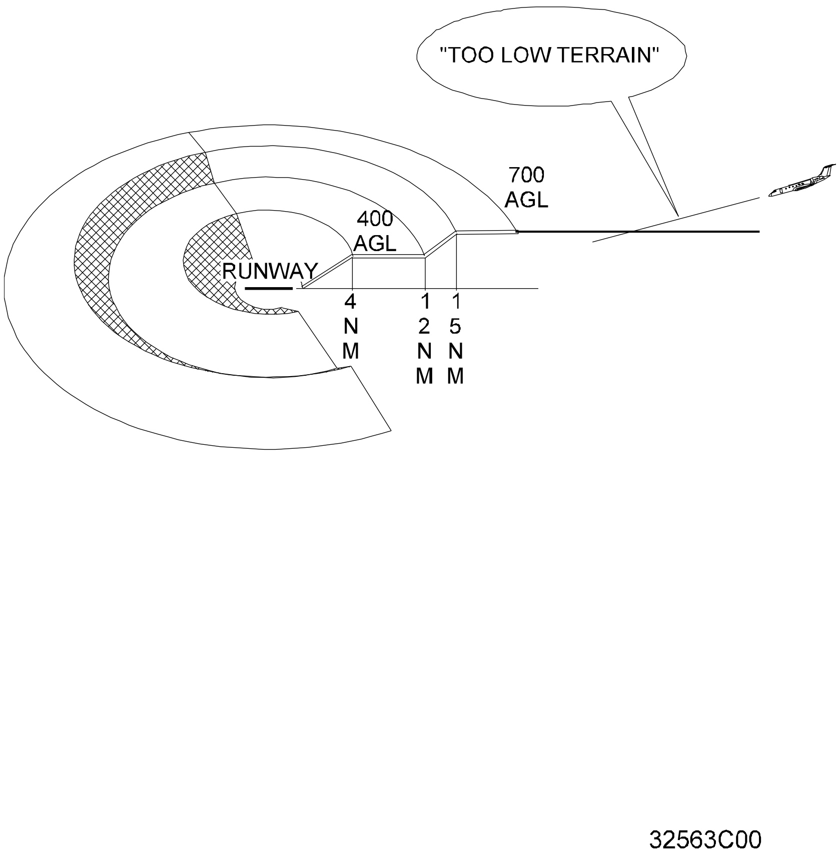
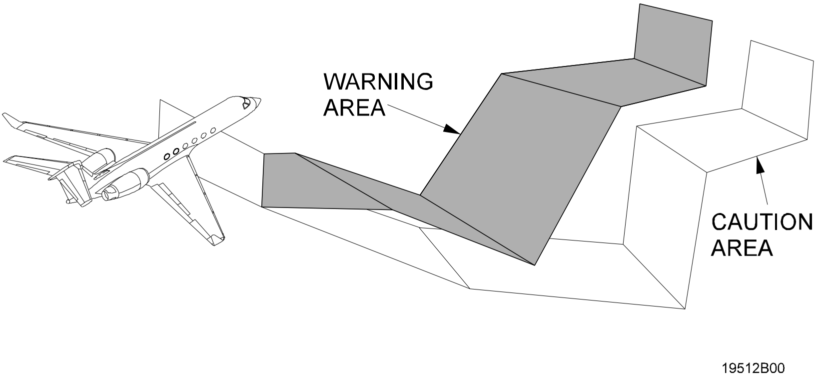


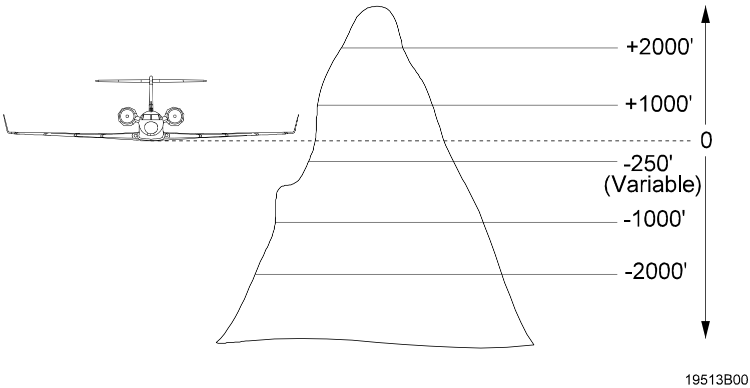
Windshear Alerting
Envelope - Mode 7
Figure 17



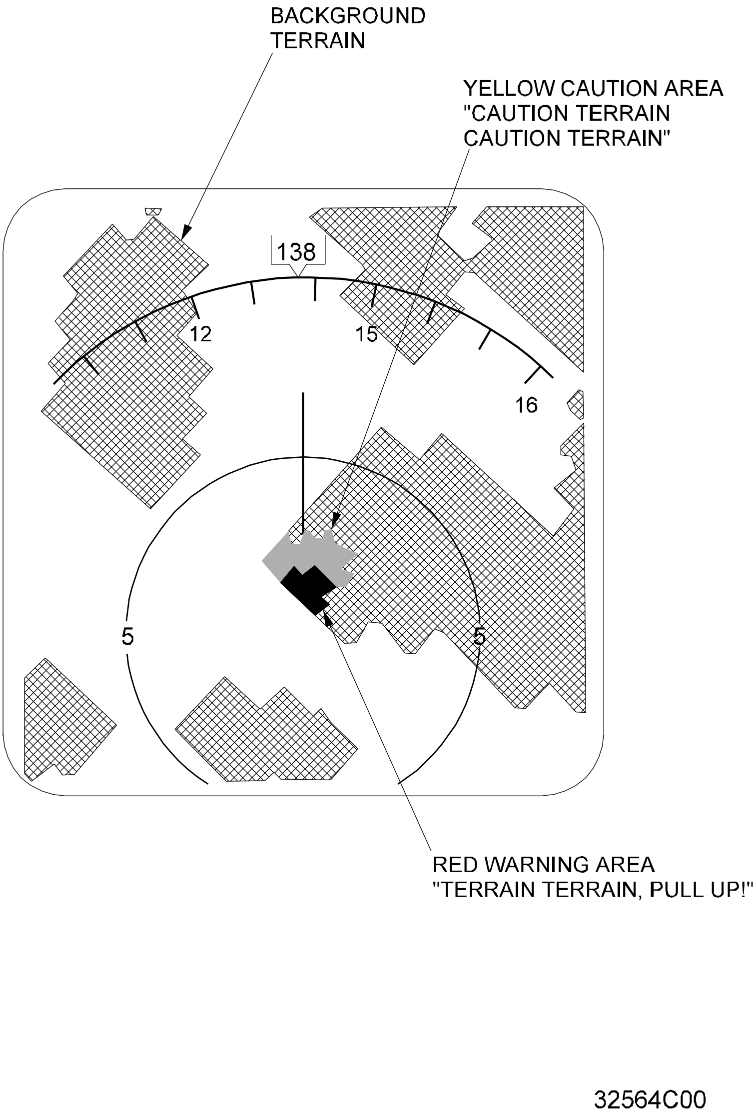
Terrain Clearance Floor Envelope
Figure 18



Forward Looking Terrain Alerting Areas
Figure 19



Terrain Awareness Display Altitudes
Figure 20



Terrain Awareness Display
Figure 21

2A-34-60: Traffic / Aircraft Alert and Collision Avoidance System (TCAS / ACAS)

1. General Description:

The Honeywell Traffic / Aircraft Alert and Collision Avoidance System (TCAS-2000 / ACAS) is production installed on GIV Serial Number (SN) 1390 and subsequent. Airplanes SN 1390 to 1433 were originally programmed with TCAS software version 6.04 and SN 1434 and subsequent with ACAS software version 7.0. Software version 7.0 is required for operation within European airspace. It is expected that all TCAS equipped airplanes will be equipped with version 7.0. This system description presumes the installation of software version 7.0.

TCAS/ACAS provides the flight crew with notifications of the presence of other transponder equipped traffic in the vicinity that may present a collision hazard. TCAS/ACAS can track up to fifty (50) airplanes simultaneously. The system provides aural alerts to the presence of traffic, visual plots on cockpit displays of the relative location of other airplanes, and aural and visual cues for evasive maneuvers if collision is imminent. If both converging airplanes are equipped with TCAS/ACAS and Mode S transponders, the systems mutually coordinate evasive maneuvers to ensure diverging flight paths.

The system functions as a low-powered airborne equivalent of the ground-based ATC system, interrogating and receiving replies from the transponders of other airplanes. Transponder replies are used to plot the position of traffic, and for traffic with Mode C and Mode S transponders, the altitude of the target is also determined.

TCAS / ACAS consists of the following subsystems / components as shown in the simplified block diagram in Figure 22:

- Two antennas, one on the top and one on the bottom of the airplane, each located on airplane centerline forward of the main cabin door.
- System computer
- Cockpit displays
- Radio Tuning Unit (RTU) and Mode S ATC Transponder

The directional antennas interrogate and receive the transponder signals of other airplanes and also continuously transmit a high speed data code (squitter) to TCAS / ACAS equipped airplanes. Transponder responses from other airplanes are processed in the TCAS / ACAS computer to determine relative closure vectors for traffic within the defined alerting envelope. The alerting envelope has altitude limit values that may be modified with selections made on the RTU.

The TCAS / ACAS computer communicates traffic information to the Display Controllers (DCs) and Symbol Generators (SGs) for subsequent display on the cockpit EFIS units. The TCAS / ACAS display is automatically selected to the NAV display on airplane power up. The flight crew may deselect the TCAS / ACAS display by using the Line Select Keys (LSKs) on the MAP menu of the DCs. The TCAS / ACAS traffic display may also be shown on the CAS screen (DU #4) by selecting TCAS / ACAS on the SYSTEMS menu on either DC. The TCAS / ACAS traffic display will automatically “pop up” on DU #4 whenever conflicting traffic penetrates the TCAS / ACAS alert envelope. Although the system tracks up to fifty (50) airplanes, only fifteen (15) airplane targets may be displayed at one time due to the limitations of the SGs.

The TCAS / ACAS computer has an internal synthetic voice component that annunciates the presence of traffic over cockpit speakers and interphone. If the

computer determines that an airplane presents an imminent collision hazard, the synthetic voice directs the flight crew to make changes in the vertical speed of the airplane to avoid the conflict. Any evasive maneuver directed by the TCAS / ACAS computer is shown on the Primary Flight Display (PFD) in a cue format that guides the flight crew to a safe separation distance.

NOTE:

TCAS / ACAS displays and alert messages are tools to increase flight crew situational awareness. Other tools include windshear and other EGPWS alerts. Displays and annunciations are prioritized so that any Windshear alert will override a EGPWS or TCAS / ACAS alert, and any EGPWS alert will override a TCAS / ACAS alert. In instances where TCAS / ACAS alerts are overridden, only Traffic Advisories are displayed and aural messages are temporarily inhibited.

2. Subsystems, Units and Components:

A. TCAS / ACAS Antennas:

Two antennas (upper and lower) antennas are installed. The antennas (shown in Figure 27) are directional receivers and omni-directional transmitters operating in the L-Band of the radio spectrum. A signal strength algorithm in the computer determines which antenna is used for best system operation. Using the directional features of the antennas and the elapsed time between interrogation and reply, the system computer determines bearing and distance to the other airplane transponder. If the received transponder signal is from an airplane with Mode C or Mode S, the altitude encoded in the signal is used to locate the airplane in three dimensions. Since TCAS / ACAS transmits and receives on the same frequency as the ATC system, a pulse suppression circuit is incorporated to prevent the TCAS / ACAS antennas from transmitting simultaneously with the onboard transponder.

B. TCAS / ACAS Computer:

The TCAS / ACAS computer is installed in the right radio rack, and is powered by ϕC of the Left Main 115V AC bus. The computer uses transponder replies received from other airplanes to monitor flight path tracks and determine potential conflicts. Conflicts are detected by comparing airplane range versus range closure rate and airplane altitude versus altitude closure rate. TCAS / ACAS can exchange data with other airplanes having Mode S transponders at a range of forty nautical miles (40 NM) and can detect (but not communicate with) Mode S airplane targets up to one hundred twenty miles (120 NM). For airplane targets with Mode A or Mode C transponders, the range is twenty nautical miles (20 NM).

A TCAS / ACAS alerting envelope is formulated based upon the amount of time available to the flight crew for evasive maneuvers at computed closure rates. The alerting envelope is a three-dimensional space surrounding the airplane that varies in size and sensitivity with altitude. Sensitivity levels determine the alarm time, size of the protected area and the vertical threshold for alerts. At higher altitudes the sensitivity level is expanded to

provide a larger protected area since traffic density is lower (and travelling at higher speeds). A protected area is defined by closure rate to a Closest Point of Approach (CPA). The protected area is expanded at close range to compensate for low closure rates, for example when two airplanes are on similar flight tracks at similar speeds, but the tracks differ sufficiently for the airplane tracks to converge at some point at a low closure rate. The vertical threshold of the protected area is expanded slightly at higher altitudes corresponding with higher sensitivity levels. TCAS / ACAS software version 7.0 is compliant with RVSM (Reduced Vertical Separation Minimums), therefore the envelope size is predicated on vertical separations of 1,000 feet below FL 420 and vertical separation of 2,000 feet above FL 420. For more specific information, consult the Honeywell Traffic Alert and Collision Avoidance System (TCAS) Pilot's Guide, publication number C28-3841-005-00, dated September 1999 (or later approved revision).

If TCAS / ACAS determines that another airplane's track will close within the defined caution envelope, a Traffic Advisory in the format of an aural message of "TRAFFIC, TRAFFIC" is annunciated over cockpit speakers and interphones. Concurrently with the aural alert, the TCAS / ACAS display will automatically "pop up" on DU #4 (the systems CAS display) with conflicting traffic represented as a solid amber circle on the TCAS / ACAS display. If traffic continues to close to within the warning envelope, the traffic is displayed as a solid red square on the TCAS / ACAS display accompanied by an aural message instructing the flight crew to take evasive action. Visual cues are presented on the Primary Flight Display (PFD) directing a change or restriction in vertical speed to avoid collision. The aural instructions and PFD cues are termed Resolution Alerts (RAs). After completion of the avoidance maneuver and traffic separation distances increase beyond the alerting envelope, an aural message "CLEAR OF TRAFFIC" is sounded, signalling the flight crew that they may return to the previously assigned flight parameters. Traffic Advisories and Resolution Alerts with accompanying display characteristics are discussed in the Cockpit Displays and Aural Messages section below.

WARNING

DO NOT RELY SOLELY ON TCAS / ACAS OR AIR TRAFFIC CONTROL FOR COLLISION AVOIDANCE:

- **TCAS / ACAS CANNOT DETECT AIRPLANES WITH INOPERATIVE OR NON-ICAO COMPLIANT TRANSPONDERS**
- **TCAS / ACAS REQUIRES A VALID ON-BOARD MODE S TRANSPONDER, A VALID BAROMETRIC ALTIMETER SOURCE AND A VALID RADIO ALTIMETER SOURCE**
- **TCAS / ACAS CAN PROVIDE RESOLUTION ALERTS ONLY FOR AIRPLANES WITH OPERATING ICAO COMPLIANT TRANSPONDERS WITH ALTITUDE REPORTING FUNCTIONS**
- **FOR AIRPLANES WITH ICAO TRANSPONDERS**

WITHOUT ALTITUDE REPORTING FUNCTIONS, TCAS / ACAS CAN ONLY PROVIDE TRAFFIC ADVISORIES

WARNING

TCAS / ACAS MAY NOT ALWAYS INHIBIT RESOLUTION ALERT MANEUVER COMMANDS IN FLIGHT REGIMES THAT MAY SIGNIFICANTLY REDUCE STALL MARGINS. EXAMPLES ARE:

- **BANK ANGLES EXCEEDING 15°**
- **ENGINE OUT (CREW SHOULD SELECT TRAFFIC ADVISORIES ONLY ON DISPLAYS)**
- **ABNORMAL CONFIGURATIONS - SUCH AS GEAR UN-RETRACTED WHICH WOULD LIMIT AIRPLANE PERFORMANCE**
- **OPERATION AT TEMPERATURES BEYOND ISA STANDARDS - ± 27.8°C (50°F)**
- **SPEEDS BELOW NORMAL OPERATING SPEEDS**
- **AT BUFFET MARGINS LESS THAN 0.3 “G”**
- **DURING A TCAS / ACAS TO TCAS / ACAS SENSE REVERSAL**

C. Cockpit Displays and Aural Messages:

The TCAS / ACAS display on Display Unit (DU) #4 (CAS) is a map view with a range of five (5) miles. At the center of the display is an airplane icon surrounded by a ring of twelve (12) dots at a range of two (2) miles. The dots represent clock positions and are shown as an aid in sighting detected traffic. See the illustration in Figure 23.

When selected to the cockpit NAV displays, TCAS / ACAS is in MAP mode and the range may by adjusted with the weather radar range controls (on the DCs). A typical display is shown in Figure 24. If TCAS / ACAS is tracking a target that is outside of the selected display range, the target will be shown at the perimeter of the display with only half (½) of the appropriate symbol visible. Although the maximum traffic detection range for TCAS / ACAS is 120 NM in front of the airplane and 15 NM behind the airplane, a smaller range setting offers better traffic discrimination in high density airspace.

TCAS / ACAS symbology is the same on both the DU #4 and NAV displays. Traffic is represented in different icon shapes and colors corresponding to the potential threat of collision:

- - a red square represents a Resolution Alert (RA) for conflicting traffic that poses a collision danger within 15 to 35 seconds
- - an amber circle represents a Traffic Alert (TA) for conflicting traffic that poses a collision danger within 20 to 48 seconds
- - a cyan (blue) diamond represents proximate traffic that is within ± 1200 feet of airplane altitude, but whose projected track does not pose a collision danger

- - a hollow cyan diamond represents other traffic more than ± 1200 feet from airplane altitude that does not hazard the airplane

When the detected traffic is equipped with a functioning Mode C or Mode S transponder, a two digit display of the altitude of the traffic in hundreds (100's) of feet accompanies the symbol. If the traffic is above airplane altitude, the two digits are shown above the symbol and preceded by a plus (+) sign, and if the traffic is below the airplane the digits are placed below the symbol and preceded by a minus (−) sign. For example, $^{+13}$ represents proximate traffic 1300 feet above the airplane, while $_{-08}$symbolizes proximate traffic 800 feet below the airplane. If the detected traffic is climbing or descending more than 500 feet per minute (FPM), an arrow is placed to the right of the symbol pointing in the direction of altitude change. Examples are $\downarrow^{+13}$ and $\uparrow_{-08}$.

If TCAS / ACAS is unable to determine the bearing of a target, the appropriate color coded symbol will be shown at the lower center of the display with altitude information if available. Failure to determine target bearing is most likely due to high bank angles masking antenna functions. Bearing information is usually available as soon as bank angle is moderated.

The flight crew may choose to have the traffic altitude displayed in absolute altitude rather than as a relative altitude difference from the airplane. If absolute altitude is selected, the traffic symbol is accompanied by the altitude readout reported by the ATC transponder. Absolute altitude reverts to relative altitude after 10 seconds.

The altitude range for TCAS / ACAS traffic surveillance is normally limited to ±2700 feet of airplane altitude (NORM on the RTU TCAS / ACAS control page). The altitude envelope may be expanded above and below the normal range with selections on the RTU. If ABOVE is selected, the envelope is set between −2700 feet and +7000 feet. If BELOW is selected, the envelope is set between +2700 feet and −7000 feet.

If the closest point of approach of any traffic is near enough to generate a RA, avoidance maneuver cues will automatically be displayed on both PFDs accompanied by aural instructions. An RA as presented on the PFD is shown in Figure 25. The Flight Director command bars are removed from view and the PFD will display climb or descent rates to avoid in the format of a red outlined trapezoid above, below or above and below the airplane symbol on the PFDs. The trapezoid enclosed areas represent vertical speeds that lead to a potential collision. A fly-to target in the form of a rectangular box outlined in green is displayed representing the desired airplane vertical speed for collision avoidance. The airplane should be maneuvered so that the airplane symbol on the PFD is within the green fly-to box. A RA that does not require modification of present vertical speed is termed a “preventative” RA, and in this case a green fly-to box is not shown, only the red vertical speed avoidance cues are presented.

The airplane symbol on the PFD is normally yellow, however during a RA display, the airplane symbol will be red if the airplane is flown in the vertical speed area(s) outlined in the red trapezoid(s), and colored green if the airplane is flown within the vertical speeds defined by the green fly-to box.

The RA vertical speed commands are programmed to be within the airplane performance capability and should usually only require altitude

changes of 300 to 500 feet and loads of ¼ ″G″ or less.

The display of RA guidance on the PFD and the MAP mode depiction of RA traffic on the NAV / CAS displays are accompanied with aural commands to emphasize the evasive maneuver. The TCAS / ACAS aural commands are tabulated below:

Aural Command	Meaning
“MONITOR VERTICAL SPEED - MONITOR VERTICAL SPEED”	Keep vertical speed outside of the ranges defined by the red trapezoid(s) on the PFD.
“CLIMB - CLIMB”	Promptly establish a rate of climb of 1500 FPM or more.
“CLIMB, CROSSING CLIMB - CLIMB, CROSSING CLIMB”	Promptly establish a rate of climb of 1500 FPM or more for a climb that will cross the flight path of conflicting traffic.
“DESCEND - DESCEND”	Promptly establish a rate of descent of 1500 FPM or more.
“DESCEND, CROSSING DESCEND - DESCEND, CROSSING DESCEND”	Promptly establish a rate of descent of 1500 FPM or more for a descent that will cross the flight path of conflicting traffic.
“REDUCE DESCENT - REDUCE DESCENT”	Promptly reduce descent rate to match the rate shown in the green outlined rectangle on the PFD.
“REDUCE CLIMB - REDUCE CLIMB”	Promptly reduce climb rate to match the rate shown in the green outlined rectangle on the PFD.
“CLIMB, CLIMB NOW - CLIMB, CLIMB NOW”	(Following a DESCEND advisory when circumstances require a reversal of direction.) Promptly establish a rate of climb of 1500 FPM or more.
“DESCEND, DESCEND NOW - DESCEND, DESCEND NOW”	(Following a CLIMB advisory when circumstances require a reversal of direction.) Promptly establish a rate of descent of 1500 FPM or more.
“INCREASE CLIMB - INCREASE CLIMB”	Promptly increase rate of climb to 2500 FPM or more.
“INCREASE DESCENT - INCREASE DESCENT”	Promptly increase rate of descent to 2500 FPM or more.
“ADJUST VERTICAL SPEED, ADJUST”	Adjust vertical speed to the range displayed on the PFD.
“MAINTAIN VERTICAL SPEED, MAINTAIN”	Maintain current climb or descent rate as shown on PFD to ensure traffic separation.
“MAINTAIN VERTICAL SPEED, CROSSING MAINTAIN”	Maintain current climb or descent rate as shown on PFD to ensure traffic separation - maneuver requires crossing flight path of conflicting traffic.
“CLEAR OF CONFLICT”	Promptly return to and / or maintain last assigned flight profile.

NOTE:

TCAS / ACAS visual and aural alerts are restricted in certain areas of the airplane operating envelope:

- No CLIMB commands are issued when the airplane cannot maintain a climb rate of 1,500 FPM.
- No INCREASE CLIMB commands are issued when the airplane cannot achieve a climb rate of 2,500 FPM.
- No INCREASE DESCENT commands are issued at altitudes less than 1,450 feet AGL (radio altitude).
- No DESCEND commands are issued when climbing at altitudes less than 1,200 feet AGL or descending at less than 1,000 feet AGL (radio altitude).
- No Resolution Advisories are issued when climbing at altitudes less than 1,100 feet AGL or descending at less than 900 feet AGL (radio altitude).
- No Aural Traffic Advisories are issued at altitudes less than 500 feet AGL (radio altitude).
- No Visual Traffic Advisories are issued for traffic with a transponder altitude readout of less than 380 feet AGL.
- In high density traffic areas, TCAS / ACAS reduces transponder interrogation power in order to limit interference with the ATC system. As a result, some targets at the outer perimeter of system surveillance range will not be shown on cockpit displays.

D. Radio Tuning Unit (RTU) and Mode S ATC Transponder:

Two Collins RTU-4220 Radio Tuning Units (RTUs) are installed in the cockpit center console, one for each crew position. TCAS / ACAS selections are made by pressing the Line Select Key (LSK) adjacent to the TCAS / ACAS section of the main display page. Pressing the LSK twice will access the TCAS / ACAS sub-display. See Figure 26 for an illustration of the TCAS / ACAS sub-display on the RTU. On the sub-display page, menu items may be selected with LSK entries. Available options are:

- Mode selections for Traffic Advisories and Resolution Alerts (TA / RA), Traffic Advisories only (T / A) or Standby (STBY)
- Display of absolute altitude (ABS) or relative altitude (REL) for TCAS / ACAS targets with Mode C or Mode S transponders (the ABS reverts to REL after 10 seconds)
- TRAFFIC selection to display other traffic that is determined not to be a collision hazard
- Selection of envelope altitude parameters for monitoring TCAS / ACAS traffic. ABOVE sets the envelope from 2,700 feet below the airplane to 7,000 feet above, BELOW sets monitoring between 2,700 feet above to 7,000 feet below the airplane, and NORMAL sets the envelope between 2,700 feet above and 2,700 feet below the airplane.

- TEST will initiate an eight (8) second system self-test that may be accomplished in the air or on the ground. If the airplane is on the ground and the TEST LSK is pushed and held in, a longer maintenance test will be initiated. Either the short or long test will also check the ATC transponder system.

Dual Collins TDR-94D Mode S transponders are installed in the nose of the airplane within the radome. Two transponder antennas are located on the bottom of the fuselage as shown in Figure 27. The transponders are controlled with LSK selections (on the ATC main display page) and the rotary knob on the RTU. Only one transponder is active at a time. LSK selections designate the active transponder and assigned ATC codes are entered with the rotary knob. An IDENT button on the upper right of the RTU is used to respond to identification requests from ATC. LSKs may also be used to turn off the altitude reporting feature of the transponder, and initiate a system test for both the transponders and TCAS / ACAS system from the RTU ATC main display page.

Annunciations presented on the ATC main display page include:

- RPLY shown in cyan (blue) when the transponder is replying to an interrogation signal
- XPDR FAIL shown in amber (yellow) when the selected active transponder has failed
- TYPE S is shown identifying that the transponder is compatible for TCAS / ACAS operation
- ID is displayed when the IDENT button has been pushed

3. Controls and Indications:

A. Circuit Breakers (CBs):

Circuit Breaker Name	CB Panel	Location	Power Source
TCAS	CP	D - 11	L MAIN 115V AC ϕC
XPNDR #1	CP	H - 11	EMER 28V DC 1A
XPNDR #2	CP	I - 11	R MAIN 28V DC

B. Advisory (Blue) CAS Messages:

CAS Message	Cause or Meaning
TCAS FAIL	TCAS / ACAS system has failed

C. TCAS / ACAS Self-Test:

Prior to initiating the eight (8) second self-test, turn on all displays with the NAV displays in MAP mode, select TA ONLY or TA / RA on the TCAS main page of the RTU display, select TCAS on the systems display of DU #4 (CAS) and select ATC transponder #1 altitude reporting ON with the transponder in standby.

Press the ATC 1 transponder TEST button until the aural annunciation "TCAS TEST" is heard. The following indications are shown on the NAV and / or CAS displays:

- A solid blue diamond ↓ (proximate traffic) positioned at one o'clock with an arrow pointing down and a relative altitude below the

symbol of -10 (indicates 1,000 feet below TCAS / ACAS test airplane)

- A hollow blue diamond (other traffic) positioned at eleven o'clock with no arrow and a relative altitude above the symbol of +10 (indicates 1,000 feet above TCAS / ACAS test airplane)
- A solid red square (RA) at three o'clock with no arrow and a relative altitude above the symbol of +02 (indicates 200 feet above TCAS / ACAS test airplane)
- A solid amber circle (TA) positioned at nine o'clock with an upward pointing arrow and a relative altitude of - 02 (indicates 200 feet below TCAS / ACAS test airplane)
- TCAS FAIL message shown in blue on the CAS display for the duration of the test
- After eight seconds, an aural annunciation of "TCAS TEST PASS" or "TCAS TEST FAIL"

NOTE:

ATC transponder #2 may be used for the self-test if transponder #1 is turned OFF.

CAUTION

ACTIVATING THE SELF-TEST WHILE AIRBORNE WILL BLOCK ALL TRAFFIC INFORMATION DISPLAY AND ANNUNCIATIONS FOR THE EIGHT SECOND DURATION OF THE TEST.

4. Limitations:

A. Flight Manual Limitations:

(1) **Pilot's Manuals:**

The Honeywell Traffic Alert and Collision Avoidance System (TCAS) Pilot's Guide, publication number C28-3841-005-00, dated September 1999 (or later approved revision) shall be immediately available to the pilots. The Honeywell SPZ-8400 Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV, publication number A28-1146-097-02, Revision 2, dated October 1999 (or later approved revision) shall be immediately available to the pilots. This applies to airplanes SN 1390 and subsequent.

(2) **TCAS Operating Constraints:**

With 6.04A software installed, all Resolution Advisory (RA) and Traffic Advisory (TA) aural messages are inhibited at a radio altitude less than 1,100 feet climbing and 900 feet descending.

With 7.0 software installed (SN 1434 and subsequent), all RA and TA aural messages are inhibited at a radio altitude less than 500 feet ±100 feet.

(3) **Clearance:**

The pilot is authorized to deviate from ATC to the extent necessary to comply with a Resolution Advisory (RA).

(4) **Traffic Advisories:**

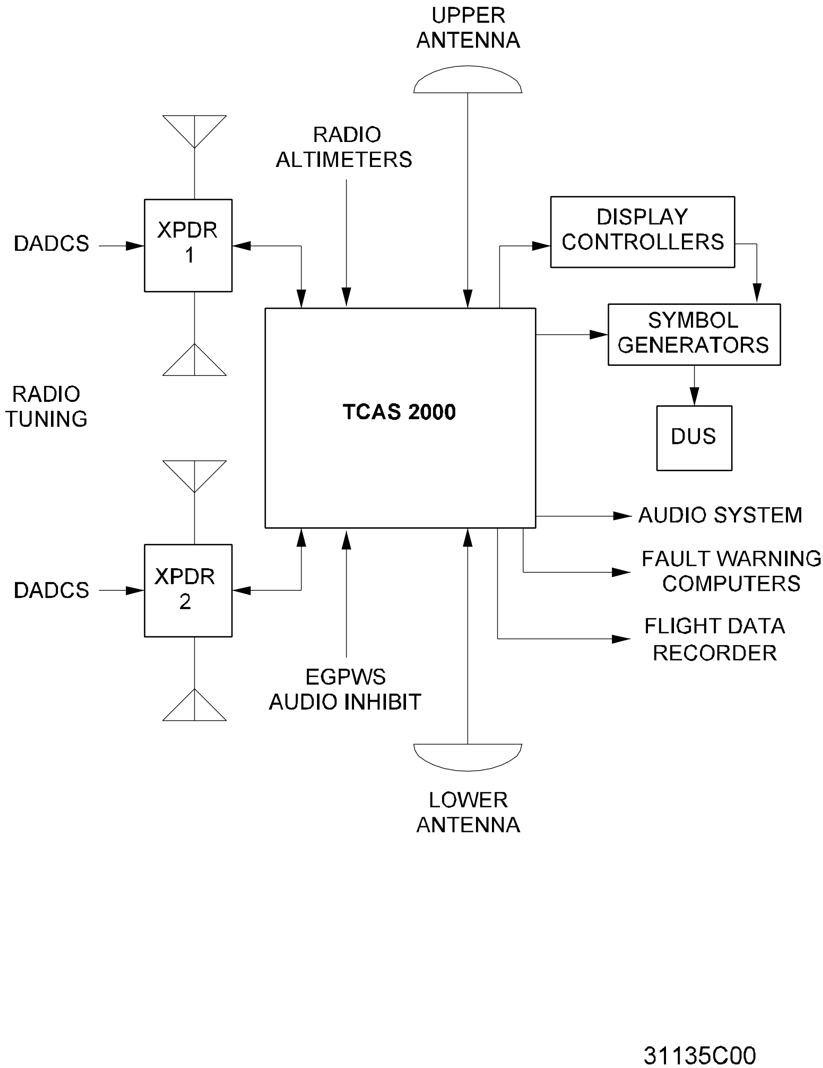
The pilot must not initiate evasive maneuvers based solely on information from a Traffic Advisory (TA). Traffic Advisory information should be used only as an aid to visual acquisition of traffic.

(5) **Resolution Advisories:**

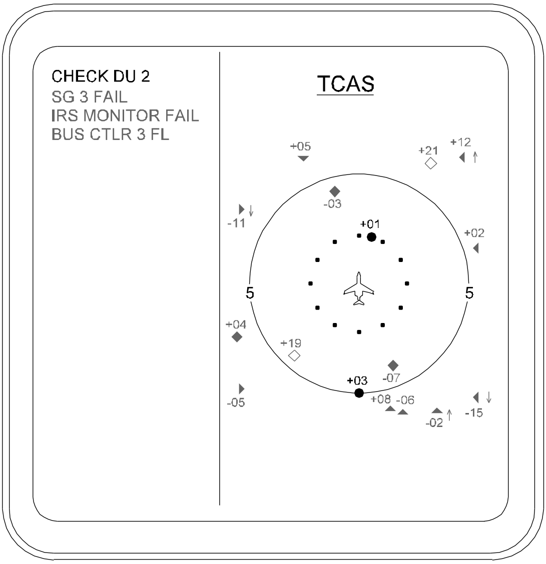
Compliance with TCAS Resolution Advisories (RA) is required unless the pilot considers it unsafe to do so. Maneuvers which are in the opposite direction of an RA are extremely hazardous and are prohibited unless it is visually determined to be the only means to assure safe separation.

(6) **Clear Of Conflict:**

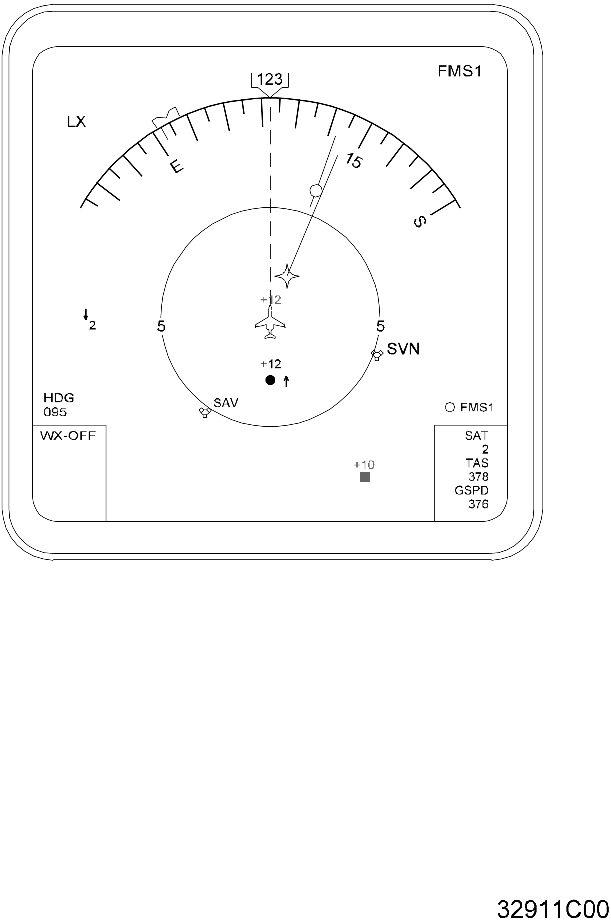
Prompt return to the ATC cleared altitude must be accomplished when “CLEAR OF CONFLICT” is announced.



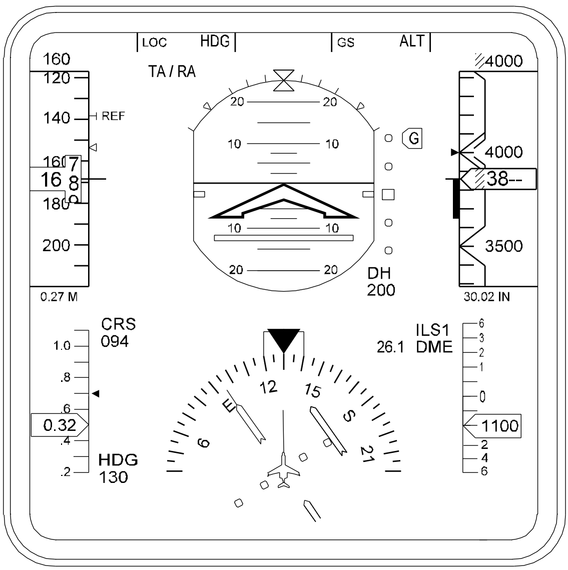
TCAS / ACAS System Simplified Block Diagram
Figure 22



TCAS / ACAS Display on DU #4
Figure 23



TCAS / ACAS Display in MAP Mode
Figure 24



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TCAS / ACAS RA Display on the PFD
Figure 25

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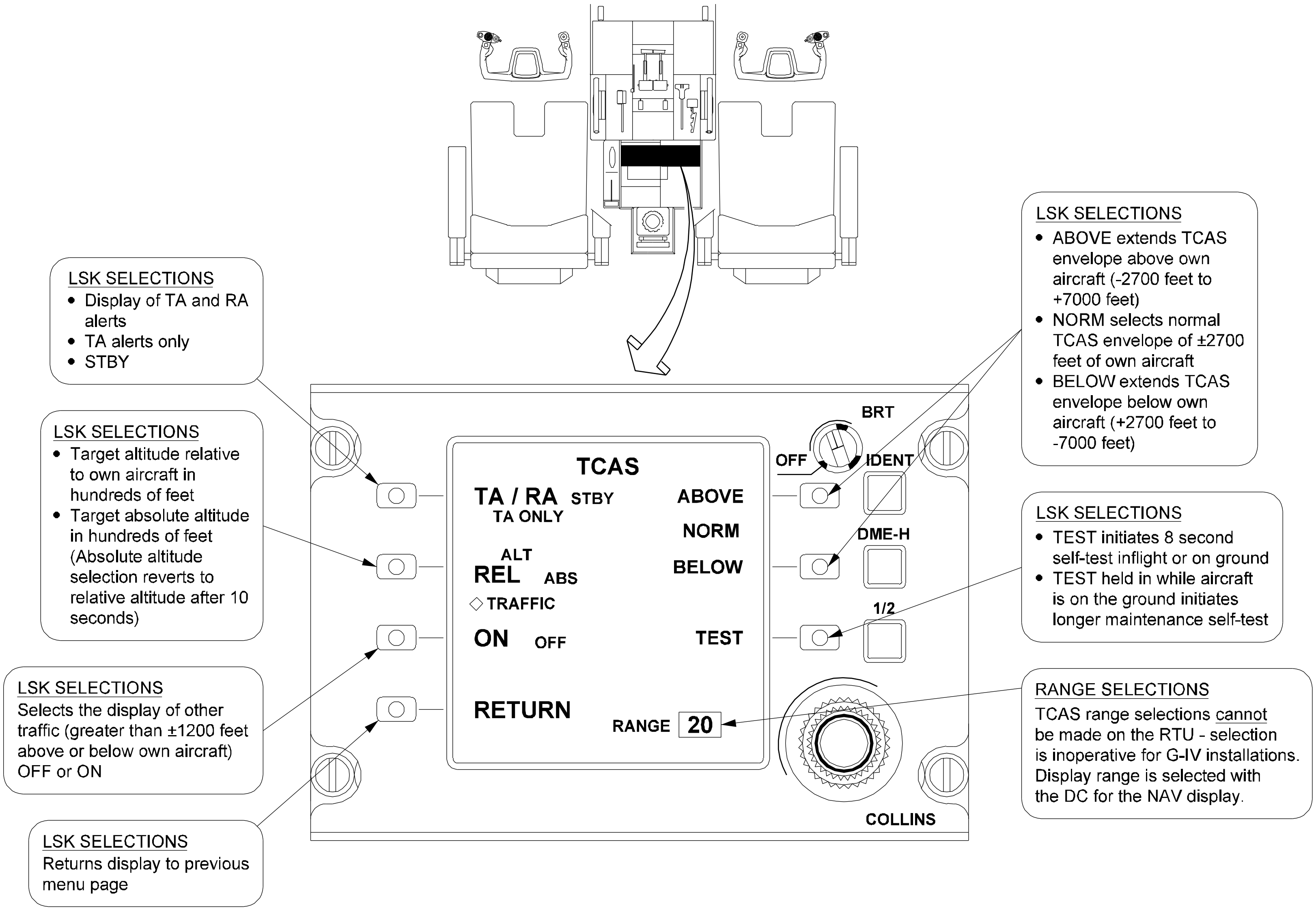
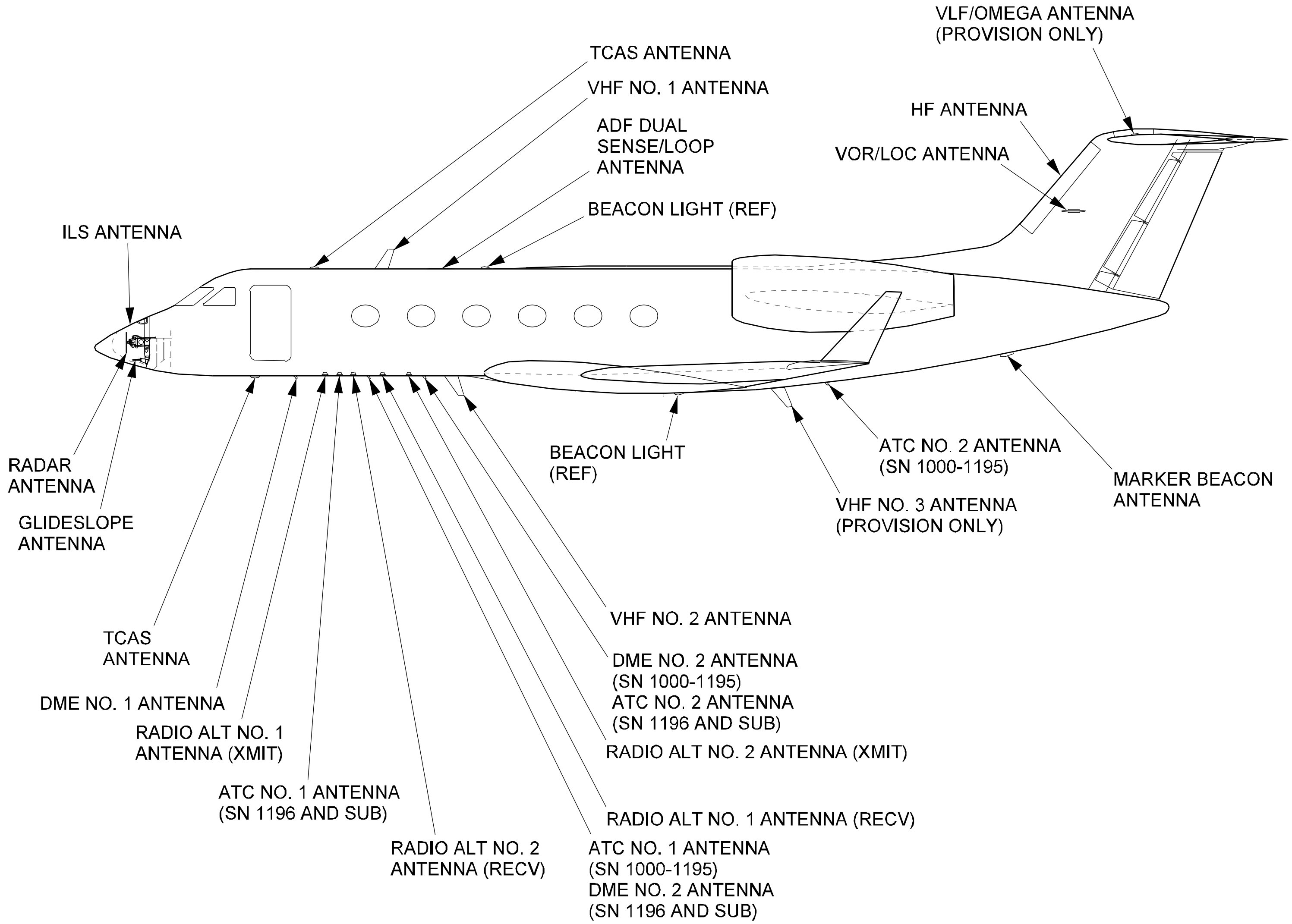


Radio Tuning Unit (RTU)
Figure 26



TCAS / ACAS
Transponder Antenna
Locations
Figure 27

2A-35-10: General

The Gulfstream IV is designed to operate at altitudes up to and including 45,000 feet. Both the airplane structure and pressurization system are designed using fail-safe criteria in order to minimize any possibility of exposing the crew and passengers to these critical altitudes. Should the need arise, however, high-pressure gaseous oxygen systems are installed to provide oxygen to all occupants. There are two types of oxygen systems installed on the GIV airplanes: the crew and passenger oxygen system and the portable oxygen system. On airplanes 1500 and subsequent, however, there are also production provisions for a therapeutic oxygen system.

In accordance with Federal Aviation Administration regulations, there must be at least one portable oxygen system (commonly referred to as a “walkaround” cylinder or bottle) aboard the airplane for flight crew use while disconnected from the airplane system. The exact type of system and its location are selected by the operator and installed by the outfitting agency. Specific information describing the portable oxygen system would thus be supplied to the operator by the outfitting agency. Provisions exist to place this data in Chapter 2C, Outfitted Systems, of this manual.

2A-35-20: Crew and Passenger Oxygen Systems

1. **General Description:**

   Two types of production standard gaseous oxygen systems are installed in Gulfstream IV airplanes. Airplanes Serial Number (SN) 1000 through 1289 have only for a single-cylinder crew oxygen system installed. For these airplanes, the outfitting agency selected by the operator installs the passenger oxygen system and performs modifications, if any, to the crew oxygen system. Specific information describing the outfitted oxygen system would thus be supplied to the operator by the outfitting agency. Provisions exist to place this data in Chapter 2C, Outfitted Systems, of this manual.

   Airplanes SN 1290 and subsequent have both the crew and passenger oxygen systems installed. Both types of configurations are discussed in this description.

   The crew oxygen system is a pressure-demand type system, while the passenger oxygen system is a continuous flow type system. When in use, they provide an emergency oxygen source to the flight crew, jump seat observer and cabin passengers. The following units and components compose the system:

   - Oxygen Cylinder and Regulator Assembly
   - Oxygen Servicing Panel
   - Oxygen System Control Panel
   - Passenger Oxygen Control Panel
   - Crew Mask/Regulator Assembly

   All GIV oxygen systems use only aviator’s breathing oxygen conforming to MIL-O-27210. Use of medical oxygen is strictly forbidden due to its high moisture content. This moisture could freeze in the supply lines, rendering the system inoperative.

2. **Description of Subsystems, Units, and Components:**

   A. **Oxygen Cylinder and Regulator Assembly:**

      (1) Airplanes SN 1000 through 1289:

(See Figure 1.)

The crew oxygen cylinder is constructed of a seamless aluminum liner with a filament overwrap of Kevlar. It has a gaseous capacity of 50 cubic feet (1,416 liters (1359 liters usuable)) and is normally pressurized to 1800 psi at 70° F (21° C). The observed pressure reading when fully serviced varies with temperature, however, and may fall in a range between 1,485 psi at 0° F (-18° C) to 2015 psi at 120° F (49° C). The crew oxygen cylinder is installed underneath the forward cabin floor, in the area approximately between the second set of cabin windows.

A regulator assembly is attached to the neck of the cylinder. The primary function of the regulator is to reduce high cylinder pressure to a nominal delivery pressure of approximately 70 psi. A manual ON/OFF knob is installed on the regulator to allow or inhibit oxygen flow directly from the cylinder. The knob is accessible from the cabin through an access panel cut into the floorboard.

In addition to controlling oxygen supply and its delivery pressure, the regulator assembly provides ports for the following connections:

(a) Oxygen Cylinder Direct Reading Gauge:

A direct reading gauge is installed on each regulator assembly in order to obtain oxygen cylinder pressure directly from the cylinder.

(b) Supply Line:

A supply line exiting the cylinder pressure regulator routes oxygen to the crew oxygen system control panel where flow can be controlled by the panel's ON/OFF toggle switch. When the toggle switch is selected ON, flow is routed to the three crew mask/regulator assemblies. A check valve is installed on the supply line to prevent system backflow.

A pressure switch is installed between the cylinder pressure regulator and check valve to monitor supply line pressure. If the switch senses a line pressure below approximately 50 psi, a discrete is sent to Data Acquisition Unit (DAU) #1. DAU #1 in turn causes the Crew Alerting System (CAS) to display an amber CREW OXYGEN OFF caution message. The purpose of this message is to caution the flight crew that the oxygen cylinder may have been manually shut off at the cylinder.

(c) Fill Line:

A fill line is installed to transfer oxygen from an external source connected to the oxygen servicing panel (left forward nose section) to the oxygen cylinder.

(d) Gauge Line:

A dedicated gauge line is installed to allow viewing of cylinder pressure. Leaving the regulator assembly, the gauge line connects to the direct reading gauges on the oxygen servicing panel (left forward nose section) and the crew oxygen system control panel (pilot's skirt panel). Flow restrictors are incorporated so that should a gauge line

rupture, the resulting oxygen loss rate would be limited to approximately one liter per minute.

(e) Overboard Discharge Line:

An overboard discharge line is installed to vent oxygen overboard through a port located on the left fuselage, forward of the wing leading edge. The cylinder regulator will allow overboard discharge under the following conditions:

- Regulated oxygen pressure exceeds 90 psi, activating the regulator's relief valve
- Oxygen cylinder pressure exceeds 2775 psi, rupturing the regulator's high pressure burst disk
- Oxygen temperature exceeds 225 (±5)° F, melting the regulator's high pressure burst disk

The overboard discharge port is covered by a overboard discharge disk. See Figure 3. In addition to preventing contaminants from entering the overboard discharge line, the disk serves as a visual indicator to the flight crew that an overboard discharge has occurred, by being blown out and off the discharge port. It is green in color and labeled OXY. H.P. RELIEF.

(2) Airplanes SN 1290 and Subsequent:

(See Figure 2.)

Each of the three oxygen cylinders (one crew and two passenger) is constructed as a seamless aluminum liner with a filament overwrap of Kevlar. Each cylinder is normally pressurized to 1800 psi at 70° F (21° C). The observed pressure reading when fully serviced varies with temperature, however, and may fall in a range between 1,485 psi at 0° F (-18° C) to 2015 psi at 120° F (49° C). All three cylinders are installed underneath the forward cabin floor. The two passenger cylinders each have a gaseous capacity of 115 cubic feet (3,257 liters (2900 liters usuable)) and are installed side by side in the area approximately between the first set of cabin windows. The crew cylinder has a gaseous capacity of 50 cubic feet (1,416 liters (1359 liters usuable)) and is installed just aft of the passenger cylinders. Provisions exist for an optional second crew cylinder to be installed beside the standard crew cylinder if selected by the operator during outfitting.

A regulator assembly is attached to the neck of each cylinder. The primary function of the regulator is to reduce high cylinder pressure to a nominal delivery pressure of approximately 70 psi. A manual ON/OFF knob is installed on each regulator to allow or inhibit oxygen flow directly from that cylinder. These knobs are accessible from the cabin through access panels cut into the floorboard.

In addition to controlling oxygen supply and its delivery pressure, each regulator assembly provides ports for the following connections:

(a) Oxygen Cylinder Direct Reading Gauge:

A direct reading gauge is installed on each oxygen cylinder

regulator in order to obtain oxygen cylinder pressure directly from the cylinder.

(b) Supply Lines:

A supply line exits each passenger cylinder pressure regulator, later joining to form a common passenger supply line. A check valve is installed on each passenger cylinder supply line (before joining occurs) to prevent system backflow. The common passenger supply line routes oxygen to the oxygen control panel where flow can be controlled by the panel's ON/OFF toggle switch. When the toggle switch is selected ON, flow is routed through a common supply line that is subsequently divided into two lines. One line delivers oxygen to the passenger oxygen control panel, where it may be further distributed to the left and right side passenger masks. The other line connects to the crew oxygen supply line downstream of the CREW OXYGEN ON/OFF toggle switch. This line enables the passenger oxygen system to deliver oxygen to the crew oxygen system. A check valve installed on this line prevents crew oxygen flow into the passenger oxygen system.

Provisions also exist for the passenger cylinder supply line to deliver oxygen to therapeutic oxygen outlets, when such optional outlets are installed.

A supply line exiting the crew cylinder pressure regulator routes oxygen to the oxygen control panel where flow can be controlled by the panel's ON/OFF toggle valve switch. When the switch is selected ON, flow is routed to the three crew mask/regulator assemblies. A check valve is installed on the supply line to prevent system backflow.

A pressure switch is installed between each cylinder pressure regulator and check valve to monitor supply line pressure. If the switch senses a line pressure below approximately 50 psi, a discrete is sent to DAU #1. DAU #1 in turn causes CAS to display an amber CREW OXYGEN OFF and/or PAX OXYGEN OFF caution message. The purpose of these messages is to caution the flight crew that the oxygen cylinder may have been manually shut off at the cylinder.

(c) Fill Lines:

Fill lines are installed to transfer oxygen from an external source connected to the oxygen servicing panel (left forward nose section) to the oxygen cylinders via the regulator assemblies. The passenger oxygen system fill line begins as a single line at the servicing panel. The line is then divided into a dedicated line for each passenger cylinder. The crew oxygen system uses a single fill line dedicated to the crew cylinder.

(d) Gauge Lines:

Dedicated gauge lines are installed to allow viewing of cylinder pressure. The crew cylinder gauge line begins as a single line leaving the regulator assembly. It is then divided

downstream into two lines: one line connecting to the direct reading gauge on the oxygen servicing panel (left forward nose section); the other line connecting to the direct reading gauge on the oxygen control panel (pilot's skirt panel).

The passenger cylinder gauge line begins as a single line leaving each regulator assembly. Both lines then join to form a common line. Further downstream, the single line is divided into two lines: one line connecting to the direct reading gauge on the oxygen servicing panel (left forward nose section); the other line connecting to the direct reading gauge on the oxygen control panel (pilot's skirt panel).

Flow restrictors are incorporated so that should a gauge line rupture, the resulting oxygen loss rate would be limited to approximately one liter per minute.

(e) Overboard Discharge Line:

Two overboard discharge lines (passenger and crew) are installed to vent oxygen overboard through ports located on the left fuselage, forward of the wing leading edge. The forward discharge port is used by the passenger oxygen system; the aft discharge port is used by the crew oxygen system. The cylinder regulator will allow overboard discharge under the following conditions:

- Regulated oxygen pressure exceeds 90 psi, activating the regulator's relief valve
- Oxygen cylinder pressure exceeds 2775 psi, rupturing the regulator's high pressure burst disk
- Oxygen temperature exceeds 225 (±5)° F, melting the regulator's high pressure burst disk

Each overboard discharge port is covered by a overboard discharge disk. See Figure 3. In addition to preventing contaminants from entering the overboard discharge line, the disk serves as a visual indicator to the flight crew that an overboard discharge has occurred, by being blown out and off the discharge port. It is green in color and labeled OXY. H.P. RELIEF.

B. Oxygen Servicing Panel:

(1) Airplanes SN 1000 through 1289:

(See Figure 4.)

The oxygen servicing panel is located under an access door on the left forward nose section. The panel consists of an external high pressure filler valve and a direct reading pressure gauge for the crew oxygen system. The filler valve, with chain and cap, is connected via a single fill line to the oxygen cylinder regulator and provides a point for servicing the system. The direct reading pressure gauge allows the crew oxygen system pressure to be checked at the panel.

(2) Airplanes SN 1290 and Subsequent:

(See Figure 5.)

The oxygen servicing panel is located under an access door on the left forward nose section. The panel consists of two external high pressure filler valves and two direct reading pressure gauges: the upper set for the passenger oxygen system; the lower set for the crew oxygen system.

The passenger oxygen system filler valve, with chain and cap, provides a point for servicing the system. It is connected to what begins as a single fill line. Downstream, however, this single line divides into two lines, one line dedicated to each passenger oxygen cylinder. The associated direct reading pressure gauge allows the passenger oxygen system pressure to be checked at the servicing panel. The gauge line is constructed in the same manner as the fill line, i.e., a single line at the gauge dividing into two lines, one line dedicated to each passenger oxygen cylinder.

The crew oxygen system filler valve, with chain and cap, is connected via a single fill line to the crew oxygen cylinder regulator and provides a point for servicing the system. The associated direct reading pressure gauge allows the crew oxygen system pressure to be checked at the servicing panel.

C. Oxygen System Control Panel:

(See Figure 6.)

(1) Airplanes SN 1000 through 1289:

The crew oxygen system control panel is located on the pilot's skirt panel. It contains an ON/OFF toggle valve switch that controls oxygen flow to the three crew mask/regulator assemblies. The control panel also contains a pressure gauge that displays crew oxygen system pressure on the oxygen gauge line. Lighting for the control panel is supplied by the cockpit lighting system.

(2) Airplanes SN 1290 and Subsequent:

The crew and passenger oxygen control panel is located on the pilot's forward instrument panel. It contains both crew and passenger pressure gauges, each displaying oxygen system pressure on the respective oxygen gauge line. The control panel also contains two ON/OFF toggle valve switches, each controlling oxygen flow to the respective oxygen systems. Lighting for the control panel is supplied by the cockpit lighting system.

D. Passenger Oxygen Control Panel (All GIV Aircraft):

(See Figure 7.)

The passenger oxygen control panel, located on the copilot's right console panel, controls and annunciates oxygen flow and displays system pressure to the passenger oxygen masks. (The type, quantity and location of passenger masks is selected by the aircraft owner and installed during aircraft outfitting.) Lighting for the control panel is supplied by the cockpit lighting system. The passenger oxygen control panel has the following components:

(1) OFF / AUTO / MAN Knob:

The OFF / AUTO / MAN knob controls the oxygen flow to the passenger oxygen masks. In the OFF position, oxygen flow to the

passenger oxygen masks is inhibited. The AUTO position allows the passenger oxygen control panel to deploy (drop) the passenger oxygen masks automatically when sensed cabin altitude reaches 13,000 ±500 feet. Oxygen flow to the passenger oxygen masks is then regulated based on cabin altitude. Manual selection to the MAN position deploys the passenger oxygen masks and provides a constant preset flow.

(2) PASS OXYGEN ON Annunciator:

When oxygen flow through the passenger oxygen control panel is sensed, a pressure switch inside the passenger oxygen control panel closes, causing a discrete to be sent to DAU #1. DAU #1 in turn causes the CAS to display an amber CABIN OXYGEN ON caution message. The pressure switch also causes the PASS OXYGEN ON annunciator to illuminate amber.

(3) OXYGEN SUPPLY PRESSURE Gauge:

The OXYGEN SUPPLY PRESSURE gauge displays oxygen system pressure available to the passenger oxygen masks.

(4) ALT TEST Port:

The ALT TEST port is used by maintenance personnel to check the automatic operation of the passenger oxygen control panel.

(5) Passenger Oxygen TEST Button (Airplanes 1457 and Subs):

Currently, there are production provisions for a PASS OXY test button on the TEST panel (copilot side of center console).

The passenger oxygen system is normally operated by **first** ensuring the OFF / AUTO / MAN knob is set to OFF on the passenger oxygen control panel. Next, passenger oxygen is selected ON at the oxygen system control panel. (This sequence ensures the passenger oxygen masks do not inadvertently deploy.) **After** passenger oxygen is selected ON, the OFF / AUTO / MAN knob is then set to AUTO. In this configuration, the passenger oxygen control panel will deploy the passenger oxygen masks automatically when sensed cabin altitude reaches 13,000 ±500 feet.

When the passenger oxygen masks are deployed, an amber CABIN OXYGEN ON will be displayed on CAS, along with the associated aural caution tone. In addition, the NO SMOKING signs in the cabin and lavatory areas will illuminate and the cabin chime will sound momentarily.

E. Crew Mask/Regulator Assembly (All GIV Aircraft):

(Descriptions and illustrations in this section are limited to the EROS MXP 300 crew mask/regulator. For further details on the EROS MXP 100 and MXP 300 crew mask/regulators, refer to EROS Operating and Maintenance Instructions, EROS document number 4NUT0045A, dated October 15, 1999 or later approved version. See Figure 8.)

The mask/regulators are normally kept in stowage compartments located on the pilot's side console, copilot's side console and left hand radio rack. The mask is removed from the stowage compartment by grasping the regulator with three fingers (middle, ring and small), while keeping the thumb and forefinger open, and withdrawing the mask. When the mask/regulator is removed from the stowage compartment, the open doors activate a shutoff valve, making oxygen available to the mask. The blinker

will display a yellow cross when oxygen is flowing.

In order to prevent inflating the mask harness before it is withdrawn from the stowage compartment, the inflation valve (red ears) is not depressed until after the mask is out of the stowage compartment. The inflation valve can then be depressed with the thumb and forefinger, inflating the harness. Releasing the inflation valve vents the harness to ambient, allowing the elastic in the harness to create a snug fit over the nose and mouth.

Each mask/regulator stowage unit contains a PRESS TO TEST AND RESET control lever (left door) and a flow indicator (blinker) that displays a yellow cross (+) for oxygen flow and black for no flow. The mask/regulator may be tested while in the stowed position as described at the end of this section.

The PRESS TO TEST AND RESET control lever allows leak testing of the regulator while stowed. When the control lever is depressed, the blinker will display a yellow cross and then return to black, indicating that the regulator is leak-tight. If the blinker remains yellow, a leak in the system should be assumed and investigated. The PRESS TO TEST AND RESET control lever also can be used to shut off oxygen flow to the mask/regulator and blinker in the event of a system failure. If the left door on the stowage compartment is first closed and then followed by depressing the PRESS TO TEST AND RESET control lever, oxygen flow will cease.

Each mask/regulator has an inflating harness, inflation valve (red ears), N-100% control lever, PRESS TO TEST/EMERGENCY control knob, microphone, hose with quick-disconnect fitting and a communication harness.

The N-100% control lever provides diluted oxygen, regulated by altitude and demand, while in the “N” (normal) position. As cabin altitude increases, an aneroid capsule in the mask/regulator expands to decrease ambient airflow the wearer breathes. This action increases the oxygen to ambient air ratio until the cabin altitude reaches 35,000 feet. At this cabin altitude, the aneroid capsule has completely shut off ambient airflow and the wearer is now breathing 100% oxygen. 100% oxygen flow is also supplied when the N-100% control lever is depressed to the “100%” position.

The PRESS TO TEST/EMERGENCY control knob has three functions. When momentarily depressed, it verifies the regulator valve will supply oxygen under constant pressure. In the NORMAL position, it allows the regulator’s aneroid capsule to regulate positive pressure to assist breathing at cabin altitudes between 36,000 feet and 45,000 feet. In the EMERGENCY position, 100% oxygen is supplied in a “positive pressure on demand” mode to assist breathing.

3. Controls and Indications:

For airplanes SN 1000 through 1289, see Figure 1, Figure 3, Figure 4 and Figure 6 through Figure 8. For airplanes SN 1290 and subsequent, see Figure 2, Figure 3 and Figure 5 through Figure 8.

A. Caution (Amber) CAS Messages:

CAS Message	Cause or Meaning
CABIN OXYGEN ON	Passenger oxygen system has been activated by either manual or automatic means.

CAS Message	Cause or Meaning
CREW OXYGEN OFF	Crew oxygen bottle is shut off at the bottle.
PAX OXYGEN OFF	Passenger oxygen bottle is shut off at the bottle.

B. Advisory (Blue) CAS Messages:

CAS Message	Cause or Meaning
AUX OXYGEN ON **(1)**	Auxiliary oxygen source is on.
MED OXYGEN ON **(1)**	Medical oxygen is on.
OXY SRVC DR OPEN **(1)**	Oxygen servicing panel access door is open.

NOTE(S):

(1) These are custom programmable messages activated as desired during outfitting.

C. Other Annunciations:

Indication	Cause or Meaning
Amber PAX OXYGEN ON light on passenger oxygen control panel.	Passenger oxygen system has been activated by either manual or automatic means.
Yellow cross (+) indicator (blinker) visible on crew mask/regulator assembly.	Oxygen is flowing to crew mask/ regulator assembly.

4. Limitations:

A. Flight Manual Limitations:

(1) Oxygen Departure Pressures (SN 1000 thru 1289):

The production-installed oxygen supply consists of one (1) 50 cubic ft. (1415 liter (1359 liters usable)) cylinder located below the forward cabin floor. The minimum oxygen supply shall be determined for each flight using Figure 9 as a guide.

(2) Oxygen Departure Pressures (SN 1290 and Subsequent):

The production-installed oxygen supply consists of one (1) 115 cubic ft. (3256 liter (2900 liters usable)) cylinder located below the forward cabin floor. The minimum oxygen supply shall be determined for each flight using Figure 10 as a guide.

B. Crew Mask/Regulator Preflight Check:

This procedure allows the flight crew the ability to perform a complete preflight check without having to remove the mask/regulator from its stowage compartment.

(1) Select the crew oxygen system toggle valve switch to ON.

(2) Check the crew oxygen system supply pressure.

(3) On the crew mask/regulator stowage compartment, perform the following based upon the type of crew mask/regulator installed:

(a) For the EROS MXP 100, depress the RESET TEST control lever on the LH door in the direction of the arrow. Oxygen is then supplied to the mask/regulator assembly.

(b) For the EROS MXP 300, depress the PRESS TO TEST AND RESET control lever on the LH door in the direction of the arrow. Oxygen is then supplied to the mask/regulator assembly.

(4) MXP 100 only: observe that a white band is visible at the top of the RESET TEST control lever (LH door).

(5) Observe that the flow indicator (blinker) momentarily displays a yellow cross (+), then returns to black. This indicates the regulator is leak-tight. If the blinker continuously displays the yellow cross, a leak in the system should be assumed and investigated.

(6) While holding the MXP 100 RESET TEST control lever (or MXP 300 PRESS TO TEST AND RESET control lever), depress and hold the red PRESS TO TEST button on the mask regulator for one (1) second while performing steps (a) and (b) below.

(a) Observe that the blinker momentarily displays a yellow cross, then returns to black. This indicates the regulator demand mechanism is working properly.

(b) While holding the MXP 100 RESET TEST control lever (or MXP 300 PRESS TO TEST AND RESET control lever) and red PRESS TO TEST button, microphone integrity must be checked by listening for oxygen flow noise through the communication set.

(7) Release the PRESS TO TEST button on the mask regulator.

(8) Release the MXP 100 RESET TEST control lever (or MXP 300 PRESS TO TEST AND RESET control lever).

(a) MXP 100 Only: observe the white band (top of RESET TEST control lever, LH door) disappears.

(b) Observe that oxygen flow ceases. Downstream pressure will be vented to ambient.

(9) Select the crew oxygen system toggle valve switch to OFF, if desired.

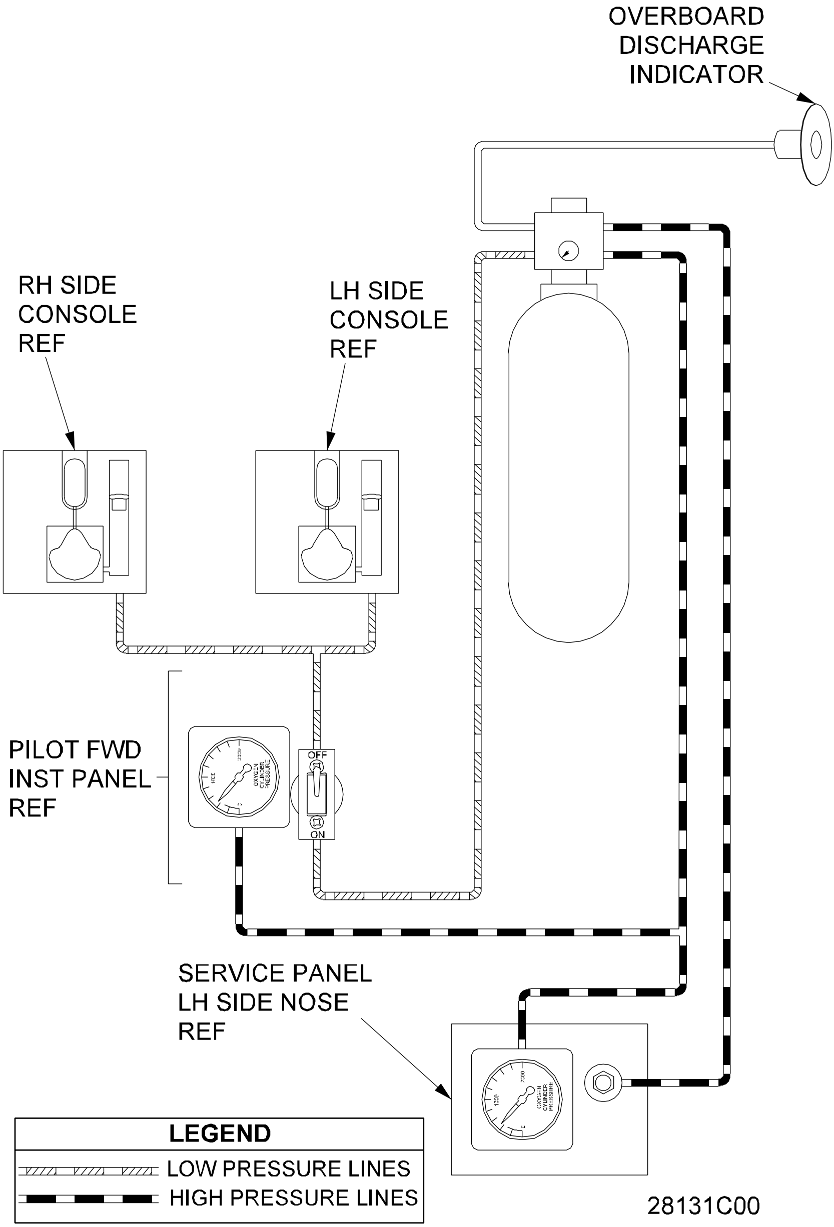
NOTE:

The MXP 100 RESET TEST control lever (or MXP 300 PRESS TO TEST AND RESET control lever) may also be used for shutting off the supply to the mask/regulator assembly in the event of a failure. Prior to depressing the MXP 100 RESET TEST control lever (or MXP 300 PRESS TO TEST AND RESET control lever), close the door.

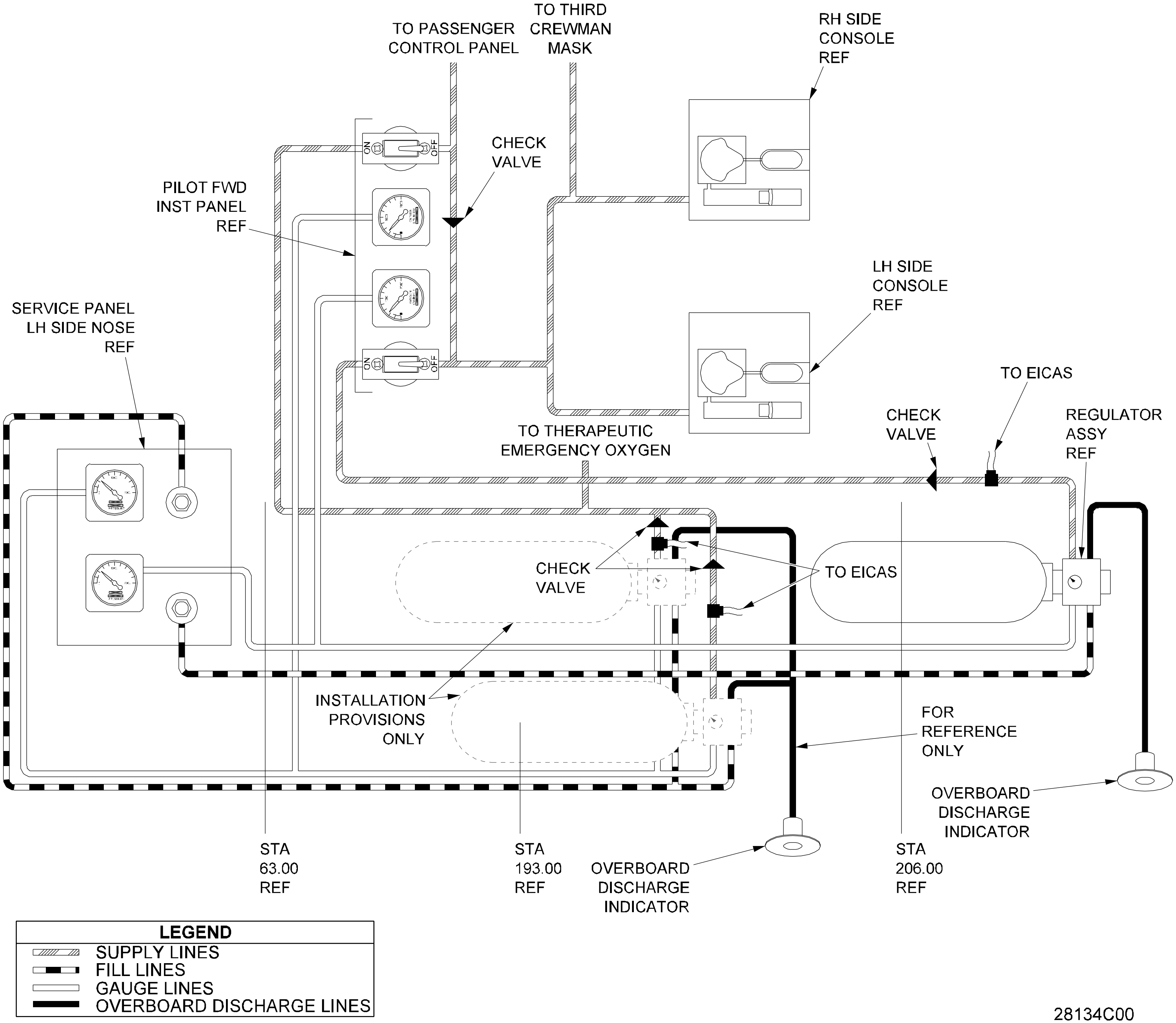
C. Average Time of Useful Consciousness:

- 15,000 to 18,000 feet = 30 minutes or more
- 22,000 feet = 5 to 10 minutes
- 25,000 feet = 3 to 5 minutes
- 28,000 feet = 2½ to 3 minutes
- 30,000 feet = 1 to 2 minutes

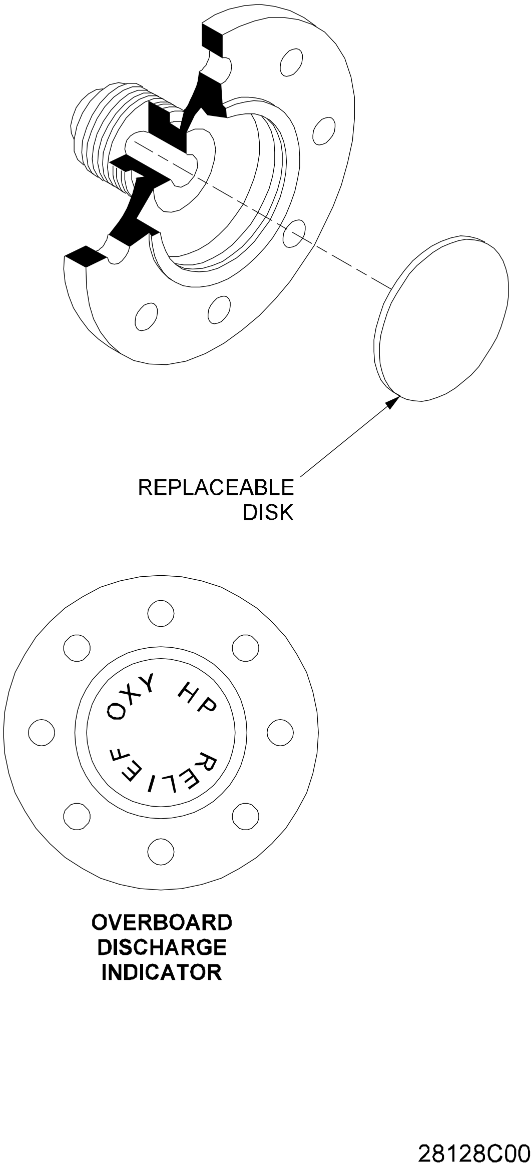
- 35,000 feet = 30 to 60 seconds
- 40,000 feet = 15 to 20 seconds
- 45,000 feet = 9 to 15 seconds
- 51,000 feet = less than 9 seconds



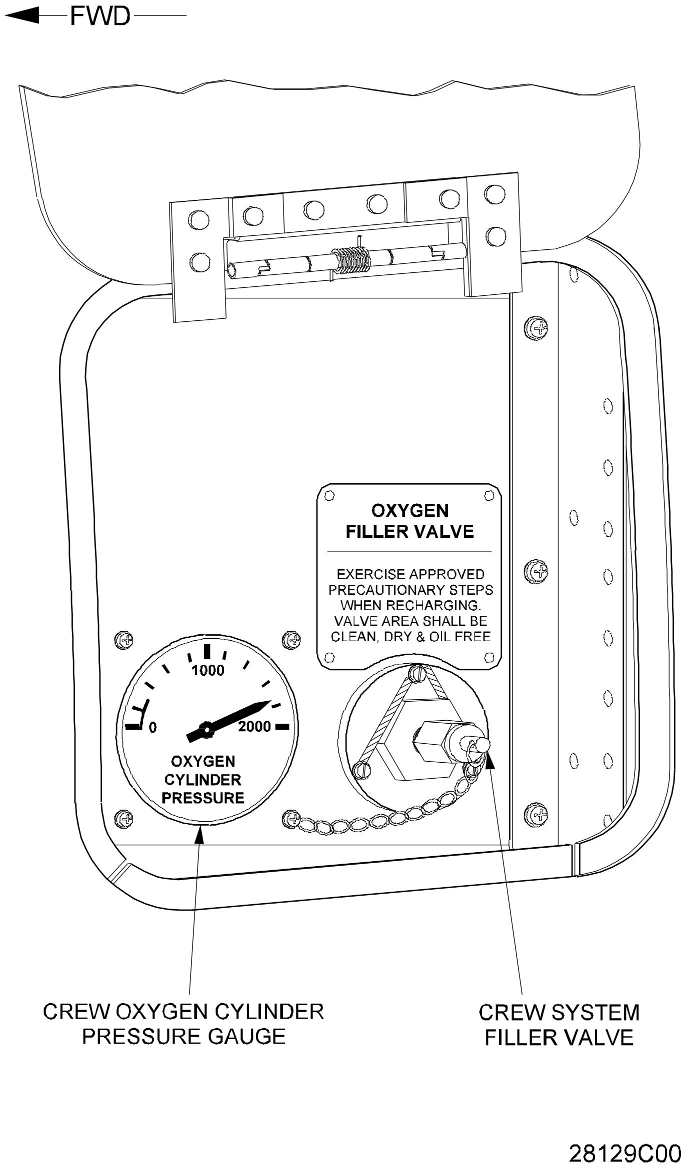
Crew Oxygen System Simplified Block Diagram: Airplanes SN 1000 Through 1289
Figure 1



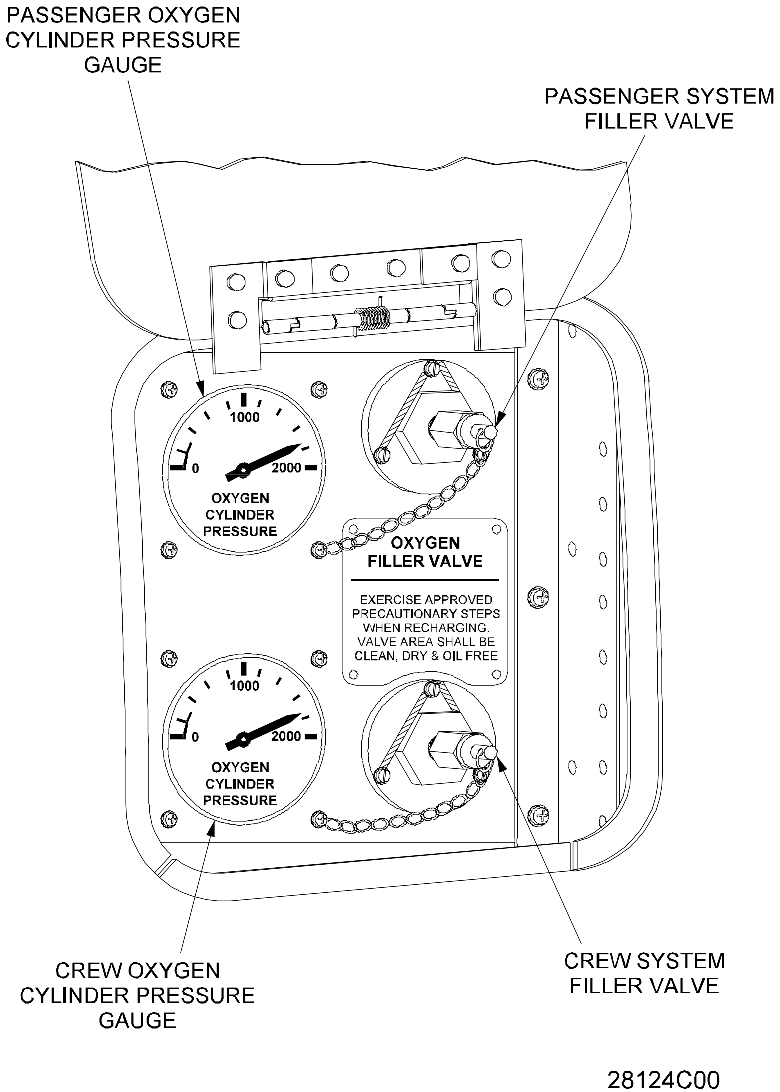
Crew and Passenger Oxygen System Simplified Block Diagram: Airplanes SN 1290 and Subsequent
Figure 2



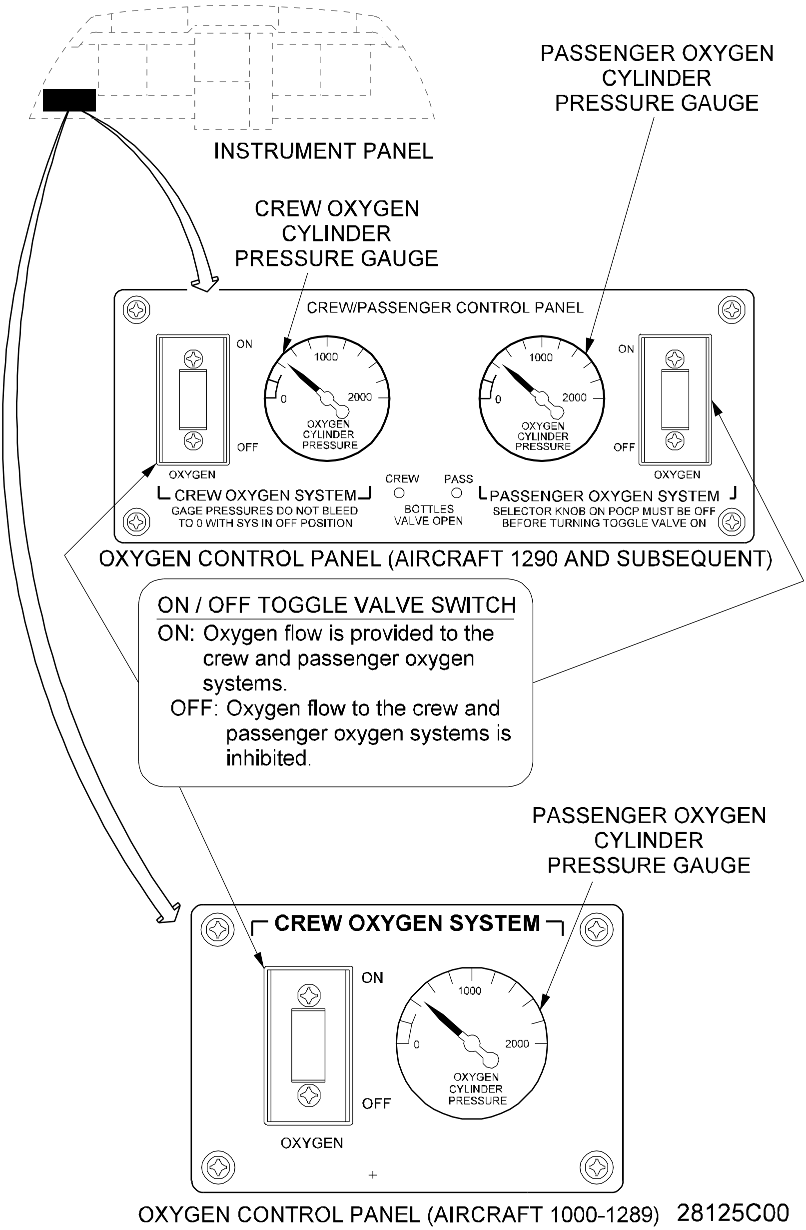
Overboard Discharge Indicator
Figure 3



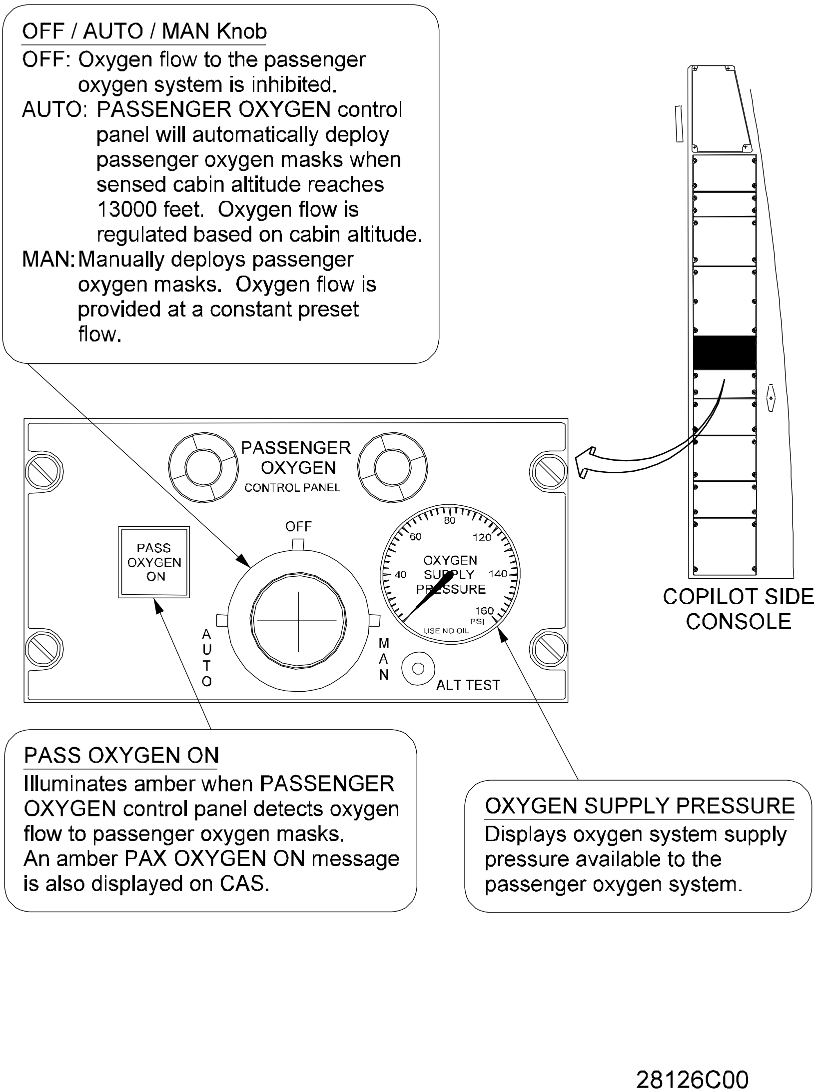
Oxygen Servicing Panel: Airplanes SN 1000 Through 1289
Figure 4



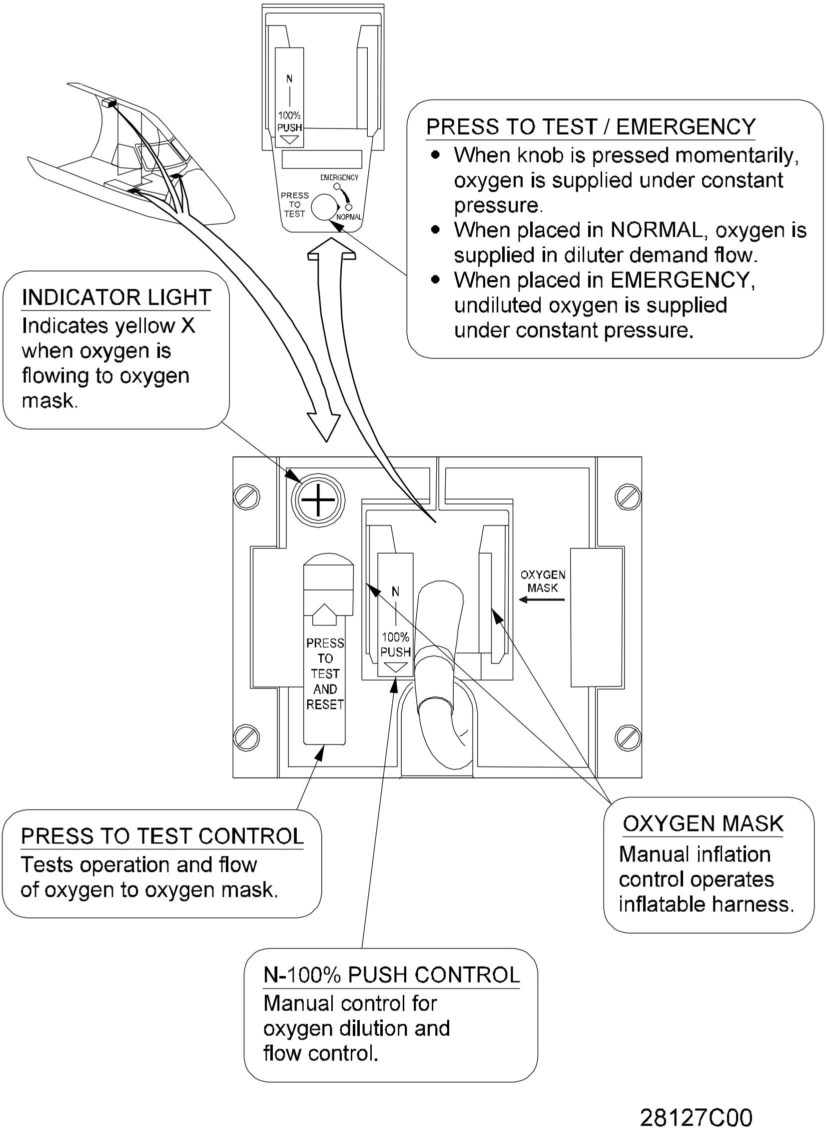
Oxygen Servicing Panel: Airplanes SN 1290 and Subsequent
Figure 5



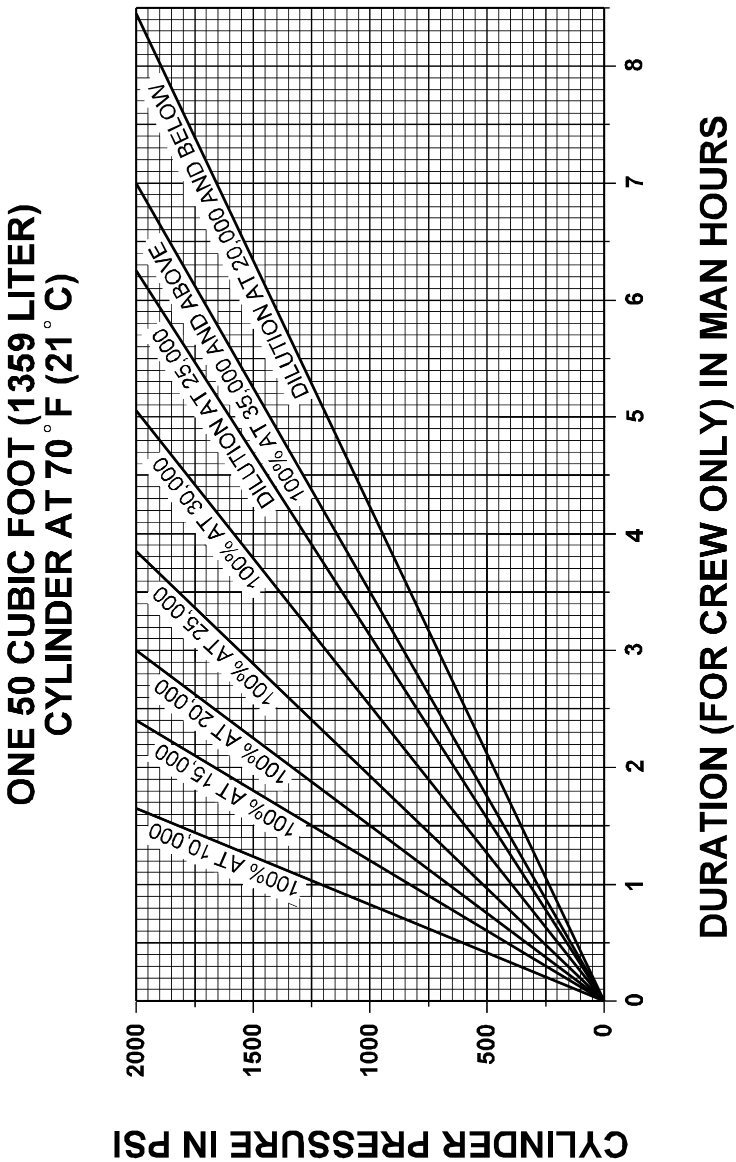
Oxygen System Control Panels
Figure 6



Passenger Oxygen Control Panel
Figure 7



EROS MXP 300 Crew Mask/Regulator Assembly
Figure 8



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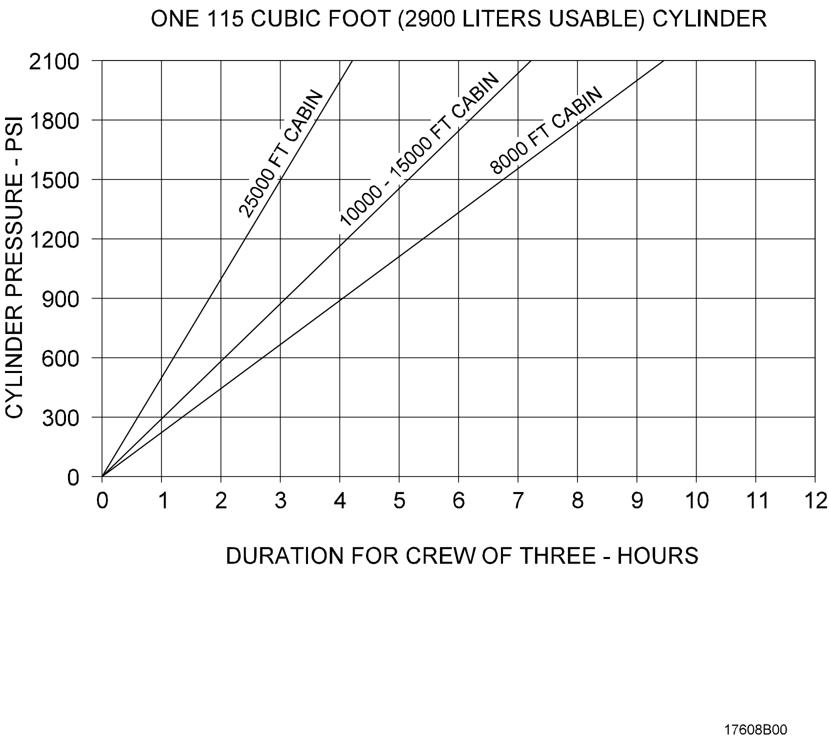
Oxygen Duration Versus Cabin Altitude: SN 1000 thru 1289
Figure 9



Oxygen Duration Versus Cabin Altitude: SN 1290 and Subsequent
Figure 10

2A-36-10: General

The purpose of the pneumatics system is to provide a common manifold, known as the bleed air manifold, with hot compressed air for use by other systems. Therefore, this section describes the sources of air for the bleed air manifold, their controls and the indications they provide to the flight crew. The airplane systems which use air from the bleed air manifold are described in the appropriate sections of this chapter.

The pneumatics system uses the following types of bleed air:

- Low pressure (LP) bleed air from the engines
- High pressure (HP) bleed air from the engines
- Bleed air from the single compressor stage of the Auxiliary Power Unit (APU)
- External air from a comparable bleed air source (external pneumatic rig)

The pneumatics system consists of two identical and independent systems that provide a temperature controlled, pressure regulated air supply for the various using systems, such as air conditioning, wing and nose cowl anti-icing, cabin pressure, engine starting and door seal systems. The pneumatics system receives its air supply from the mid-stage and high-stage ports of the engines. Cooling air is supplied to the precooler by a fan port while bleed air is extracted from either the mid-stage or high-stage port, depending on need. Temperature control is provided by both an electro-pneumatic control system (bleed air manifold temperature control) and a fully pneumatic subsystem (anti-ice augmentation temperature control). Pressure regulation is fully pneumatic for both manifold pressure regulation and dual setting high-stage switching pressure control.

The bleed air manifold can be termed that portion of the system from the right and left bleed air pressure regulator and shutoff valves and precoolers inboard. It is essentially a distribution point for all airplane pneumatic services, except those tapped off at each engine for engine services. The manifold is, as a whole, located in the tail compartment. Manifold air temperature is held to approximately 400° F; air pressure is held to approximately 40.5 ±3.5 psi (±5.5 psi under high input air flows).

Bleed air from the APU is used to pressurize the bleed air manifold for use by airplane services such as ground air conditioning and engine starting. The temperature of this air is a held to a maximum of 591° F; air pressure is held to a maximum of 53.9 psi.

The airplane incorporates an external air connection which may be used on the ground for the same services as the APU. When used with a source comparable to the APU, the external air will pressurize the bleed air manifold. The external air connection point is located in the underside of the fuselage, just forward of the hydraulic system service access door.

Door seals requiring pneumatic pressure for inflation receive pressure from the bleed air manifold through the door seal regulator. The regulator reduces the pressure to 17-19 psi for this application. The door seal system is described in Section 2A-52-00, Doors.

Two ducts are tapped into the bleed air manifold: one for the cockpit air conditioning system; the other for the cabin air conditioning system. Each duct has an air conditioning shutoff valve, which is a combination shutoff and flow regulating valve. One of its functions is to maintain a constant flow to the system. It also has a shutoff function, which will close the valve when it is desirable to terminate airflow into this system. The remainder of the air conditioning system is described in Section 2A-21-00, Air Conditioning.

Two additional ducts are also tapped into the bleed air manifold: one for the left wing

anti-icing; the other for the right wing anti-icing. Each duct has an anti-ice valve controlling the airflow. Downstream of the valves, the right wing and left wing ducts join, delivering part of the air into a crossover duct. When on, the wing anti-icing system creates the highest demand on the bleed air manifold. The wing anti-icing system is described in Section 2A-30-00, Ice and Rain Protection.

2A-36-20: Pneumatics Distribution and Indication System

1. **General Description:**

The pneumatics distribution and indication system supplies 7th stage or 12th stage bleed air to the bleed air manifold where it is available to the systems that require bleed air, and provides visual indication of the pressure and temperature of the bleed air within the bleed air manifold. System design is such that 7th stage (mid stage) bleed air is the preferred air source and 12th stage (high stage) bleed air serves as an alternate or supplemental source whenever 7th stage pressures are not adequate for the airplane's needs. A check valve is incorporated in the 7th stage duct to prevent backflow of the higher-pressure 12th stage air. This check valve is commonly referred to as the LP check valve.

In the standard airplane configuration, 7th stage bleed air will supply all bleed flow during takeoff, climb and cruise conditions. The 12th stage bleed valve will remain closed due to existing 7th stage pressure being greater than 18 ±2 psi. Should 7th stage pressure fall below 18 ±2 psi for any reason, the 12th stage bleed valve will open and regulate pressure to 18 ±2 psi. In all cases, bleed air passes through manifold pressure regulator valves into the left or right bleed air manifold, where pressure is maintained at a maximum of 40.5 ±3.5 psi by these valves. Under high input air flows, manifold pressure tolerance increases to ±5.5 psi.

The pneumatics distribution and indication system consists of identical and independent left and right sides. For the purposes of this description, however, the following components are discussed in the singular sense to avoid repetition:

- HP Bleed Air Pressure Regulator and Shutoff Valve
- Bleed Air Pressure Regulator and Shutoff Valve
- Fan Air and Fan Air Modulating Valve
- Precooler
- Temperature and Pressure Sensors/Switches

In addition to the above-listed left and right side components, the following subsystems and components are included:

- Bleed Air Isolation Valve
- APU Bleed Air Distribution
- External Air Distribution
- Door Seal Pressure Distribution
- Total Air Temperature Probe Aspiration

2. **Description of Subsystems, Units, and Components:**

(See Figure 1, Figure 2 and Figure 3.)

NOTE:

Items A through E are identical to the left and right sides, therefore only one side will be discussed.

A. HP Bleed Air Pressure Regulator and Shutoff Valve:

The HP bleed air pressure regulator and shutoff valve (commonly referred to as the HP valve) is a butterfly-type modulation valve installed in the high-stage bleed air line to regulate, modulate or shut off 12th stage bleed air. It works in conjunction with the solenoid "A" and "B" HP control valve and the dual port solenoid shutoff valve to provide three distinct modes of operation (these modes are described below). The valves receive power from the Essential 28V DC bus through the L BLEED AIR and R BLEED AIR circuit breakers, respectively. A description of this valve's two controlling components follows:

(1) 12th Stage Servo Controller (Solenoid A and B HP Control Valve):

The 12th stage servo controller (solenoid A and B HP control valve) provides the necessary commands to the HP bleed air pressure regulator and shutoff valve to achieve the proper mode of control. The controller is a remote pressure regulating servo with a solenoid shutoff feature, a relief valve, and a thermostat connection for sensing downstream temperature. It is linked to the bleed air temperature sensor and the bleed air temperature anticipator sensor via the dual port solenoid shutoff valve.

(2) Dual Port Solenoid Shutoff Valve:

The dual port solenoid shutoff valve is installed to give a desired mode of system operation. The unit is pneumatically linked between the 12th stage servo controller (solenoid A and B HP control valve), the HP bleed air pressure regulator and shutoff valve, and the bleed air temperature sensor/anticipator sensor control set. Pneumatic control air is received from the solenoid A and B HP control valve via the actuator port. Control sensor plumbing is connected to the control port. The pneumatic signal is then manipulated by the dual port solenoid shutoff valve to give the proper HP bleed air pressure regulator and shutoff valve control.

As stated previously, the HP bleed air pressure regulator and shutoff valve provides three distinct modes of operation. These are the pressure regulation mode, the temperature modulation mode and the single pack mode. A description of these modes follows:

(3) Pressure Regulation Mode:

In the regulation mode, the HP bleed air pressure regulator and shutoff valve opens or closes to regulate bleed air to a constant pressure. If 7th stage pressure falls below 18 ±2 psi during low power settings such as idle descent or taxi, solenoid A of the 12th stage servo controller is energized to regulate the HP bleed air pressure regulator and shutoff valve and maintain pressure at 18 ±2 psi. During normal operations, the dual port solenoid shutoff valve remains closed.

It should be noted here that although 18 ±2 psi is a design reference pressure, it may not always be attainable due to ambient conditions. Pressures in the range of 12-16 psi at idle power are not considered abnormal as long as they are stable.

On airplanes Serial Number (SN) 1310 and subsequent and airplanes SN 1000 through 1309 having Aircraft Service Change

(ASC) 313, the rate at which cabin pressure changes during rapid power reductions at high altitudes is reduced. This ASC provides an electrical circuit, routed through the 2500 FT AGL horn mute relay, to reset bleed air pressure regulation from 18 psi to 30 psi when the power levers are retarded. In a retarded power lever descent (without anti-icing selected on) manifold pressure is maintained above 20 psi. Selection of any anti-icing on disables this circuit.

(4) Temperature Modulation Mode:

If wing and cowl anti-icing are selected on, pneumatic signals from the bleed air temperature sensor and the bleed air temperature anticipator sensor (both installed upstream of the precooler) coordinate with the 12th stage servo controller to reset the HP bleed air pressure regulator and shutoff valve. The valve then modulates mixed 7th and 12th stage air temperature to 520°-590° F (271°-310° C), thus modulating due to temperature, not pressure. With wing and cowl anti-icing are selected on, the dual port solenoid shutoff valve is de-energized open.

(5) Single Pack Mode:

With only one refrigeration pack selected on, both A and B solenoids of the 12th stage servo controller are energized. The HP bleed air pressure regulator and shutoff valve then resets to regulate pressure to 30 psi. This allows for increased performance due to single pack operations. Because the HP bleed air pressure regulator and shutoff valve is operating in single pack mode, the dual port solenoid shutoff valve remains closed.

If anti-icing is selected on during single pack operations, the system switches back to temperature control mode and the dual port solenoid shutoff valve is de-energized open.

B. Bleed Air Pressure Regulator and Shutoff Valve:

The bleed air pressure regulator and shutoff valve (also referred to the “40 psi valve”) is an electrically controlled, pneumatically actuated, spring-loaded closed butterfly-type valve. Located immediately downstream of the 7th and 12th stage mixing junction, this valve regulates air having passed through the HP bleed air pressure regulator and shutoff valve to 40.5 psi nominal in the bleed air manifold. The valves receive power from the Essential 28V DC bus through the L BLEED AIR and R BLEED AIR circuit breakers, respectively.

The bleed air pressure regulator and shutoff valve is opened or closed by the flight crew using the L/R ENG BLEED AIR switches located in the BLEED AIR section of the cockpit overhead panel. Depressing the switch routes Essential 28V DC bus power to an electrically actuated solenoid, opening the valve and allowing pressure regulated bleed air through the precooler to the bleed air manifold. The OFF legend in the switch is extinguished.

C. Fan Air and Fan Air Modulating Valve:

Each engine’s fan supplies low pressure bypass air (commonly referred to as LP or fan air) for use by both the airplane and its engines. For the engines, fan air is used to pressurize engine oil system components and cool various other engine components. For the airplane, fan air is routed

through the precoolers to reduce the temperature of 7th and/or 12th stage bleed air.

The precooler fan air modulating valve (installed in the engine pylon) controls fan air airflow to the precooler in order to maintain a nominal 400° F (204° C) bleed air temperature. Precooler outlet air temperature is continuously monitored by a precooler outlet temperature sensor and a precooler temperature control anticipator sensor. Outputs from these sensors are transmitted to the precooler temperature controller. The controller, in turn, will change the output voltage to the servo air pressure regulator and torque motor. The servo air pressure regulator and torque motor then changes the electrical signal to a filtered pneumatic signal, positioning the precooler fan air modulating valve accordingly.

D. Precooler:

The precooler, located in the engine pylon, functions as a heat exchanger for the pneumatic system. HP bleed air flows from the bleed air pressure regulator and shutoff valve directly to the precooler, where fan air is introduced by the fan air modulating valve. Fan air and bleed air do not mix, rather, bleed air passes through the interior of the precooler while fan air flows around the exterior and exits through louvers in the bottom of the pylon.

E. Temperature and Pressure Sensors/Switches:

(1) Bleed Air Temperature Sensor:

The bleed air temperature sensor is located in the bleed air manifold, upstream of the precooler. It senses the mixed 7th and 12th stage air temperature and then modulates the HP bleed air pressure regulator and shutoff valve via the dual port solenoid shutoff valve and the 12th stage servo controller (solenoid “A” and “B” HP control valve) until a 520°-590° F (271°-310° C) temperature is maintained downstream of the mixing point.

(2) Bleed Air Temperature Anticipator Sensor:

The bleed air temperature anticipator sensor is also located in the bleed air manifold, upstream of the precooler. Functioning as a bleed-off anticipator, it responds to the rate of temperature change, not the temperature change itself. Rapid heating of the sensor causes an internal ball to become unseated which, in turn, bleeds off a portion of the pneumatic signal that modulates the HP bleed air pressure regulator and shutoff valve. This action prevents excessive overshoots of the HP bleed air pressure regulator and shutoff valve and bleed air temperature sensor combination which may cause unstable control.

(3) Precooler Outlet Temperature Sensor:

The precooler outlet temperature sensor is installed in the bleed air manifold, downstream of the precooler. As part of the engine fan air control system, it works in conjunction with the precooler temperature control anticipator sensor, the precooler outlet temperature controller and the precooler fan air modulating valve’s servo air pressure regulator and torque motor to maintain engine fan airflow through the precooler at the desired amount. As the bleed air manifold temperature changes, the sensor reacts accordingly by

changing resistance. This change in resistance is transmitted to the precooler outlet temperature controller, which in turn will change the output voltage to the servo air pressure regulator and torque motor. The servo air pressure regulator and torque motor then changes the electrical signal to a pneumatic signal to position the precooler fan air modulating valve accordingly. The temperature sensors (as well as the temperature controllers and servo air pressure regulator and torque motors) receive power from the Essential 28V DC bus through the L BLEED AIR and R BLEED AIR circuit breakers, respectively.

(4) Precooler Temperature Control Anticipator Sensor:

The precooler temperature control anticipator sensor is also installed in the bleed air manifold, downstream of the precooler. Like the bleed air temperature anticipator sensor, it responds to the rate of temperature change, not the temperature change itself. This prevents excessive overshoots of the fan air modulating valve and precooler outlet temperature sensor combination which may cause unstable control. The anticipator's outputs, combined with the precooler outlet temperature sensor outputs, are used by the precooler outlet temperature controller to create an electrical signal for the precooler valve servo air pressure regulator and torque motor which, in turn, controls the precooler fan air modulating valve. The temperature anticipator sensors receive power from the Essential 28V DC bus through the L BLEED AIR and R BLEED AIR circuit breakers, respectively.

(5) Bleed Air Overpressure Switch:

A bleed air overpressure switch is installed in the bleed air manifold duct to provide input signals when manifold pressures reach 75±5 psi. When this threshold is reached, an amber L-R BLEED PRESS HI caution message is displayed on CAS. The switches receive power from the warning lights power system.

(6) Bleed Air Pressure Transmitter:

A bleed air pressure transmitter (transducer) is installed in the bleed air manifold duct to provide input signals to the bleed air pressure indicator's digital readout of the bleed air system pressure. The transmitter also provides input signals used to present the L/R BLEED AIR PRESS readings on the APU/BLEED and ENGINE START system pages. The transmitters and the indicators receive power from the Essential 28V DC bus through the L BLEED AIR IND and R BLEED AIR IND circuit breakers, respectively.

(7) Bleed Air Overtemperature Switch (550°F):

A bleed air overtemperature switch is installed to alert the crew when manifold bleed discharge temperatures reach 550°F (288° C). When this threshold is reached, an amber L-R BLEED AIR HOT caution message is displayed on CAS. The switches receive power from the equipment overheat protection system.

F. Bleed Air Isolation Valve:

Located in the tail compartment, the bleed air isolation valve is installed in the crossover duct between the left and right bleed air manifolds. It

provides a means to isolate the left and right side pneumatic systems when closed (its normal position). Conversely, when open, it joins the left and right side pneumatic systems, allowing the use of APU bleed air for both ECS packs during two pack air conditioning operations. In addition, it provides crossbleed capability during single engine operations to enable engine starting and cranking using the opposite side's pneumatic system, as well as APU bleed air. The bleed air isolation valve receives power from the Essential 115V AC bus (ϕA) through the BLEED AIR ISO S/O V circuit breaker.

During normal operations, the isolation valve is closed and the left and right side pneumatic systems are isolated from each other. Manual selection of the ISOLATION switch (cockpit overhead panel, BLEED AIR section) opens the isolation valve. A "bar" in the switch illuminates to fill in the crossover manifold line engraved on the BLEED AIR panel, signifying that the left and right side pneumatic systems are no longer isolated. In addition to the switch legend, a blue ISOLATION VLV OPEN advisory message is displayed on CAS. With the isolation valve open, crossbleed air and APU bleed air are available to both pneumatic systems for use by both refrigeration packs and for engine starting as required.

In addition to manual control, the bleed air isolation valve is capable of automatic control while the airplane is on the ground by certain bleed air configurations. If the isolation valve is closed, selection of APU BLEED AIR to ON or selection of either the MASTER CRANK or MASTER START switches to ON will automatically open the isolation valve. When APU BLEED AIR is selected off or the MASTER CRANK or MASTER START switches are selected off, the isolation valve will automatically close.

A manual override slot drive is provided on the valve itself so that the valve can be positioned with a screwdriver without the need for electrical power. The slot drive is connected to a common shaft, ensuring the synchronization of internal limit switches with valve position at all times regardless of whether the valve is actuated electrically or manually.

On airplanes SN 1445 and subsequent and SN 1000-1444 with ASC 422A, the existing bleed air system wiring is modified to provide an annunciation for an incorrect bleed air configuration. This annunciation serves to caution the flight crew to the possibility of engine overtemperature if the isolation valve is open while the APU and/or engine bleed air switches are set to certain positions while on the ground. On SPZ-8400 equipped airplanes, an amber BLEED CONFIG caution message is displayed on CAS. On SPZ-8000 equipped airplanes, an amber BLEED CONFIG light illuminates above each navigation display. See Figure 5. After a five second delay, display of the annunciation is prompted when:

- Both ENG BLEED AIR switches are selected ON (OFF legend extinguished) with the isolation valve OPEN, **or:**
- Either or both ENG BLEED AIR switches are selected ON and APU BLEED AIR is selected ON (isolation valve opens automatically), **or:**
- Any two of the three available bleed sources (L ENG, R ENG and/or

APU) are selected ON and either the MASTER CRANK or MASTER START switch is selected ON (isolation valve opens automatically)

NOTE:

To incorporate ASC 422A on SPZ-8000 equipped airplanes, ASCs 327, 327A, 327B and any associated amendments must first be incorporated.

G. APU Bleed Air Distribution:

The APU bleed air duct is connected to the right side of the bleed air manifold. Selection of the APU BLEED AIR switch (cockpit overhead panel, BLEED AIR section) to ON opens the APU's load control valve and the isolation valve. The ON legend in the switch illuminates. APU bleed air then flows through a check valve in the APU bleed air duct into the bleed air manifold. Should bleed air manifold pressure exceed APU bleed air pressure, the check valve in the APU bleed air duct will close to prevent reverse flow.

APU bleed air is inhibited while airborne. Only electrical power is available from the APU.

H. External Air Distribution:

(See Figure 9.)

The external air duct is connected to the left side of the bleed air manifold. On the opposite end of the duct is a coupling where the external air cart is connected. The external air connection point is located in the underside of the fuselage, just forward of the hydraulic system service access door. With the proper external air cart connected, the flight crew may use external air as required for engine starting. Should bleed air manifold pressure exceed external air duct pressure, a check valve installed in the external air duct will close to prevent reverse flow.

I. Door Seal Pressure Distribution:

(See Figure 1, Figure 3 and Figure 6 through Figure 8.)

The door seal pressure line and pressure regulator are connected to the bleed air manifold. Location of the line on the manifold is such that either the left or right bleed air system can furnish a supply of bleed air. The regulator is designed to provide 18 ±1 psi output to inflate the main entrance door and baggage door seals. For airplanes having the cargo door modification, door sealing is also provided by the door seal system for the cargo door.

On airplanes SN 1285 and subsequent and airplanes SN 1000 through 1284 having ASC 364, redundancy is added to the door seal system by routing an additional constant high pressure air source to the pressure regulator. (See Figure 2 and Figure 3 instead of Figure 1 and Figure 3.) This ensures maximum flow output from the pressure regulator to the door seals. Additional pressure lines, one for each engine, are ported into the 12th stage bleed air duct upstream of the HP bleed air pressure regulator and shutoff valve. The line is routed into the tail compartment, where it joins into the existing door seal pressure line. The existing door seal pressure line is then re-routed to separate the total air temperature probe aspiration pressure supply and water system pressure tap (for later outfitting) from

the door seal system. Thus the door seal system and pressure regulator have a separate manifold.

J. **Total Air Temperature Probe Aspiration:**

A Total Air Temperature (TAT) probe aspiration pressure line and solenoid-operated shutoff valve (commonly referred to as the total temp valve) are connected to the bleed air manifold. Location of the line on the manifold is such that either the left or right bleed air system can furnish a supply of bleed air to the shutoff valve.

The TAT probe is located on the lower right side of the forward fuselage below the angle of attack probe. To ensure accurate calibration readings, bleed air is introduced through the TAT probe when the airplane is on the ground. Through this process, known as aspiration, bleed air is supplied to the probe from the shutoff valve when the nutcracker shifts to the ground mode. This shutoff valve receives power from the Right Main 28V DC bus.

3. Controls and Indications:

(See Figure 4.)

A. Circuit Breakers:

Circuit Breaker Name	CB Panel	Location	Power Source
L BLEED AIR	PO	A-10	ESS DC Bus
R BLEED AIR	PO	B-10	ESS DC Bus
BLEED AIR ISO S/O V	PO	D-12	ESS AC Bus ϕA
L BLEED AIR IND	PO	B-12	ESS DC Bus
R BLEED AIR IND	PO	C-12	ESS DC Bus

B. Caution (Amber) Crew Alerting System (CAS) Messages:

CAS Message	Cause or Meaning
L-R BLEED AIR HOT	Bleed air temperature is above 550° F (288° C).
BLEED CONFIG	(1) Programmable custom message activated as desired during outfitting. (2) Isolation valve is OPEN with engine bleed selected ON.
L-R BLEED PRESS HI	Bleed air pressure has exceeded 75 psi.
DOOR SEAL PRESSURE	Programmable custom message activated as desired during outfitting.

NOTE(S):

(1) Airplanes not having ASC 422A.

(2) Airplanes SN 1445 & subs; SN 1000-1444 having ASC 422A.

C. Advisory (Blue) CAS Messages:

CAS Message	Cause or Meaning
DR SUPPLY PRES LOW	Programmable custom message activated as desired during outfitting.
ISOLATION VLV OPEN	Isolation valve is open.

NOTE:

A description of the Engine Instruments and Crew Alerting System (EICAS) can be found in Section 5 of Honeywell's <u>SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV</u>.

4. Limitations:

A. Flight Manual Limitations:

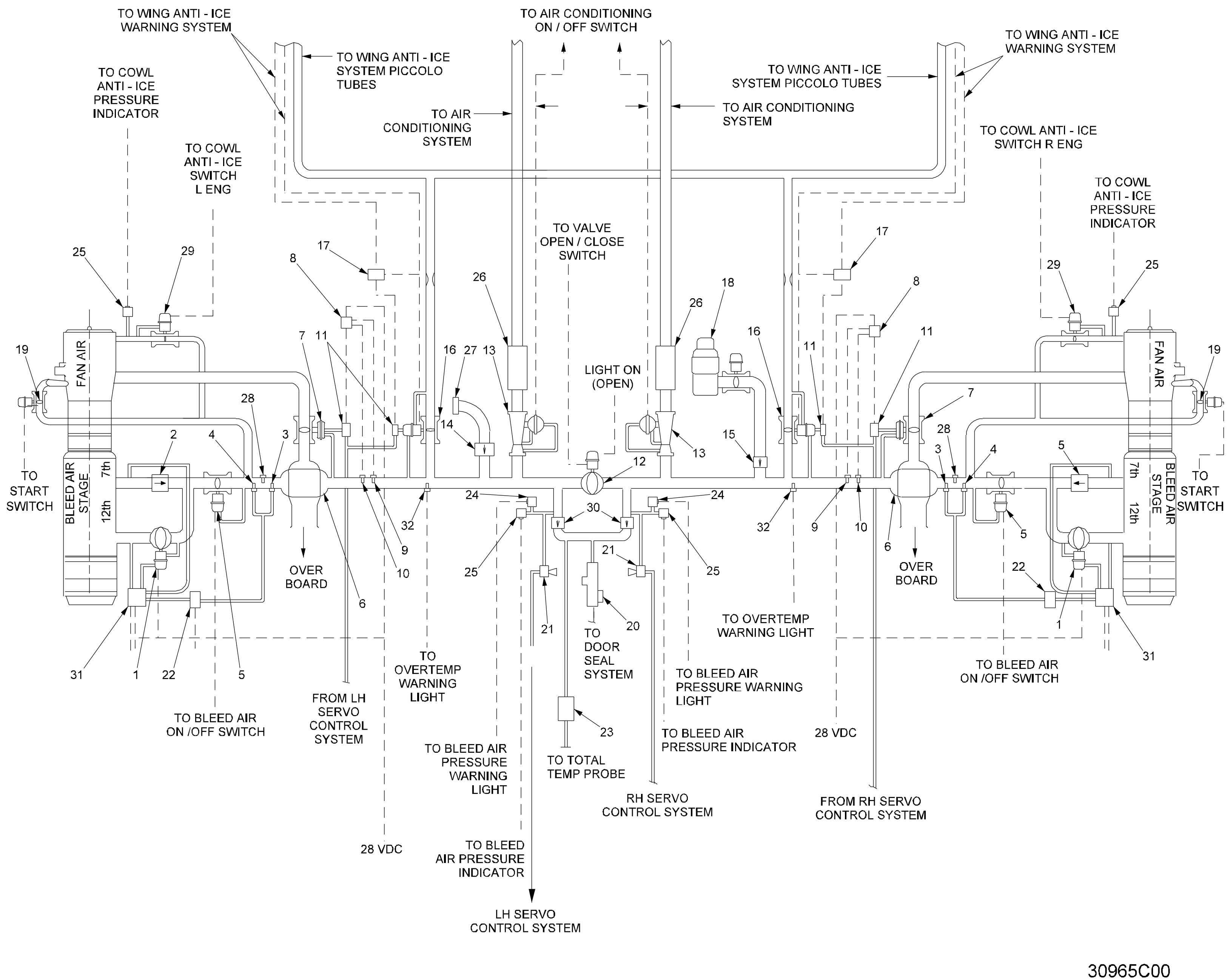
Do not operate above 41,000 ft without both engine bleeds ON and each engine being bled by either the air conditioning system or engine cowl anti-ice. See Section 05-01-10, Air Conditioning System Shut Down Or Inoperative.

B. System Notes:

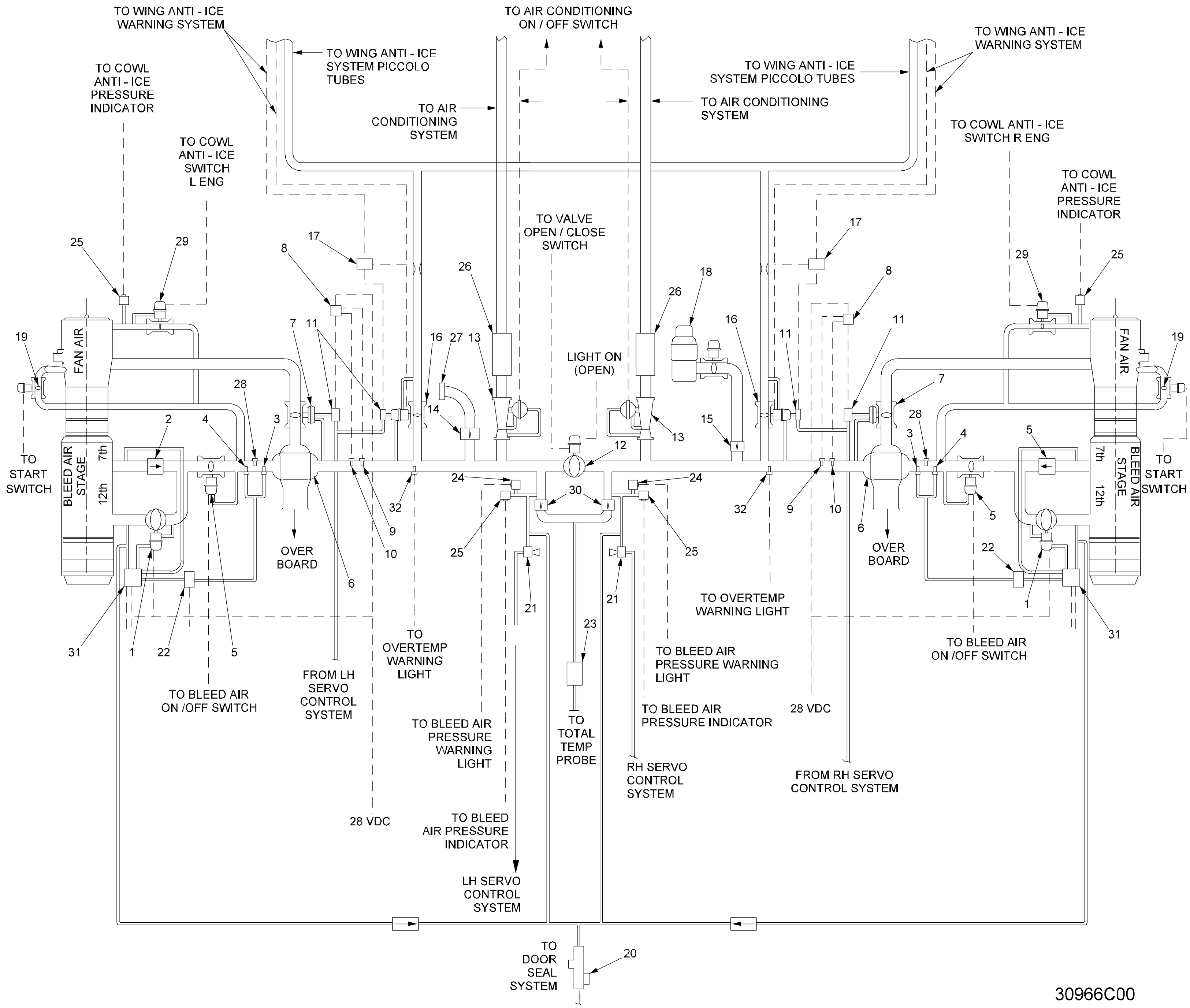
(1) The bleed air isolation valve must be closed and APU bleed air OFF before the autothrottle can be engaged.

(2) Only one source of bleed air, either APU or engines, should be selected after engines are started during normal operations. This is to prevent thermal transients on the APU or possible damage to the APU when the power levers are moved from idle. With the engines as the source of bleed air, ensure the isolation valve is closed.

(3) To provide cooling air flow to the cabin during warm weather operations (if taxi operations are conducted with engine bleed air as the source for the ECS packs), it is recommended that one power lever be advanced above idle with the isolation valve open and the opposite engine bleed air be selected off.



Pneumatics System Block Diagram: SN 1000-1284 Not Having ASC 364
Figure 1



Pneumatics System Block Diagram: SN 1285 and Subsequent; SN 1000-1284 Having ASC 364
Figure 2

ITEM NO.	DESCRIPTION
1	VALVE, H.P. BLEED AIR PRESSURE REG. & S/O
2	VALVE, L.P. BLEED AIR CHECK
3	SENSOR, BLEED AIR TEMPERATURE
4	SENSOR, BLEED AIR TEMPERATURE ANTICIPATOR
5	VALVE, BLEED AIR PRESSURE REG. & S/O
6	HEAT EXCHANGER, BLEED AIR PRECOOLER
7	VALVE, PRECOOLER FAN AIR MODULATING, 4" DIA
8	CONTROLLER, PRECOOLER OUTLET TEMPERATURE
9	SENSOR, PRECOOLER OUTLET TEMPERATURE
10	SENSOR, PRECOOLER TEMPERATURE CONTROL ANTICIPATOR
11	VALVE, SERVO AIR PRESSURE REG
12	VALVE, BLEED AIR ISOLATION S/O
13	VALVE, AIR CONDITIONING SYSTEM SHUTOFF & FLOW CONTROL
14	VALVE, EXTERNAL AIR CHECK
15	VALVE, AUX POWER UNIT AIR CHECK.
16	VALVE, WING ANTI-ICE PRESSURE REG S/O & TEMPERATURE CONTROL
17	CONTROLLER, WING ANTI-ICE SYSTEM TEMPERATURE
18	AUXILIARY POWER UNIT (APU)
19	AIR TURBINE STARTER & BLEED CONTROL VALVE
20	VALVE, DOOR SEAL PRESSURE REGULATOR
21	FILTER, SERVO AIR PRESSURE
22	VALVE, DUAL PORT SOLENOID SHUTOFF
23	VALVE, SOLENOID SHUTOFF, TOTAL TEMP PROBE
24	SWITCH, BLEED AIR OVERPRESSURE (65 PSI)
25	TRANSMITTER, BLEED AIR/COWL PRESSURE
26	FILTER, CABIN/COCKPIT OZONE
27	NIPPLE, EXTERNAL AIR
28	SWITCH, COWL ANTI-ICE DUCT OVERHEAT (675° F)
29	VALVE, COWL ANTI-ICE PRESSURE REGULATOR & SHUTOFF
30	VALVE, CHECK BLEED AIR DOOR SEAL SYSTEM
31	VALVE, SOLENOID A & B, H.P. VALVE CONTROL
32	SWITCH, BLEED AIR OVERTEMP (550° F)

LEGEND

——— (two single lines)	BLEED AIR & AIR COND. DISTRIBUTION LINES
═══ (double line)	PNEUMATIC CONTROL SYSTEM
– – – – (dashed line)	ELECTRICAL LINES

31256C00

Pneumatics System Block Diagram Key
Figure 3

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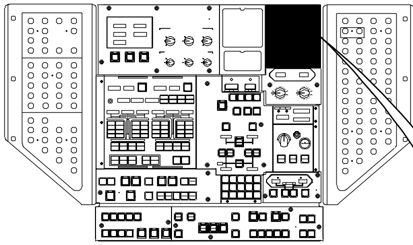


L ENG / R ENG BLEED AIR Pressure Indicator
Provides digital display (in psi) of bleed air system pressure sensed by associated manifold's bleed air pressure transmitter. Adjacent color range bar shows acceptable range of 0-60 psi and a caution range of greater than 60 psi.
NOTE: Pressures exceeding 75 psi cause an amber L-R BLEED PRESS HI caution message to be displayed on CAS.

ISOLATION Valve
Closed (Normal Position):
- Left and right side pneumatic systems are isolated from each other
- APU bleed air is available to only right side pneumatic system
- External bleed air is available to only left side pneumatic system
- Bar legend in switch is extinguished

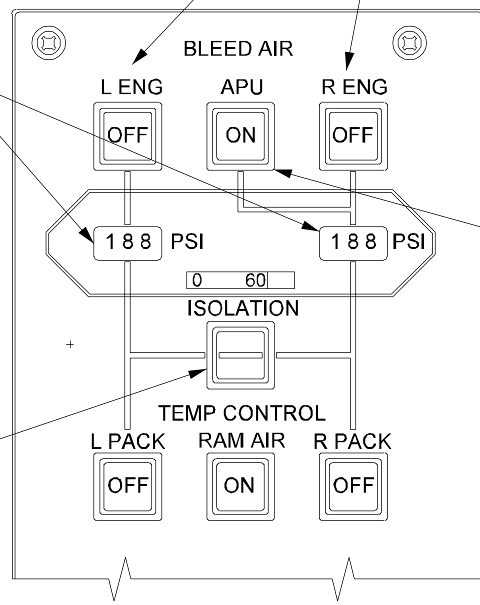
Open (Manually or Automatically - on Ground Only):
- Bar legend in switch illuminates
- ESS 115V AC power is supplied to isolation valve
- Left and right side pneumatic systems are joined
- APU bleed air available for "two pack" air conditioning ground operations
- Crossbleed capability available for single engine ground operations, enabling engine cranking/starting using opposite side pneumatic system or APU bleed air
- Blue ISOLATION VLV OPEN advisory message is displayed on CAS

Automatic Operation:
If closed, selection of following automatically opens isolation valve (on ground only):
- APU BLEED AIR to ON
- MASTER CRANK to ON
- MASTER START to ON

When selected off, isolation valve automatically closes.
NOTE: APU bleed air is inhibited while airborne.

L ENG / R ENG BLEED AIR
On:
- OFF legend extinguishes
- ESS 28V DC power is supplied to associated bleed air pressure regulator and shutoff valve
- Associated bleed air pressure regulator and shutoff valve opens, allowing pressure regulated bleed air to bleed air manifold

OFF:
- OFF legend illuminates
- ESS 28V DC power is removed from associated bleed air pressure regulator and shutoff valve
- Associated bleed air pressure regulator and shutoff valve closes, inhibiting bleed air to bleed air manifold



APU BLEED AIR
(Assumes APU running)
ON:
- ON legend illuminates
- APU load control valve opens
- APU bleed air is supplied to bleed air manifold

Off:
- ON legend extinguishes
- APU load control valve closes
- APU bleed air flow is inhibited

NOTE: APU bleed air is inhibited while airborne.

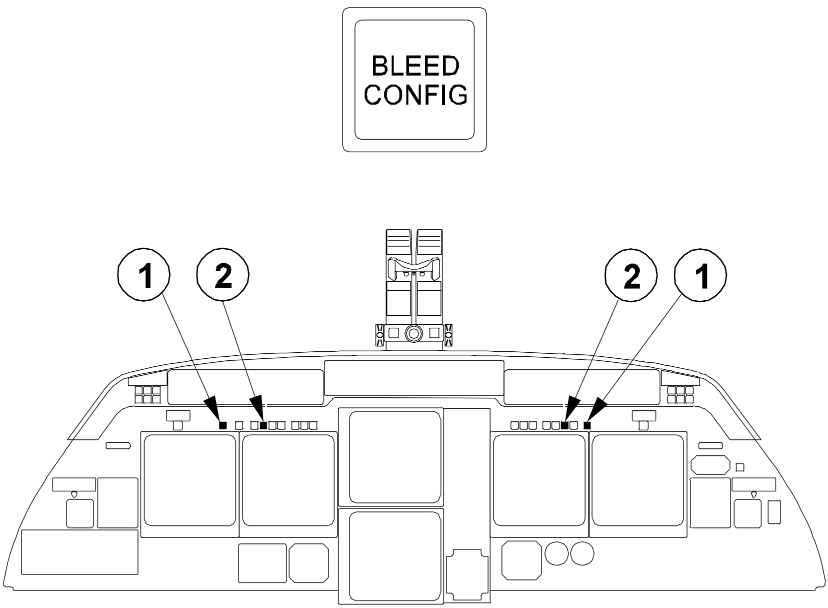
28294C01

BLEED AIR Control Panel
Figure 4

BLEED CONFIG

(SPZ-8000 equipped airplanes having ASC 422/422A)
Illuminates amber when:

- Both ENG BLEED AIR switches are selected ON with the isolation valve OPEN, or:
- Either or both ENG BLEED AIR switches are selected ON and APU BLEED AIR is selected ON, or:
- Any two of the three available bleed sources (L ENG, R ENG and/or APU) are selected ON and either the MASTER CRANK or MASTER START switch is selected ON.



1 AIRPLANES HAVING ELWS

2 AIRPLANES NOT HAVING ELWS OR HAVING ASC 420

35655C01

BLEED CONFIG Indicator: SPZ-8000 Equipped Airplanes Having ASC 422A
Figure 5

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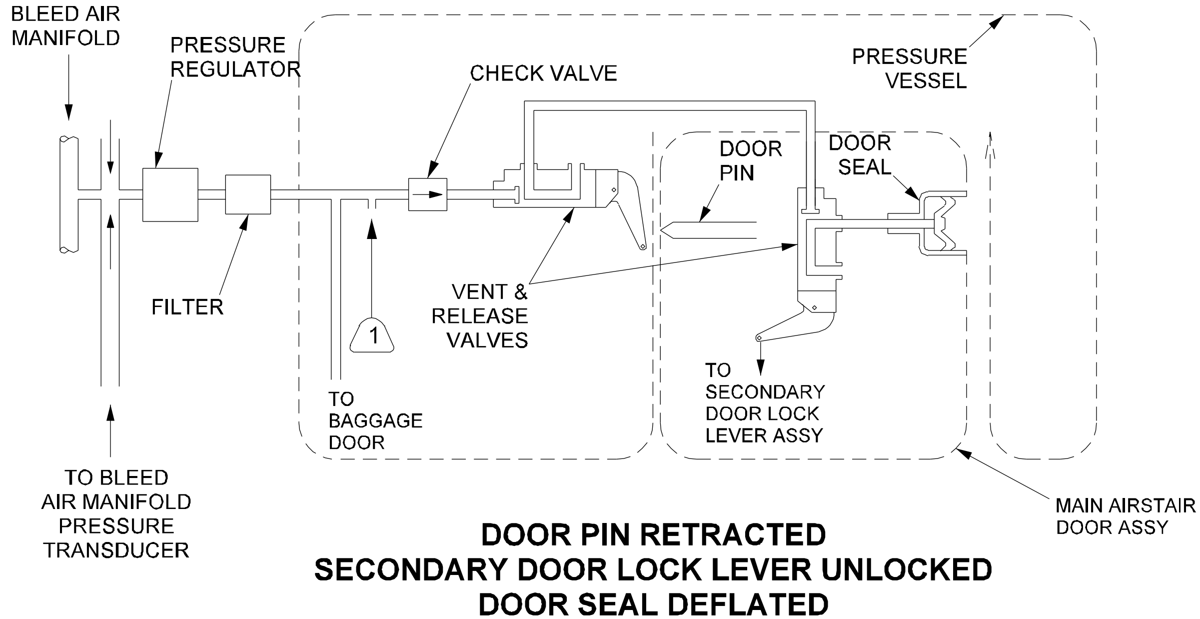

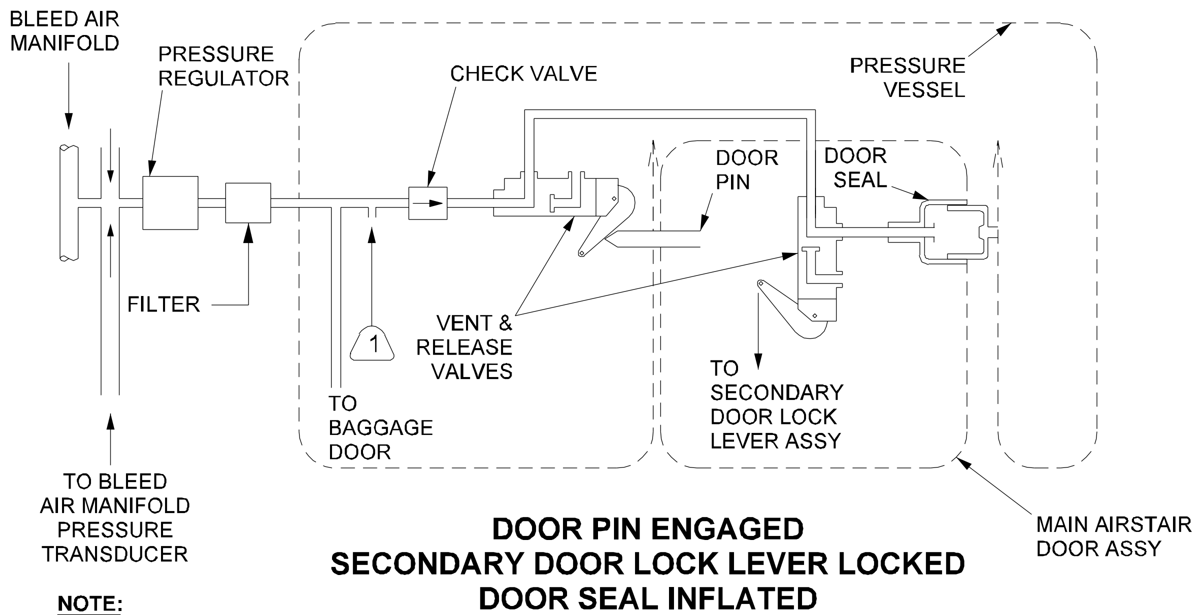

NOTE:

1 HOLE (0.027" - 0.032" DIA.)

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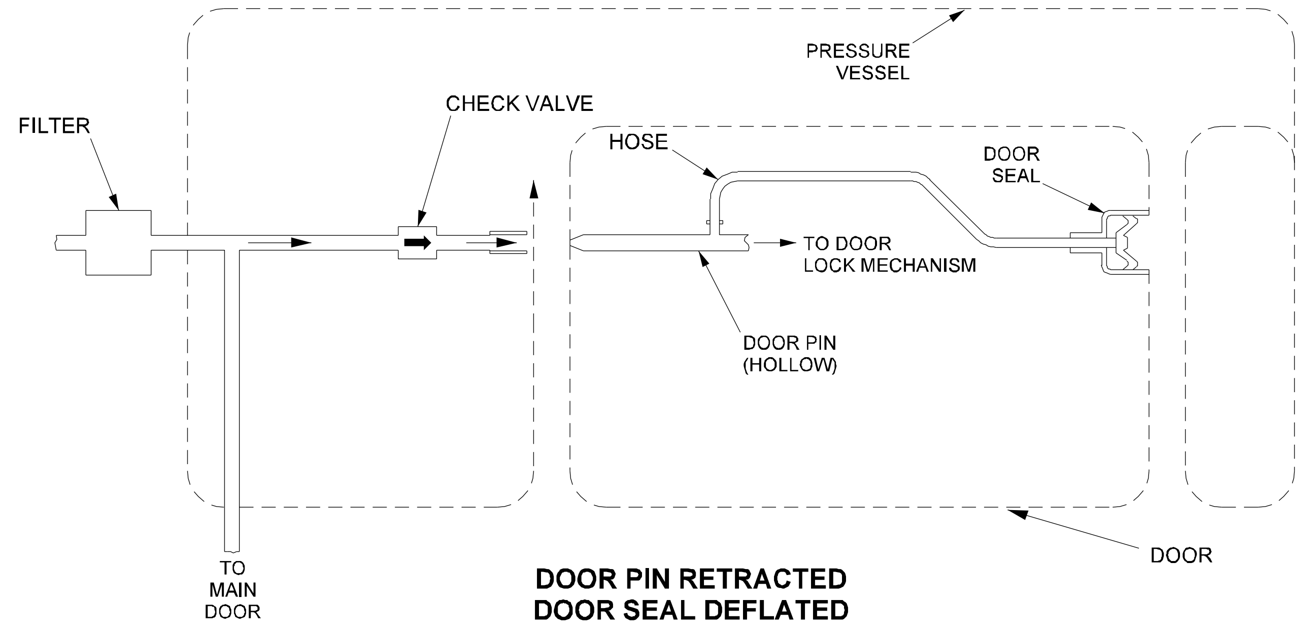
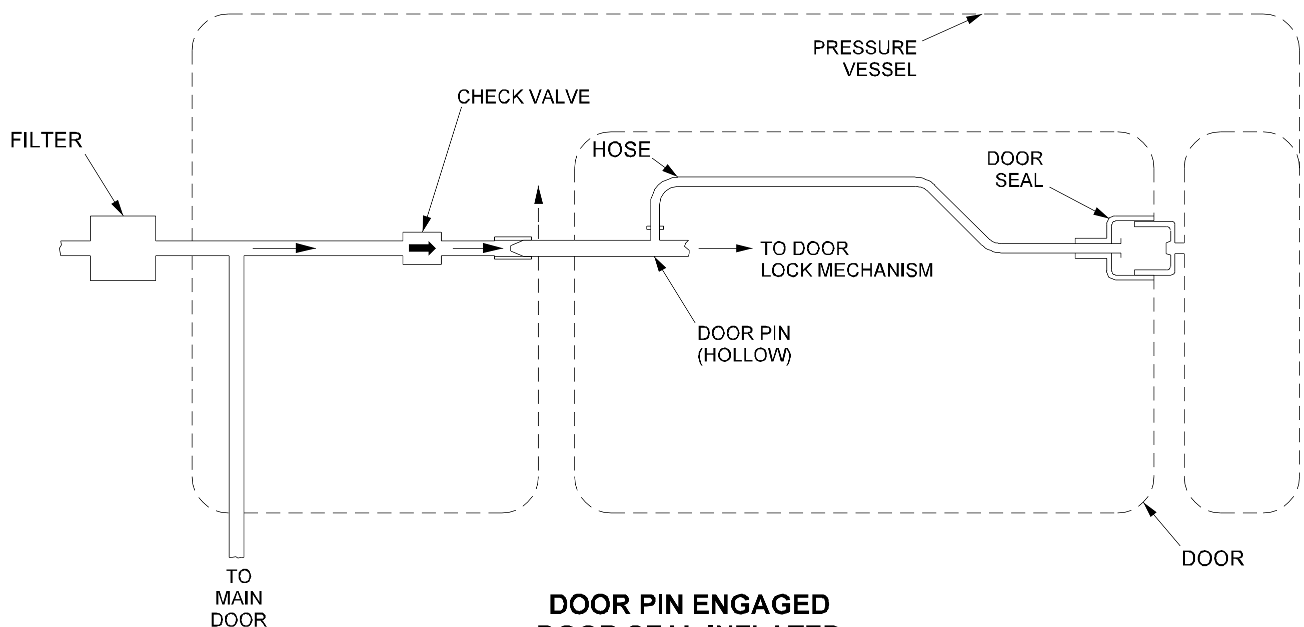
Main Entrance Door Seal System
Figure 6



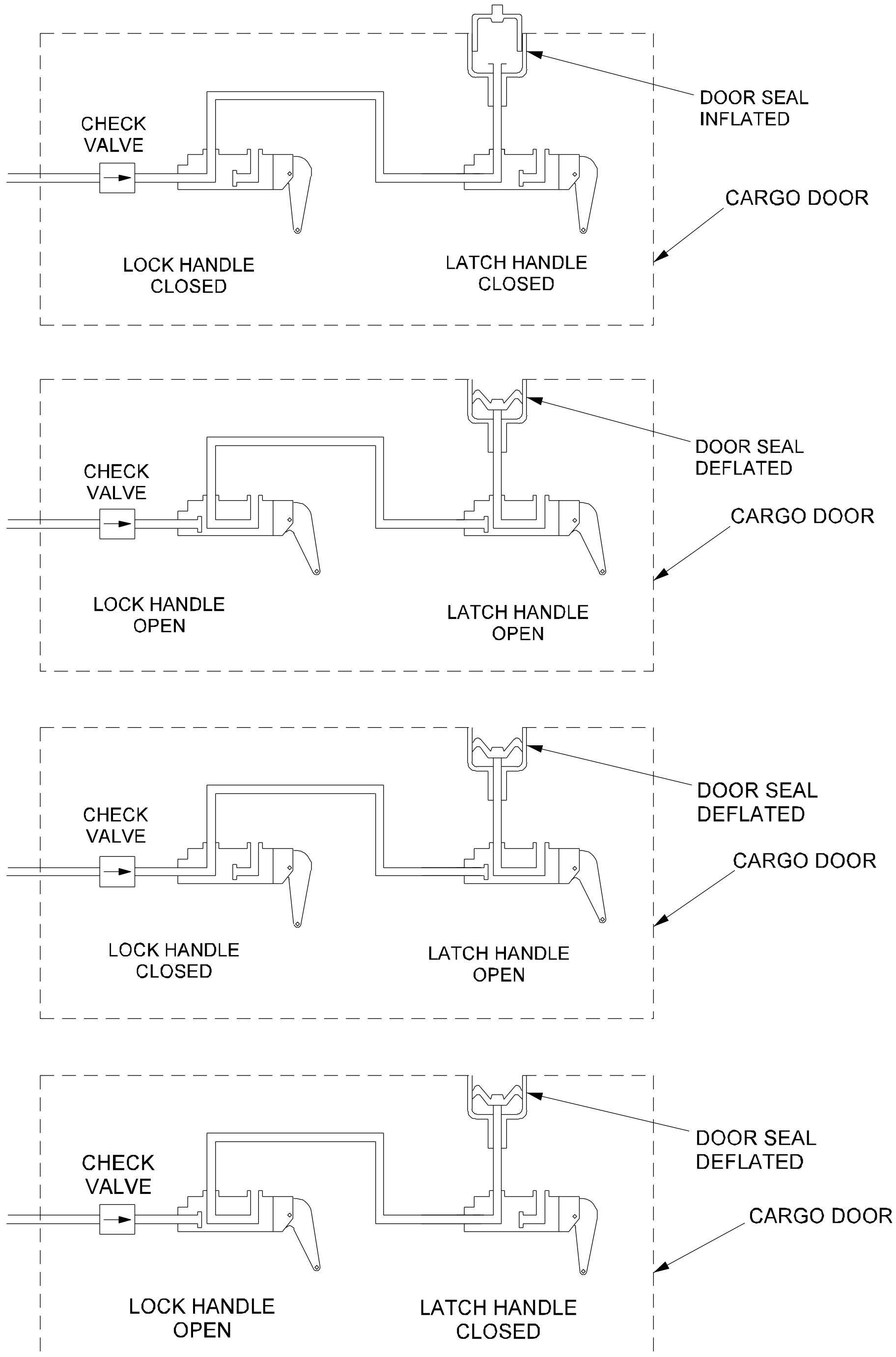
DOOR PIN RETRACTED
DOOR SEAL DEFLATED



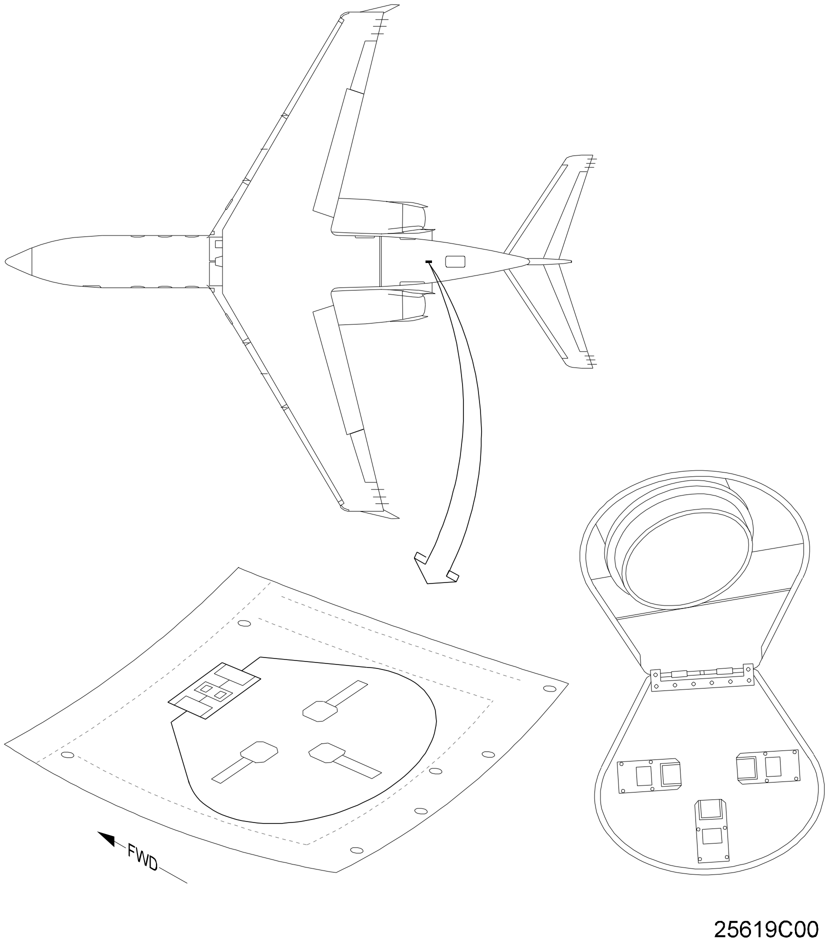
DOOR PIN ENGAGED
DOOR SEAL INFLATED

30968C00

Baggage Door Seal
System
Figure 7



Cargo Door Seal System
Figure 8



External Starting Air Connection
Figure 9

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2A-49-10: General

The Gulfstream IV is equipped with a GTCP 36-100(G) gas turbine Auxiliary Power Unit (APU) that serves as an auxiliary supply of electrical power and pneumatic pressure for airplane systems. The APU is installed in the tail section of the airplane, aft of the pressure bulkhead within a stainless steel and titanium enclosure. The APU is mounted transversely on supports, or rails, for ease of maintenance and repair.

The APU can supply both electrical and pneumatic power on the ground, but is restricted to providing only electrical power in flight. On the ground, the APU may be started with the airplane batteries or with an external DC power source. When the APU reaches normal operating RPM, it is capable of powering the full airplane electrical system and supplying pneumatic bleed air for air conditioning and engine starting. APU bleed air output is rated at a nominal 253°C (487°F) at a pressure of 47.6 psi minimum. Minimum bleed air pressure under full load is 30 psi.

In the air, the APU alternator can produce thirty (30) kVA at twenty-two thousand (22,000) feet, with a linear reduction with altitude to 15 kVA at thirty thousand (30,000) feet for airplanes SN 1000-1155 (except 1034). On airplanes Serial Number (SN) 1156 and subsequent (and 1034) the APU air intake is modified with an airscoop that increases airflow to the APU, allowing a higher operating envelope. For these airplanes, the APU alternator will supply 30 kVA up to 30,000 feet with a linear reduction of sixty-seven percent (67%) to 20 kVA at maximum operating altitude of 35,000 feet. This expansion of the operating envelope requires shedding some of the airplane electrical load. On SN 1156 to 1429 (and 1034) an Electrical Load Warning System (ELWS) provides indications to the crew of excessive electrical loads and automatically sheds some non-essential items to reduce the electrical load on the APU. For airplanes SN 1430 and subsequent, electrical loads are monitored and reduced if necessary by the flight crew in response to Crew Alerting System (CAS) messages. For more information, see the description of the APU AC Power System in section 2A-24-00 of this manual.

The APU assembly is composed of the following subsystems:

- 2A-49-20: APU Powerplant Assembly and Accessories
- 2A-49-30: APU Controls and Indications

2A-49-20: APU Powerplant Assembly and Accessories

1. **APU Powerplant:**

   The APU powerplant assembly is composed of the following subsystems:

   - Basic Powerplant
   - Starting and Ignition
   - Electronic and Fuel Controls
   - Lubrication

   A. **Basic Powerplant:**

      The APU powerplant consists of a single-stage compressor and single-stage turbine mounted on a common rotating shaft. The compressor is surrounded by an air chamber (plenum) that is connected to the air intake duct. The turbine surround receives pressurized air from the compressor, routing some to the combustor where air is mixed with fuel, ignited and directed onto the turbine blades. Some compressed air is reserved for

turbine cooling, flowing around the turbine and joining the exhaust flow prior to venting overboard. See the illustration in Figure 1.

A planetary gearbox is attached to and driven by the rotating compressor section. The gearbox converts the high RPM of the compressor to lower values for driving the oil pump, fuel control and pump and the AC alternator. The gearbox also provides a mounting for the APU starter.

Air for the APU is drawn through an intake door located on the top of the airplane near the vertical stabilizer fairing (with an air scoop on Serial Number (SN) 1034 and 1156 and subsequent). Rotation of the compressor draws in outside air (supplemented by ram pressure for those airplanes equipped with an air scoop). The air flow is accelerated by the rotating compressor and directed through a diffuser that converts air velocity to pressure. Pressurized air is routed to the turbine plenum. The mechanical design of the plenum directs a precise amount of air into the combustion liner where it is mixed with fuel and ignited. The remaining pressurized air is used to cool the APU exhaust and/or extracted through the load control valve for engine starting or air conditioning.

A surge valve, mounted on the turbine section, pneumatically unloads the APU during starting to aid acceleration, and opens in flight (controlled by the nutcracker system) to maintain stable APU RPM. Air bled off by the surge valve is vented into the APU exhaust.

Spent exhaust gases from the APU turbine section are directed through a tailpipe exhaust assembly overboard, venting beneath the right engine nacelle. The tailpipe shroud has a larger diameter than the APU turbine exhaust exit opening. As exhaust gases pass through the shroud, ambient air in the APU compartment is drawn into the tailpipe assembly by the ejector pump action of the gas flow. Ambient air flow decreases the temperature of the APU exhaust and creates a lower pressure within the compartment, causing additional airflow that cools APU components and accessories.

Modifications have been made to the APU ducting and exhaust components to enhance APU component cooling and to moderate the effects of exhaust impingement on the airplane exterior:

- Airplanes SN 1133 and subsequent have a check valve in the alternator cooling duct that draws in air from the aft equipment bay for cooling
- Airplanes SN 1310 and subsequent have additional cooling ducts piped from the air intake manifold to enhance airflow and prevent soot buildup on APU internal components
- Airplanes SN 1310 and subsequent, and prior airplanes having part three (3) of ASC 390 have a venturi in the APU bleed air duct that limits the initial volume of air extracted from the APU for engine start or air conditioning. Limiting the amount of air that may immediately be removed from the compressor diffuser prevents temperature spikes associated with the loss of cooling air in the turbine section.
- Airplanes SN 1311 (sic) and subsequent and prior airplanes having part two (2) of ASC 390 have an exhaust deflector that directs the gases away from the right engine nacelle.

B. Starting and Ignition:

The APU start sequence is initiated by crew activation of the switchlight controls on the APU panel on the cockpit overhead. When the APU MASTER switchlight is selected ON, the APU air inlet opens, the APU OIL PRESS LOW circuit is armed and the APU fuel shutoff valve is opened. When the APU air inlet door is fully open, the APU START switchlight is armed. Pressing the APU START switch powers the APU starter from the airplane batteries or external DC power source, and initiates the automatic start sequence controlled by the Electronic Control Unit or ECU (see the description of the start sequence in the Controls and Indications topic of this section).

The APU starter, mounted on the gearcase, is a DC powered electric motor that spins the APU up to a speed sufficient to support ignition and light off, approximately sixty percent (60%) RPM. A clutch delivers starter torque to the gearbox. When the APU RPM reaches approximately 60%, DC power is removed from the starter, the clutch disengages the starter from the gearbox, and the starter slows to a stop.

It is important that the crew have an indication of APU starter disengagement. If the starter remains engaged and the APU continues to accelerate to 100% RPM, damage and/or catastrophic failure of the starter may occur. In initial production of the GIV airplanes, there was no indicator monitoring DC power application to the starter. ASC 13, applicable to SN 1000-1155, and incorporated into production assembly with SN 1156 and subsequent, provided circuitry to the APU START switchlight to illuminate the ON legend whenever DC power is applied to the starter. Starter cutout can be monitored by confirming that the ON legend extinguishes above 60% RPM. ASC 212 (applicable to airplanes SN 1000-1155 having ASC 13) and ASC 212 AM1 (applicable to airplanes SN 1156 and subsequent) improved on the reliability of the monitoring circuit by powering the circuit from the Essential DC bus and adding auxiliary contacts to the circuit. (The circuitry installed with ASC 13 and incorporated in SN 1156 was powered by the Battery Tie bus that is de-energized when the APU reaches 60% RPM, and thus could not provide starter monitoring should the starter remain powered at higher APU speeds.)

The ECU sequences the start process initiated with the APU START switchlight. RPM is displayed on the APU control panel as soon as the starter begins rotating the APU gearbox. At 10% RPM, the fuel control opens, injecting atomized fuel into the combustor and the igniter is energized.

The igniter is a high energy capacitive discharge unit consisting of a single plug with a long shank protected by a heat shield. The plug is mounted on either the turbine plenum or the left side of the turbine intake (depending on APU model). The igniter plug produces approximately five thousand (5,000) volts and remains energized until the APU reaches 95% RPM with an additional four (4) second delay to ensure stable APU idle RPM. There is no indication of ignitor cut-out. A high-voltage APU ignition system is installed on airplanes SN 1363 and subsequent, and available for retrofit into pervious airplanes as ASC 404. The higher voltage increases service life and improves in-flight starting.

C. Electronic and Fuel Controls:

APU operation is monitored and controlled by the ECU through inputs from the cockpit control panel, APU speed sensing and temperature elements and outputs to the Fuel Control Unit (FCU). The ECU is a solid state electronic control mounted outside of the APU enclosure to minimize exposure to heat and vibration. On airplanes SN 1436 and subsequent, a second ECU is installed as a "cold spare", making a replacement ECU readily available if needed.

The unit receives RPM information from a speed sensing monopole device installed on the APU gearbox. The monopole is a cylindrical pointer with a magnetically charged tip. The tip is closely aligned a rotating gear train in the gearbox. As the teeth of the gear pass the tip of the monopole, an AC electric current is generated with a frequency proportional to gear rotation (and APU) speed. At an APU RPM of 100%, the frequency is approximately 17,600 Hz. The ECU senses this frequency as APU RPM.

The operating temperature of the APU is sensed by a single thermocouple positioned in the APU exhaust stream aft of the turbine. The thermocouple generates a voltage proportional to Exhaust Gas Temperature (EGT) and transmits the voltage to the ECU where it is translated as EGT.

The ECU uses the RPM and EGT inputs to correctly sequence APU starting, adjust the fuel flow through the FCU to maintain the APU within normal operating ranges, and with additional inputs from oil temperature, oil pressure and amperage readings from system sub-components, protect the APU with an auto-shutdown feature if abnormal conditions are detected. The conditions prompting and automatic APU shutdown are discussed in the Controls and Indications topic of this section.

The ECU controls the speed and temperature of the APU by adjusting the amount of fuel available for combustion in the APU (ignition is not necessary above 95% RPM). Fuel for the APU is drawn from the left fuel tank hopper through a shutoff valve. The left tank boost pump (or the right tank boost pump with the crossflow open) pressurizes the fuel line. When APU RPM reaches 10% during start, the shutoff valve opens and fuel passes through the fuel line to the APU FCU.

The FCU is a high pressure gear pump attached to and driven by the APU oil pump. Within the FCU are filters, a relief valve, a servo valve to maintain constant fuel pressure, a metering valve, a shutoff valve and a fuel nozzle. Fuel drawn from the left wing tank is filtered, pressurized, metered to satisfy APU load requirements, passed through the shutoff valve and sprayed into the combustor through a dual orifice nozzle. During APU start, fuel is initially supplied through the primary orifice. As the APU accelerates, fuel pressure increases with the increasing speed of the gearbox (and oil pump rotation and pressure) and fuel requirements of the APU increase. The secondary orifice opens to supply additional fuel to support APU acceleration and maintain full RPM.

The protective functions of the ECU shut down the APU by closing the FCU shutoff valve (positioned just prior to the fuel nozzle) if normal limitations are exceeded. Closure of the shutoff valve causes a surge in fuel pressure. The relief valve opens with the pressure surge and routes the excess fuel back to the pump inlet. As APU RPM slows from fuel starvation, fuel pressure decreases and the relief valve closes.

D. Lubrication:

The oil pump that drives the FCU fuel pump supplies pressure and splash lubrication to all gears, shafts and bearings within the APU. The oil pump draws oil from a sump beneath the pump through a suction line. Oil passes through a pressure regulating valve, an oil filter and a low oil pressure switch before distribution to the APU lubrication points. The oil is then gravity scavenged back to the oil sump. If APU oil pressure drops below thirty-one (31) psi, the low oil pressure switch will cause the illumination of the red OIL PRESS LOW light on the APU control panel. Cooling is provided by external cooling fins on the sump that transfer heat to the surrounding air. The oil sump (or tank) has a high oil temperature switch and a drain plug with a magnetic chip detector. If cooling air is insufficient and the APU oil temperature reaches 141°C - 147°C, the red OIL TEMP HIGH light will illuminate on the APU control panel. APU oil level is read on a dipstick attached to the sump (tank) filler opening cap. Normal capacity is 2.5 U.S. quarts. For airplanes SN 1000 and subsequent not having ASC 462, APU access panel 106-APU-2 must be removed in order to access the dipstick / filler cap. For airplanes SN 1000 and subsequent having ASC 462, an access panel is installed on panel 106-APU-2 to provide convenient access to the dipstick / filler cap.

2. APU Accessories:

The APU is equipped with the following accessories:

- Electrical Alternator
- Bleed Air System
- Fire Detection, Warning and Extinguishing System

A. Electrical Alternator:

The APU alternator is identical and interchangeable with the engine-driven alternators. It is mounted on the APU gearbox and cooled by an integrated fan. The alternator delivers 115 V AC, 400 Hz (±20) three phase power at loads up to 30 kVA, enabling the alternator to power all airplane electrical systems. Because the APU alternator runs at a constant speed (approximately 8,000 RPM) it does not require a converter and can power airplane buses directly.

The alternator can provide electrical power in flight as well as on the ground, however the amount of load (kVA) available from the alternator decreases at higher altitudes due to APU performance degradation with decreasing air density. For airplanes SN 1000-1155, the available load decreases linearly from 30 kVA to 15 kVA from 22,000 feet to 30,000 feet. On airplanes SN 1156 and subsequent the APU air intake is modified with an air scoop to provide more air to the APU at higher altitudes. For these airplanes, 30 kVA is available up to 30,000 feet with a reduction to 20 kVA at 35,000 feet, the maximum APU operating altitude.

An APU ALTNR OFF light (blue) on the APU overhead panel illuminates when the APU has reached 95% RPM for four (4) seconds, signalling that the alternator may be selected ON to power airplane systems. The APU alternator is connected to the airplane electrical system by pressing the AUX PWR switchlight on the ELECTRIC MASTER section of the ELECTRIC POWER MONITOR panel to ON. The voltage and frequency of the APU alternator may then be monitored by pressing the AUX switchlight

next to the AC VOLTS / FREQ display. The AUX PWR % display on the AC POWER CONTROL should read 100 with the APU alternator powering all airplane buses.

Operation of the APU alternator is monitored by the same self-protective functions as the engine-driven alternators. The APU alternator will trip off for over and undervoltage, over and under frequency, polarity, overcurrent, short circuit and feeder fault malfunctions. Like the engine-driven alternators, the APU alternator has dual bearings. If the main bearing fails, the APU ALT FAILED BRG light on the overhead annunciator panel will illuminate accompanied by a CAS message. If the APU alternator exceeds normal temperature limits (300°F / 149°C), a CAS message of APU ALT HOT will be displayed.

B. Bleed Air System:

Bleed air may be extracted from the APU compressor for engine starting or operating the air conditioning system while the airplane is on the ground (the APU cannot be used for bleed air while the airplane is in flight). After starting the APU and RPM has reached at least 95% for four (4) seconds, and the EGT has stabilized for one (1) minute, the APU switchlight on the BLEED AIR panel may be selected ON. After six (6) seconds, verify the ISOLATION valve is OPEN. Maximum APU EGT with APU BLEED AIR selected ON is 680°C. When the switch is selected, a signal is sent to the ECU to open the load control valve, porting compressed air into the pneumatic manifold. When air is bled from the APU compressor, less is available to cool the turbine section of the APU. For this reason, the load control valve does not open immediately, but takes one to one and a half (1-1½) seconds to fully open, allowing the ECU to adjust fuel flow to avoid EGT spikes. On airplanes SN 1000-1310 having part 3 of ASC 390 and production airplanes SN 1311 and subsequent, a venturi is installed downstream of the load control valve to further moderate the initial air extraction flow and the rate of EGT rise.

To avoid reverse air flow into the APU that would disrupt the path of cooling air around the turbine, a check valve is installed in the pneumatic system ducting that prevents the flow of engine bleed air back to the APU. An interlock is also incorporated that closes the load control valve if the APU air bleed is open and engine air bleeds are open and engine power levers are advanced from the idle position. In normal circumstances only one bleed air source should be selected at a time (either APU or engine bleed).

The ECU maintains the APU at normal operating temperatures (665°C and 100% RPM) during bleed air extraction by balancing fuel flow with air bleed demands. The ECU can add fuel flow to maintain operating temperatures, but can reduce EGT only by reducing the amount of air bled from the APU (since RPM cannot by reduced), thereby increasing compressor discharge air available to cool the turbine section. Operation of the load control valve and ECU result in the following indications:

- Initial selection of APU ON for bleed air on the BLEED AIR panel will result in an initial rise in APU EGT until the ECU balances fuel flow
- The APU can deliver more bleed air pressure on cold days than on hot days
- If APU EGT rises above normal, the ECU will start to close the load control valve (if open) to cool the APU. If EGT reaches 732°C, the

load control valve will be fully closed and no air will be available from the APU

C. **Fire Detection, Warning and Extinguishing System:**

There are four fire detectors positioned around the APU. Three of the detectors are installed in the walls of the APU enclosure and one is installed in the APU air inlet. The detector installed on the top of the turbine section near the load control valve is set at a trip point of 600°F, the other three are set at 450°F. The fire detection and extinguishing system is powered by 28V DC, and will operate from airplane batteries in order to test the system prior to starting the APU. If the temperature within the APU enclosure reaches a detector trip point, a ground is furnished to the detection circuits and the following indications are displayed:

- Red APU FIRE light on the APU overhead panel
- Red APU FIRE light on the Standby Warning Lights Panel for SPZ-8000 airplanes
- APU FIRE message on CAS
- Both red MASTER WARNING lights illuminate
- The APU FIRE checklist is displayed on the lower center portion of the copilot's navigation display (DU 5) in MAP or COMP or PLAN mode (airplanes SN 1144 and subsequent and SN 1000 through 1143 with ASC 178 incorporated)
- A warning bell / tone sounds in the nose wheel well if the airplane is on the ground

The ECU initiates an automatic shutdown sequence that accomplishes the following:

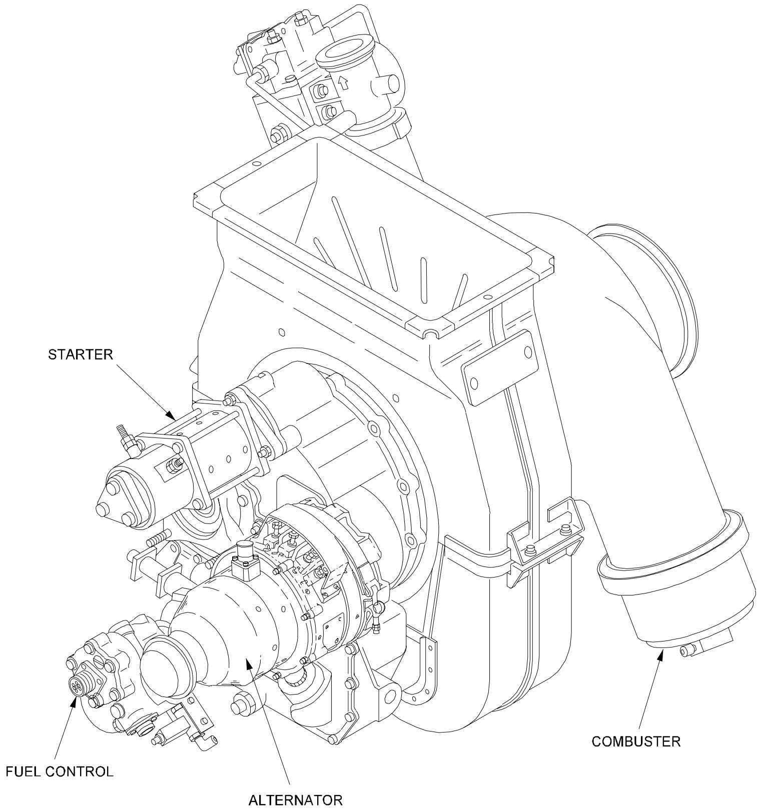
- The APU shutdown relay is activated and held closed by the fire detection circuit
- Fuel is shut off at the ECU
- The air inlet door closes, de-energizing the APU MASTER switch
- Fuel valve closes at the left wing tank

The APU fire extinguisher must be manually discharged with the switch on the APU overhead panel. When the switch is pushed, the contents of the APU fire bottle are discharged into the APU enclosure. The bottle contains 2½ lbs. of CF_3Br pressurized with nitrogen gas. Normal bottle pressure at 70°F is 600 psi (+50/-25). The bottle is a single use installation - the entire bottle contents are discharged if the APU FIRE EXT switch is pushed. A confirmation that the bottle has discharged is shown in the legend of the switch that will illuminate FIRE EXT DSCHD when the bottle is depleted.

A test of the APU fire detection system is performed prior to starting the APU. The test is initiated by pressing the APU TEST switch on the FIRE TEST panel on the cockpit overhead. During the fire test, all of the indications listed above will be displayed.

NOTE:

Pushing the APU FIRE TEST switch while the APU is running will shut down the APU.



Auxiliary Power Unit
Figure 1

2A-49-30: APU Controls and Indications

1. Cockpit APU Controls and Indications:

Controls and indications for the APU are installed on five panels in the cockpit. Illustrations of the panels are shown in Figure 2 and in sections 2A-24-00, 2A-26-00 and 2A-36-00 of this manual.

A. APU Panel:

The APU panel has the following controls and indications:

- APU MASTER - alternate action switch illuminates ON in white when selected to connect battery power to the APU for starting and open APU air inlet. When the air inlet is fully open, a limit switch

routes power to the ECU, opens the fuel shutoff valve in the left wing tank, arms the APU START switch and overspeed switch, and illuminates of the red OIL PRESS LOW light signalling that the APU is ready to start. On airplanes SN 1096 and subsequent, and SN 1041-1095 having ASC 119, aborting an APU start prior to 60% RPM by selecting the APU MASTER switch off will disengage the APU starter. On all other airplanes, the OVSP TEST - STOP switch must be used to disengage the starter if a start is discontinued prior to the APU reaching 60% RPM.

- START - momentary action switch illuminates ON in white when power is applied to the APU starter initiating the start sequence controlled by the ECU. The start may by monitored by observing the EGT and RPM readouts on the APU panel. For airplanes not having ASCs 13, 212 and 212 AM1, the ON indication only monitors the APU start relay. For airplanes SN 1000-1155 having ASCs 13 and 212 and SN 1156 and subsequent having ASC 212 AM1, the ON light monitors actual DC power application to the APU starter. The ON light should extinguish at 60% RPM, normal starter cutout speed.
- OVSP TEST-STOP - momentary action switch illuminates TEST in white when selected. Pushing switch sends an overspeed signal of 114% RPM to the ECU, prompting the ECU to shut down the APU. Automatic overspeed shutdown is commanded by the ECU at 110% RPM. This switch is the normal means of shutting down the APU.

  When the OVSP TEST-STOP switch is pressed and APU RPM is below 10%, the APU fire shutoff relay removes power from the entire system. This causes the OIL PRESS LOW light to extinguish, the air inlet door to close and the emergency fuel shutoff valve (to the APU) to close. If the OVSP TEST-STOP switch is pressed and APU RPM is above 10%, the overspeed shutoff relay sends an overspeed signal to the ECU. This operation will not remove all power from the system, but it will cause the emergency fuel shutoff valve to close. The OIL PRESS LOW light will remain illuminated until the APU MASTER switch is placed in the OFF position.
- APU FIRE EXT - guarded single action switch illuminates amber text FIRE EXT DISCHD after APU fire bottle has been used
- APU FIRE - light capsule illuminates APU FIRE in red if a fire is detected in the APU enclosure. ECU automatically shuts down the APU.
- OIL TEMP HIGH - light capsule illuminates OIL TEMP HIGH in red and ECU automatically shuts down the APU if oil temperature exceeds 141°- 147°C. APU MASTER switch must be on for the warning to be armed.
- OIL PRESS LOW - light capsule illuminates OIL PRESS LOW in red when APU air inlet door is fully open and APU is ready for start. When APU is operating (RPM above 95%) the light will illuminate and the ECU automatically shuts down the APU if oil pressure falls below 31 psi for longer than 10 (±2) seconds. APU MASTER switch must be on for the warning to be armed.
- ALTNR OFF - light capsule illuminates ALTNR OFF in white when

APU alternator is available for electric power (APU RPM at 95% or above for more than 4 seconds) and AUX PWR switch on the ELECTRIC MASTER panel has not been selected. When AUX PWR is selected on the ELECTRIC MASTER panel, the ALTNR OFF light on the APU panel will extinguish.

- EGT - digital readout of APU EGT in °C displayed in amber numerals. An amber warning circular light above the readout comes on if APU is at 100% RPM and EGT reaches 688°C.
- RPM - digital readout of APU RPM displayed in amber numerals. An amber warning circular light above the readout comes on if APU RPM exceeds 104%.

B. BLEED AIR Panel:

Selecting APU bleed air for engine starting and air conditioning (ground operation only) is accomplished by pressing the APU switch on the BLEED AIR panel:

- APU - alternate action switch, signals ECU to open the APU load control valve (after APU has reached a stabilized RPM of 95% or more for at least 4 seconds). Indications of load control valve operation are ON (amber) indicated in the switchlight, a rise in APU EGT and readout of APU bleed air pressure on the BLEED AIR panel. Selecting APU bleed air ON will also open the ISOLATION valve in the pneumatic system to allow APU bleed air to pressurize both sides of the system. For more information regarding APU bleed air and the airplane pneumatic system, see section 2A-36-00 of this manual.

NOTE:

Both engine power levers must be at the idle position (whether engines are operating or not) for the load control valve to open and supply bleed air to the pneumatic system.

C. ELECTRIC MASTER Panel:

Selecting the APU alternator to power the airplane electrical systems is accomplished by pressing the AUX PWR switch on the ELECTRIC MASTER panel:

- AUX PWR - pressing the switchlight (with APU RPM stabilized at 95% or more for at least 4 seconds) connects APU alternator AC power to the airplane electrical system. ON will illuminate in amber above the switch and APU alternator frequency, voltage and load percentage may be read on the ELECTRIC POWER MONITOR panel.

D. Standby Warning Lights Panel (SPZ-8000 Equipped Airplanes):

- APU ALT FAILED BRG - light capsule illuminates amber when the main bearing of the APU alternator has failed and the APU is operating on the alternate bearing. APU operation using the alternate bearing is limited to fifteen (15) hours at full load or fifty (50) hours with no load (air bleed only).

E. Electrical Load Warning System (ELWS):

For airplanes SN 1156-1429 with the ELWS installed, a gauge on the lower pilot side forward instrument panel indicates the electrical load of the APU when it is used to power the electrical system in flight. The gauge aids the flight crew in monitoring the APU alternator when operating near the limits of the APU altitude envelope. The ELWS contains a processor unit, called the Electrical Load Warning Computer (ELWC) that uses inputs from the Digital Air Data Computers (DADCs) to aid in configuring the airplane electrical load to meet APU alternator capabilities. If the APU alternator is powering the right AC electrical bus and the airplane altitude exceeds 34,000 feet, the ELWS will load shed heat to the right windshield controller. If the APU is powering the left AC electrical bus, the left windshield heat controller will be shed above 34,000 feet. If the APU alternator is powering the right AC electrical bus and the left converter / alternator fails (single alternator operation), the ELWS will shed the galley master electrical load in addition to the right windshield heat controller if airplane altitude exceeds 34,000 feet. See the description of the ELWS and illustration of the gauge in section 2A-24-00 of this manual.

F. Forward Instrument Panel:

Two APU electrical load indicators, shown in Figure 3, are included in the warning/caution lights panels located above the EFIS display units 2 and 5 on the pilot and copilot forward instrument panels. The indicators signal APU electrical load and/or load monitoring abnormalities when using the APU at high altitudes (above 30,000 feet). For more information, see the discussion in section 2A-24-00 of this manual.

- APU LOAD (amber) / APU LOAD (red) - is a split annunciator installed on airplanes SN 1156-1252 equipped with SPZ-8000 DAFCS. The upper half contains the red text annunciation of APU LOAD that is illuminated if electrical loads exceed limits and must be reduced when operating above 30,000 feet. The lower half contains the amber text annunciation of APU LOAD that illuminates when APU electrical loads are nearing limits when at high altitudes.
- ELWS FAIL (amber) / ELWS CONFIG (amber) - is a split annunciator installed on airplanes SN 1034, and 1156-1429. The upper half contains the amber text annunciation of ELWS FAIL that is illuminated when the system is unreliable for monitoring APU electrical loads. The lower half contains the amber text ELWS CONFIG that illuminates when the Electrical Power Monitoring Panel (EPMP) is not properly configured for ELWS operation.

2. Tail Compartment Controls and Indications:

A. APU Remote Failure Indicator Box:

A remote failure indicator box for the APU is mounted in the tail compartment to facilitate maintenance on the APU. The box has magnetic “cat’s eye” indicators that are activated whenever a malfunction causes an automatic shutdown of the APU. The box has indicators for:

- H-OIL TEMP - high oil temperature auto-shutdown
- L-OIL PRESS - low oil pressure auto-shutdown
- OV CUR - auto-shutdown caused by an overcurrent in the APU ECU circuitry (not the alternator)

- OV TEMP - high EGT auto-shutdown

The indicator box also has two toggle switches. One toggle switch resets the failure indicators, the other toggle switch shuts down the APU using the same overspeed signal as the OVSP TEST-STOP switch on the cockpit APU control panel.

B. Hourmeter:

An hourmeter is installed in the tail compartment near the APU enclosure (location varies) to log the operating time of the APU. The meter is activated when the APU RPM reaches 95% and displays the cumulative time of APU operation. The meter is used to determine when to perform required maintenance checks that are specified after a certain number of APU operating hours.

3. APU Start Sequence:

During an APU start, the following indications are normally observed (it is assumed that normal preflight and prestart procedures outlined in section 03-02-00 of this manual have been accomplished):

- APU MASTER switch ON - air inlet door opens, fuel shutoff valve in left wing tank opens, and, when air inlet fully open, OIL PRESS LOW indicator illuminated
- START switch depressed and held until RPM rise indicated, switch is then held in electrically
- At 10% RPM, APU fuel valve opens and ignition is powered, EGT rises
- At 25% RPM, an acceleration timer started by ECU. Timer monitors start for 16 seconds or until RPM reaches 95% for 4 seconds
- At 30% RPM, OIL PRESS LOW light should extinguish and oil pressure should indicate approximately 40 psi
- Acceleration up to 60%, EGT should not exceed 988°C
- At 60% RPM, the START switch disengages, indicated by the ON legend in switch extinguishing, surge valve opens, EGT begins decreasing
- At 95% RPM, after a 4 second ECU initiated delay, ignition is terminated, the bleed air load control valve and alternator are available, and the hourmeter begins tracking APU operating hours
- At 100% (±3%), EGT should stabilize at approximately 665°C. After RPM has stabilized for 1 to 2 minutes, electrical and pneumatic loads may be placed on the APU.

4. CAS Messages:

A. Warning (Red) CAS Messages:

CAS Message	Cause or Meaning
APU FIRE	APU fire detected
APU LOAD **(1)**	APU electrical load not within limits for operation above 30,000 feet

(1) **(1)** SN 1034, 1156-1252 with SPZ-8400 and 1253-1429

B. Caution (Amber) CAS Messages:

CAS Message	Cause or Meaning
APU ALT BRG FAIL	Main bearing of APU alternator has failed and alternator is operating using the auxiliary bearing
APU ALT HOT	APU alternator above 300°F /149°C
APU LOAD **(1)**	APU electrical load not within limits for operation above 30,000 feet
APU MASTER WARN	APU MASTER switch in the ON position but APU is not running
AUX AC POWER FAIL	APU alternator has failed or dropped off line

(1) **(1)** SN 1034, 1156-1252 with SPZ-8400 and 1253-1429

C. Advisory (Blue) CAS Messages:

CAS Message	Cause or Meaning
APU ALT OFF	AUX PWR switch on Electric Master panel not selected ON with APU alternator operating
APU EXCEEDANCE	FWC has recorded an APU operating limit exceedance

5. Circuit Breakers (CBs):

Circuit Breaker Name	CB Panel	Location	Power Source
APU CONT	P	G-10 **(1)** L-9 **(2)**	BATT #1 BUS B / BATT #2 BUS B / ESS 28V DC
APU START	P	H-10	ESS 28V DC
APU PWR #1	P	J-10	BATT #1 BUS B
APU PWR #2	P	K-10	BATT #2 BUS B
APU PWR #3	P	L-10	ESS 28V DC
APU FIRE WARN	P	B-4 **(3)** A-4 **(4)**	ESS 28V DC
APU FIRE EXT	P	I-10	ESS 28V DC
AIR INLET DOOR	PO	C-11	ESS 28V DC
ELWS #1 **(5)**	PO	A-9	ESS 28V DC **(6)**
ELWS #2 **(7)**	PO	B–9	ESS 28V DC **(8)**

(1) **(1)** SN 1001, 1018 & subs
(2) SN 1000, 1002-1017
(3) SN 1280-1309, 1436 & subs
(4) SN 1000-1279, 1310-1435
(5) SN 1034, 1156-1429
(6) L MAIN 28V DC for SN 1183-1207
(7) SN 1034, 1156-1429
(8) R MAIN 28V DC for SN 1183-1207

6. Flight Manual Limitations:

A. APU Operating Limits:

(1) General:

The APU can be operated on the ground, during takeoff, in flight and during landing. In flight it is an optional source of electrical power via the AUX PWR switch instead of one or both engine-driven alternators. The APU cannot be used to supply pressurization airflow in flight.

(2) Maximum Permissible EGT – Airplanes Not Having ASC 465 (36-150[G] APU):

- Up to 60% RPM during start: 988°C
- 60% to 100% RPM during start: 821°C to 732°C (linear decrease)
- Running: 732°C

(3) Maximum Permissible EGT – Airplanes Having ASC 465:

- Up to 50% RPM during start: 973°C
- 51% to 87% RPM during start: 973°C to 732°C (linear decrease)
- 87% to 100% RPM during start: 732°C
- Running: 732°C

(4) Maximum Rotor Speed:

The maximum rotor speed for all conditions is 110%.

B. APU Starting Limits:

(1) When Powered By Airplane Batteries:

Continuous operation of the APU starter when powered by airplane batteries is limited to thirty (30) seconds per start with a maximum of three (3) consecutive start attempts. Before attempting another start, allow twenty (20) minutes for starter cool-down. Three (3) additional start attempts may be made, after which a one (1) hour cool down period must be observed before the next full starter cycle is commenced.

(2) When Powered By An External DC Power Source:

Continuous operation of the APU starter when powered by an external DC power source is limited to fifteen (15) seconds per start with a maximum of two (2) consecutive start attempts. Before attempting another start, allow twenty (20) minutes for starter cool-down. Two (2) additional start attempts may be made and, if unsuccessful, a one (1) hour cool down period must be observed before the next full starter cycle is commenced.

C. APU Altitude / Airspeed Envelope:

(1) SN 1000 thru 1155 Without APU Loadmeter

	Guaranteed Starting:	**Guaranteed Running:**
Altitude:	15,000 Feet and Below	30,000 Feet and Below
Airspeed:	250 KCAS Maximum	V_{mo} / M_{mo}

(2) SN 1000 – 1155 With APU Loadmeter Installed (or Removed by ASC 420); SN 1156 and Subs, SN 1430 and Subs, Airplanes Having ASC 465 (36-150[G] APU)

	Guaranteed Starting:	**Guaranteed Running:**
Altitude:	15,000 Feet and Below	35,000 Feet and Below
Airspeed:	250 KCAS Maximum	V_{mo} / M_{mo}

See Figure 7 or Figure 5.

NOTE:

An inspection is required within ten (10) APU operating hours if the APU is operated above 30,000 ft for more than one (1) hour during flight, or if the APU is operated above 30,000 ft more than five (5) times. Refer to the APU Maintenance Manual for specific inspection requirements.

NOTE:

Successful consecutive starts are limited to six (6) at ten (10) minute intervals per start.

D. APU Alternator Electrical Load Envelope:

(1) The following limitation applies for GIV airplanes SN 1480 and subs, SN 1156 through 1309 with Loadmeter installed but with or without ASC 470, and for SN 1000 through SN 1479 having either:

- APU Loadmeter installed, **or**:
- ASC 420 (APU Loadmeter Removal)

OR:

(2) ASC 427 (APU Enclosure Sooting Mod) AND ALSO HAVING EITHER:

- APU Loadmeter installed, **or**:
- ASC 420 (APU Loadmeter Removal)

(3) The following limitation applies for GIV SN 1000 through 1309 without APU Loadmeter and with or without ASC 470:

The APU alternator can deliver 100% electrical power (30 kVA) on ground or in-flight from Sea Level to 25,000 feet. From 25,000 feet to 30,000 feet, the limit load decreases linearly to 83% (25 kVA). From 30,000 feet to 35,000 feet the limit load decreases to 67% electrical power (20 kVA). Load shedding may be required. See Figure 4.

(4) The following limitation applies for GIV airplanes having ASC 465 (36-150[G] APU):

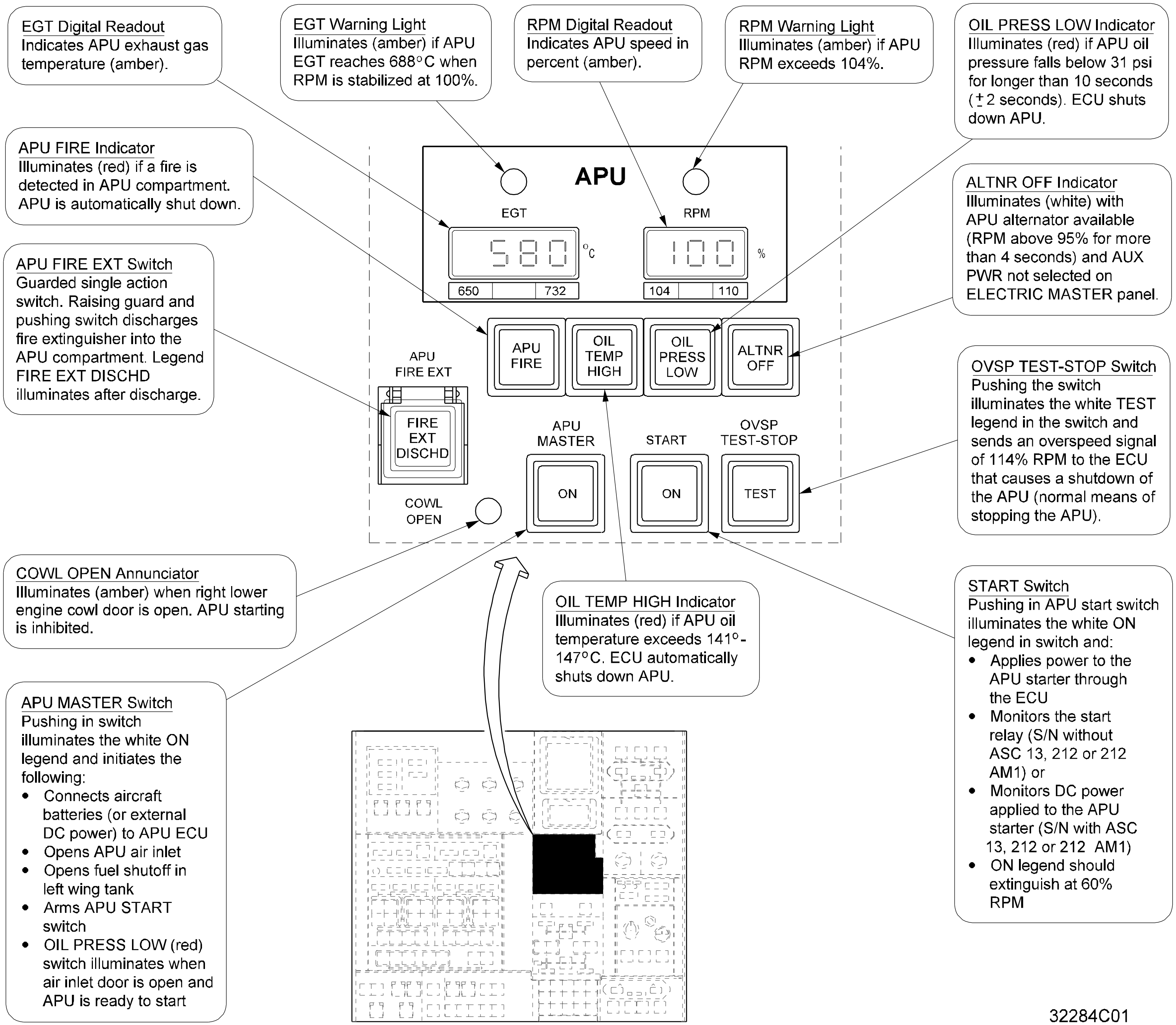
The APU alternator can deliver 100% electrical power (30 kVA) on the ground or inflight from Sea Level to 15,000 feet (20,000 feet if airspeed is maintained below 300 KIAS). From 15,000 feet to 30,000 feet the limit load is 75% (22.5 kVA). From 30,000 feet to 35,000 feet the limit load is 50% (15 kVA). Load shedding may be required. See Figure 6.

(5) The following limitation applies for all other GIV airplanes (SN 1000 – 1155 without APU loadmeter):

The APU alternator can deliver 100% electrical power (30 kVA) on ground or in-flight from Sea Level to 22,000 feet. From 22,000 feet to 30,000 feet, the limit load decreases linearly to 50% (15 kVA). See Figure 8.

7. System Notes:

If the APU is started with the right lower engine cowling open, the high temperature APU exhaust will cause significant damage to the engine cowling. A visual inspection of the area surrounding the right engine should be accomplished prior to starting the APU. On airplanes SN 1455 and subsequent, an amber COWL OPEN annunciator is installed on the APU control panel. With the annunciator illuminated, APU starting is inhibited until the right lower engine cowling is closed. On airplanes SN 1000 through 1454, this feature was available for installation as an option during outfitting.



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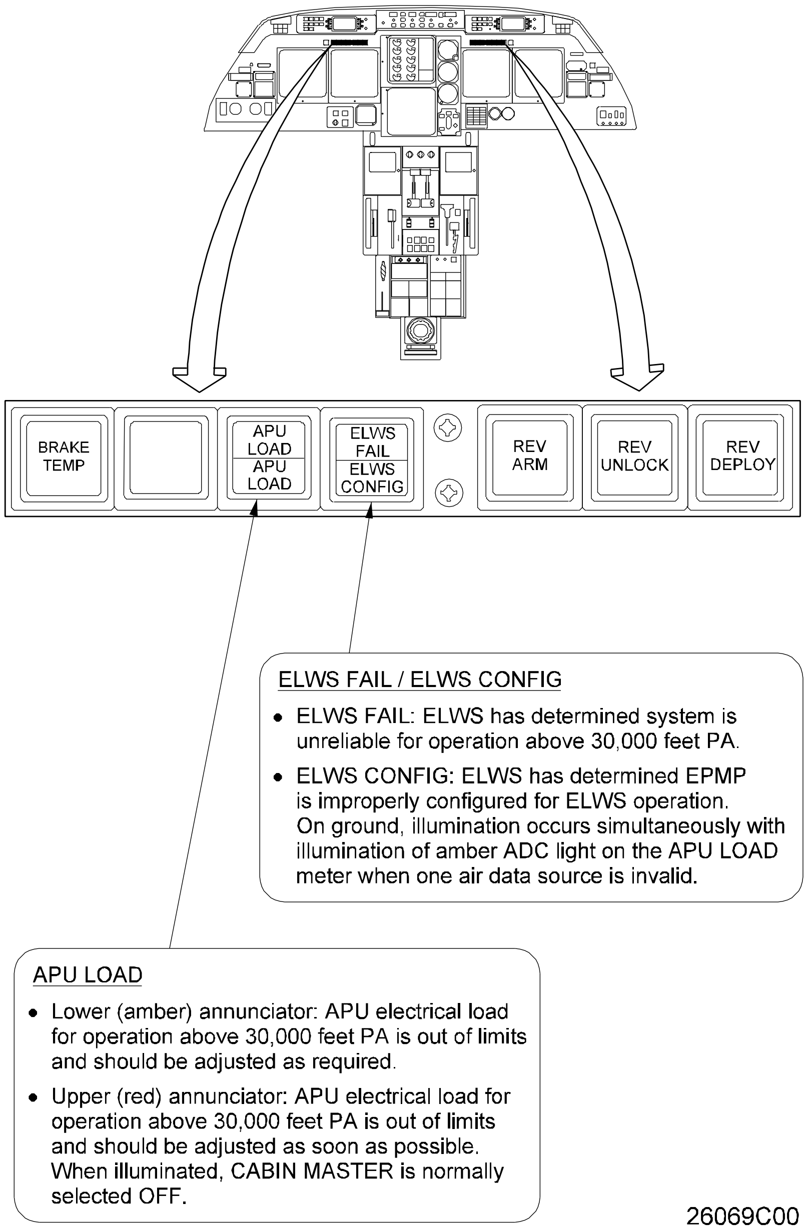
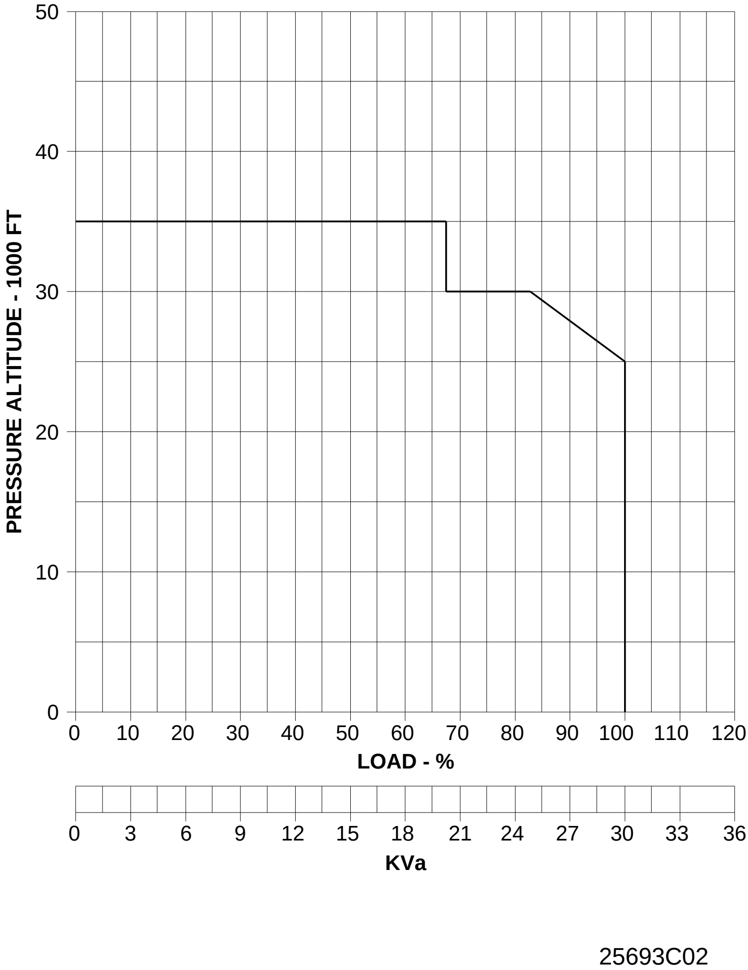
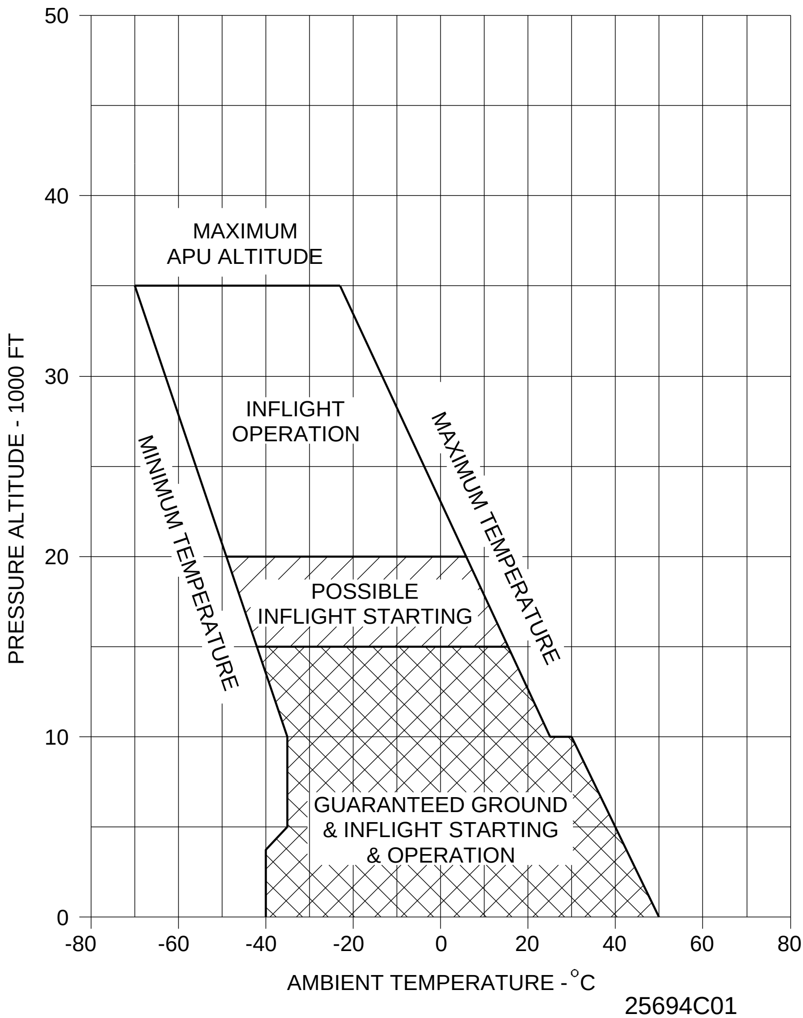
APU Control Panel
Figure 2



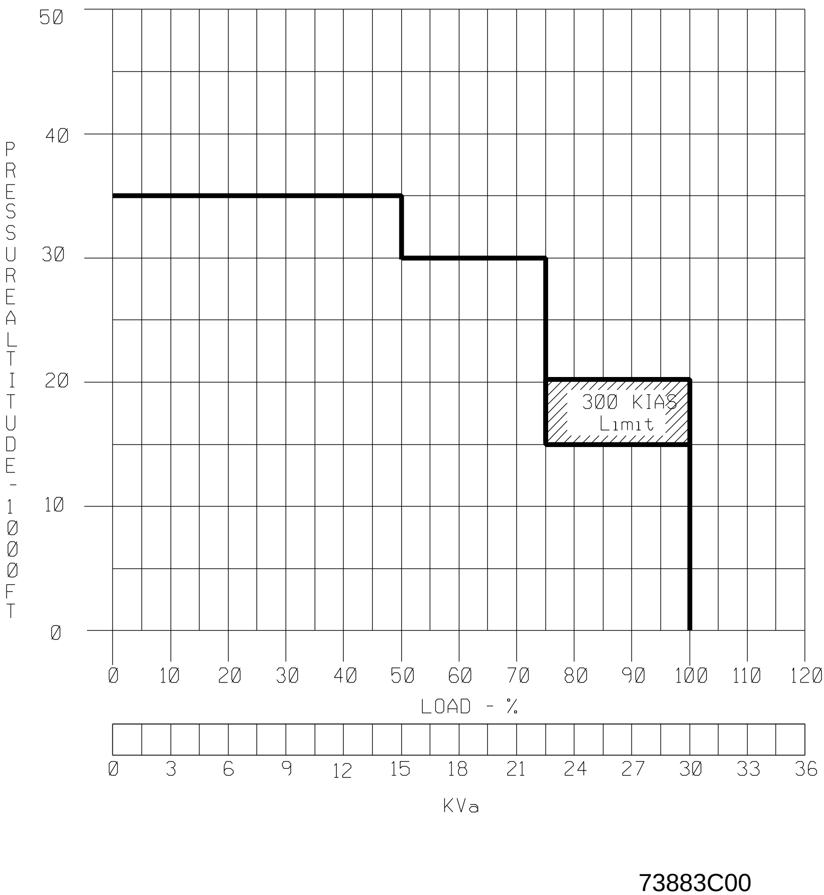
APU Load and ELWS Warning Indications
Figure 3



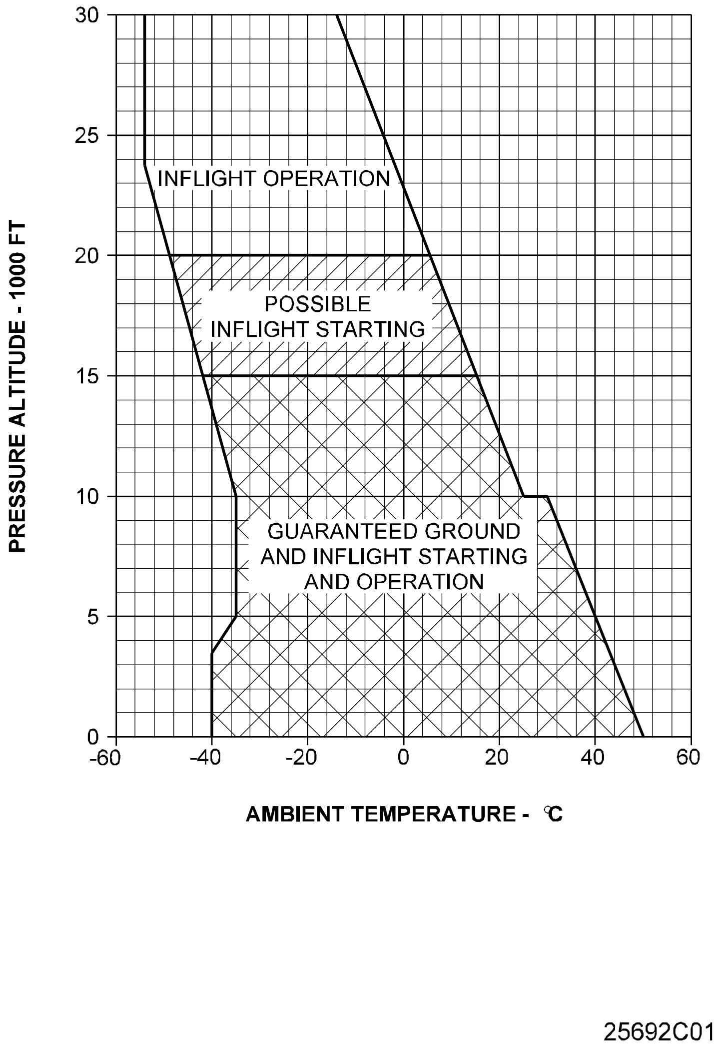
ELWS/ASC 420 APU Alternator Electrical Load Envelope: SN 1000 – 1479 With APU Loadmeter Installed (or Removed By ASC 420); SN 1480 & Subs; Airplanes Having ASC 390A or ASC 427
Figure 4



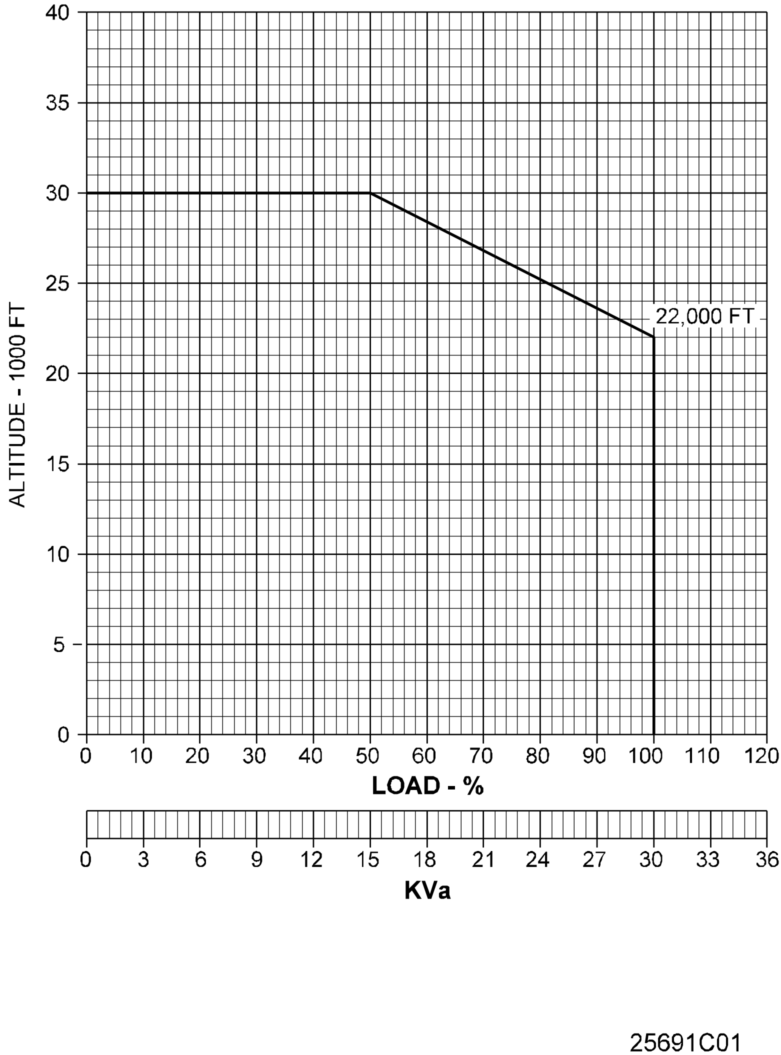
ELWS/ASC 420 APU Operating Envelope: SN 1000 – 1155 With APU Loadmeter Installed (or Removed By ASC 420); SN 1156 and Subs; SN 1430 and Subs; Airplanes Having ASC 465
Figure 5



APU Alternator Electrical Load Envelope: Airplanes Having ASC 465
Figure 6



APU Operating Envelope: SN 1000 – 1155 Without APU Loadmeter
Figure 7



APU Alternator Electrical Load Envelope: SN 1000 – 1155 Without APU Loadmeter
Figure 8

2A-52-10: General

1. Fuselage Doors:

The Gulfstream IV is fitted with doors for access to the pressurized and unpressurized sections of the fuselage. Doors installed in the pressurized section are equipped with seals that maintain airplane pressurization integrity. For the doors in the unpressurized section, no seals are required and use of these doors is restricted to ground operations.

A. Pressurized Section Doors:

The following doors and openings are fitted into the pressurized section of the airplane fuselage:

- Main entrance door
- Baggage compartment door
- Cargo door (optional installation)
- Emergency exit windows

B. Unpressurized Section Doors:

The following doors and openings are fitted into the unpressurized section of the airplane:

- Tail compartment door
- Service access doors

2. Door Controls and Indications:

The main entrance door and the optional cargo door are hydraulically powered through switch selections on the respective control panels. Each control panel has appropriate indicators for operating the door. The remaining doors in the pressurized and unpressurized areas of the fuselage are manually operated.

All doors are monitored by door warning circuits that provide indications to the cockpit if the doors are not closed and locked or latched.

2A-52-20: Fuselage Doors

1. Pressurized Section Doors:

The pressurized section of the fuselage is equipped with a main entrance door, a baggage compartment door, overwing emergency exits, and at customer option, a cargo door fitted into the fuselage forward of the right wing leading edge. See Figure 1 and Figure 2. All doors that are used in normal operations are equipped with inflatable seals around the circumference of the doors. The seals are inflated by pneumatic pressure from the airplane bleed air manifold to seal the opening around the doors to preserve cabin pressurization integrity. The overwing emergency exits are plug type openings with fixed non-inflatable seals designed to allow removal of the exits inward when the airplane is not pressurized (on the ground).

A. Main Entrance Door:

The main entrance door, shown in Figure 3, is located on the forward section of the left fuselage approximately midway between the cockpit and the wing leading edge, and is the primary access to the airplane. The thirty-

six (36) inch by sixty-two (62) inch door is hinged at the bottom and is equipped with a folding stairway and handrails integrated into the interior of the door structure. The outside of the door conforms to the curvature of the fuselage and is manufactured in a similar manner to the fuselage, with ribs and a covering metallic skin. The integrated stairway folds at mid-point, allowing the stairs to extend from the entranceway threshold to just above ground level. The handrails unfold as the stairway extends, locking into position at full stairway extension. When the main entrance door is closed, the handrails unlock and the stairway folds into a secured position against the interior of the door.

The door is closed hydraulically, using pressure supplied by the auxiliary hydraulic system that is reduced for door operation to 1400 psi (± 100) by a reducing valve in the door lines. The door is opened manually, with the weight of the door and attached stairway rotating the door assembly out and down by free fall. The door opening rate is moderated by a restrictor in the hydraulic lines that uses trapped fluid as a brake to slow door and stairway operation.

Door operation may be controlled from either inside or outside of the airplane. The mechanical door latches and locks are interconnected, so that operating either the inside or outside handles also moves the other. The electrical circuitry for closing the door is similarly interconnected, so that either inside or outside controls may close the door.

To close the door from outside, switches within a covered door control panel, located under the door beneath the oxygen servicing panel, are used to power the Essential DC bus and auxiliary hydraulic system to retract the stairway and close the door. A BATT SWITCH within the panel is selected ON to power the Essential DC bus and the auxiliary hydraulic system on airplanes Serial Number (SN) 1000-1155. The OUTSIDE DOOR SWITCH is then positioned to CLOSE to retract the stairway and close the door. On airplanes SN 1156 and subsequent, the OUTSIDE DOOR SWITCH will automatically power the Essential DC bus and auxiliary hydraulic system, and there is no need to turn on the BATT SWITCH to operate the door.

NOTE:

On airplanes SN 1000-1155, the external BATT SWITCH must be turned OFF after the door is closed to prevent draining the airplane batteries.

To close the door from inside the airplane, a DOOR SAFETY SWITCH on the lower outside of the pilot circuit breaker panel is moved from the SAFE position, routing power to the INSIDE DOOR CONT SWITCH, located above the DOOR SAFETY SWITCH for SN 1000-1279, or located on the cockpit door edge behind the pilot seat for SN 1280 and subsequent. The control switch may then be selected to CLOSE, powering the Essential DC bus and auxiliary hydraulic system to close the door.

Airplanes not having ASC 453A: Once the door is closed, it must be latched and locked. From the outside of the airplane, the door is locked by pushing in the outer locking handle, then pushing in the inner latch to fair with the outside of the door. From inside the airplane, the door is latched and locked by pulling up on the red ball-shaped latch and then rotating the

locking handle inboard and over-center. The red latch is then released.

Airplanes having ASC 453A: When door stops in door sill (AUX pump still running), gently pull door inward until door is fully seated into sill, if required. After assuring that the door is seated within the sill, rotate the primary locking handle downward and inward until handle travels overcenter and can go no further. Release the primary locking handle. With door properly closed, primary locking handle cannot be moved.

The door is locked by the movement of the inner or outer locking handles that position six bayonet-type fittings from the door into openings in the door frame. The bayonet fittings lock the door in place and close microswitches within the door frame openings. The microswitches provide electrical inputs to the door warning system, door hydraulic system and to the door seal pressurization system. The microswitches are connected in series so that each switch must be in the correct position in order to generate the proper signal.

Automatic activation of the main entrance door emergency lighting is available on SN 1467 and subsequent through the use of a second set of emergency batteries. If the main entrance door is open and the EMERG POWER ARM switchlight (cockpit overhead panel) has been previously been selected to ARM, activation of the emergency batteries will allow emergency batteries No. 3 and 4 to automatically power the interior emergency lighting and main entrance door emergency lighting.

B. Optional Cargo Door:

At customer option, a cargo door may be installed on the right side of the fuselage forward of the wing leading edge. The door is eighty and one-half (80 ½) inches high and eighty-one (81) inches wide and opens out and upward. The door is restricted to ground operation by the nutcracker (squat) switch system. Like the main entrance door, the cargo door is electrically controlled and hydraulically operated, has a pneumatically inflated seal, and is locked and latched mechanically. A depiction of the door is shown in Figure 4.

A door control panel is located just forward of the door on the inside of the airplane cabin. The control panel has four indicator / switchlights and an operating switch. Hydraulic power is furnished by the auxiliary hydraulic system that must be activated using the switch on the cockpit overhead panel. To open the door, the auxiliary hydraulic system must first be powered. The LOCK HDL REL switchlight is then pressed, and the UNLK legend appears in the switchlight. The UNLK legend indicates that the relay disabling the door locking handle has released, and the handle may be unlocked. The relay will only release the locking handle for ten (10) seconds during which time the handle may be moved to unlock the door. The door lock handle is mechanically connected to a vent door that opens when the lock handle is move to the unlock position. The vent door insures the release of cabin pressure prior to opening the cargo door. The position of the vent door, as viewed from outside the airplane, may be used as an indicator of the position of the door locking handle.

Once the door is unlocked, the door latching handle is moved to the open position. Moving the handle to the open position releases six (6) mechanical locks at the bottom of the door and opens the six microswitches in the door warning circuit. In the open position the handle

opens relays enabling the auxiliary hydraulic system, reduced to 1400 psi (± 100) by the main entrance door reducer valve, to power the cargo door. The availability of hydraulic power is indicated by illumination of the DOOR HYD PWR indicator. The door switch, located on the lower section of the control panel, is selected to the OPEN position and the hydraulic actuator opens the cargo door. When the door is fully open, releasing the door switch de-energizes the hydraulic power relay, and trapped hydraulic pressure holds the door in the open position. Closing the door is the reverse of the opening sequence.

C. Baggage Compartment Door:

The airplane baggage compartment, located in the aft part of the pressurized section of the fuselage, is a accessible in flight from an interior door in the passenger compartment, and on the ground through an exterior door fitted into the left side of the fuselage below the left engine pylon. See Figure 5. The exterior door opens inward and upward to provide access for loading and unloading baggage. Mechanically linked interior and exterior handles rotate to open the door. Like the main entrance door, the baggage compartment door is fitted with four bayonet-type plungers that lock the door and position microswitches to provide the circuitry for door warning and door seal inflation. The baggage door may be used as a supplementary emergency exit from the airplane since the baggage compartment is accessible from the cabin.

D. Emergency Overwing Exit Windows:

Two overwing emergency exits are installed on each side of the airplane in the passenger seating area. The four overwing exits are the primary escape exits since the position of the exits provides adequate ground or water clearance for exiting the airplane in the event of a forced landing or ditching. The overwing exits are operated by pulling outward on the marked exit handles on the inside of the windows, unlatching the windows from the window seals. The windows may then be removed into the interior of the airplane, freeing the emergency access. There is an emergency exterior handle for each overwing window exit that is accessed through a door beneath each window. Placards on the airplane exterior provide instructions for opening the panels and rotating the handle within to open the corresponding emergency window.

2. Door Seal System:

A. Pressurized Doors:

The main entrance door, baggage compartment door and optional cargo door are surrounded by pneumatic seals that inflate to maintain cabin pressurization in flight. The seals are manufactured from natural or synthetic rubber compounds reinforced with knitted elastic fabric and covered with neoprene. The door seals are connected to a dedicated pneumatic supply line plumbed into the bleed air manifold. A pressure regulator in the supply line maintains inflation at seventeen (17) to nineteen (19) psi and incorporates a safety relief valve to prevent over-inflation and blowouts of the seals. Check valves installed in the line maintain seal inflation in the event of loss of bleed air pressure, ensuring the integrity of cabin pressurization.

Door seals are activated when the doors are in the correct locked and latched position, signalled by the position of the bayonet fittings on the

main entrance and baggage doors or locking and latching handles on the optional cargo door. The seals inflate as engine bleed air enters the manifold for cabin pressurization, pushing out against the frame of each door.

On airplanes SN 1000-1284 having ASC 364, and SN 1285 and subsequent, additional pneumatic supply lines, one from each engine, are installed to the door seal pressure regulator. The additional lines ensure a constant pressure and flow output from the door seal regulator.

On airplanes SN 1000-1284 having ASC 424, and SN 1285 and subsequent, a high flow pressure regulator and pressure filter are installed to improve door seal system pressure and minimize system contamination. The new pressure regulator has approximately ten (10) times the mass flow capability at a higher output pressure, improved reliability and contamination resistance, and is specifically tailored to the GIV high altitude/high cabin pressure differential. The new, stainless steel pressure filter is improved to minimize contamination and is relocated upstream of the pressure regulator.

ASC 438 is available as an additional ASC for airplanes already having ASC 364 and 424. ASC 438 removes an existing air conditioning duct located between the air conditioning pack and water separator and installs a new moisture removal condenser downstream of the door seal system regulator.

ASC 439B installs a door seal advanced warning system on airplanes previously having ASC 424 (can also be installed in conjunction with ASC 439B). High and low pressure switches are installed downstream of the door seal regulator to provide indications of abnormal door seal supply pressures. Similar pressure switches are installed upstream of the main entrance and baggage doors to provide indications of low door seal pressure. Cockpit indications are displayed as follows:

- **Airplanes having SPZ-8000:** A “split capsule” indicator located on the cockpit overhead panel displays the following messages:
  - **DRSL H:** Output pressure from door seal regulator is greater than 27 psig ±1.5.
  - **DRSL L:** Output pressure from door seal regulator is less than 12 ±1.5 psig and/or the main entrance or baggage door seal pressure is less than 2.5 ±0.5 psig relative to cabin pressure.
- **Airplanes having SPZ-8400:** CAS messages and their meanings are as follows:
  - **DOOR SEAL PRES LOW:** Main entrance door and/or baggage door seal pressure is less than 2.5 ±0.5 psig relative to cabin pressure.
  - **DOOR SUPPLY HIGH:** Output pressure from door seal regulator is greater than 27.5 ±1.5 psig.
  - **DOOR SUPPLY LOW:** Output pressure from door seal regulator is less than 12 ±1.5 psig.

B. Smoke Evacuation Valve:

On airplanes SN 1000-1155 having ASC 157, SN 1034 and 1156 and subsequent, a smoke evacuation valve is installed in the pneumatic line to the baggage door inflatable seal. See the illustration in Figure 6. The evacuation valve is located within a panel on top of the frame of the door opening from the passenger cabin into the baggage compartment. The valve has two positions: NORMAL OPS that selects normal operation of the baggage door seal, and EVAC SMOKE that closes the pneumatic line to the baggage door seal, deflating the seal and allowing any smoke accumulating in the aft section of the pressurized cabin to be drawn out of the airplane around the deflated baggage compartment door seal.

3. Unpressurized Section Doors:

A. Tail Compartment Door:

Access to the unpressurized compartment in the aft fuselage is provided by a door and a telescoping ladder attached to the inside of the door. See Figure 7. The tail compartment door is opened and closed from outside the airplane with a handle installed flush with the airplane surface, accessed by a push-to-release button on the handle. When the handle is released, it extends and can be rotated to retract the bayonet fittings that lock the door. The door opens downward and the attached ladder is manually extended to provide access to the compartment. A microswitch on the bayonet fitting recess is connected to the door warning system. The microswitch will also extinguish any tail compartment lights left on when the door is latched.

If for some reason the telescoping ladder has moved from the stowed position while the door is closed and interferes with opening the door, a small access panel on the aft part of the door may be removed and the ladder repositioned to operate the door.

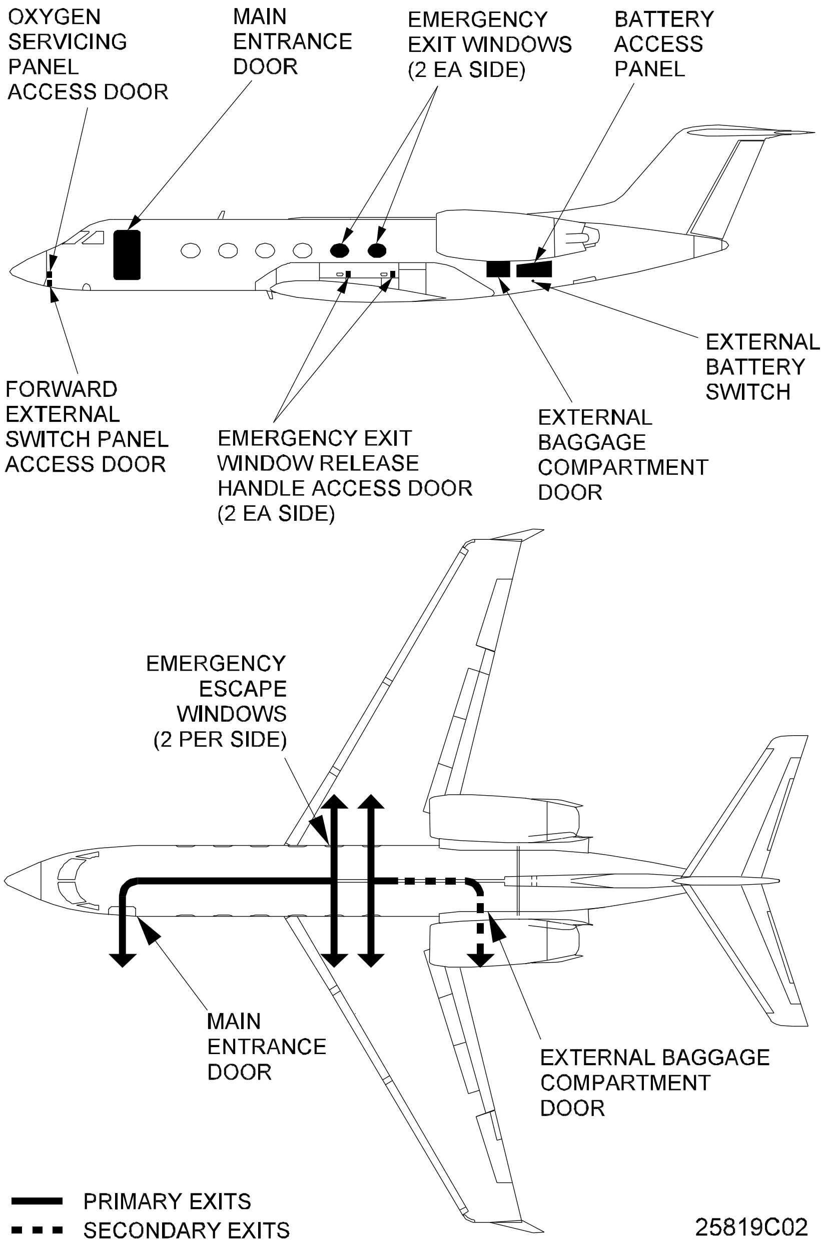
B. Service Compartment Doors:

The doors of the airplane service panels have a striker plate that closes a microswitch within the panel. When the door to the panel is opened, the microswitch is opened, powering a light within the panel and sending an open signal to the door warning system.

The following service panels are located on the exterior of the airplane. Not all airplanes have the provision for panel internal lighting or for door position monitoring with the door warning system.

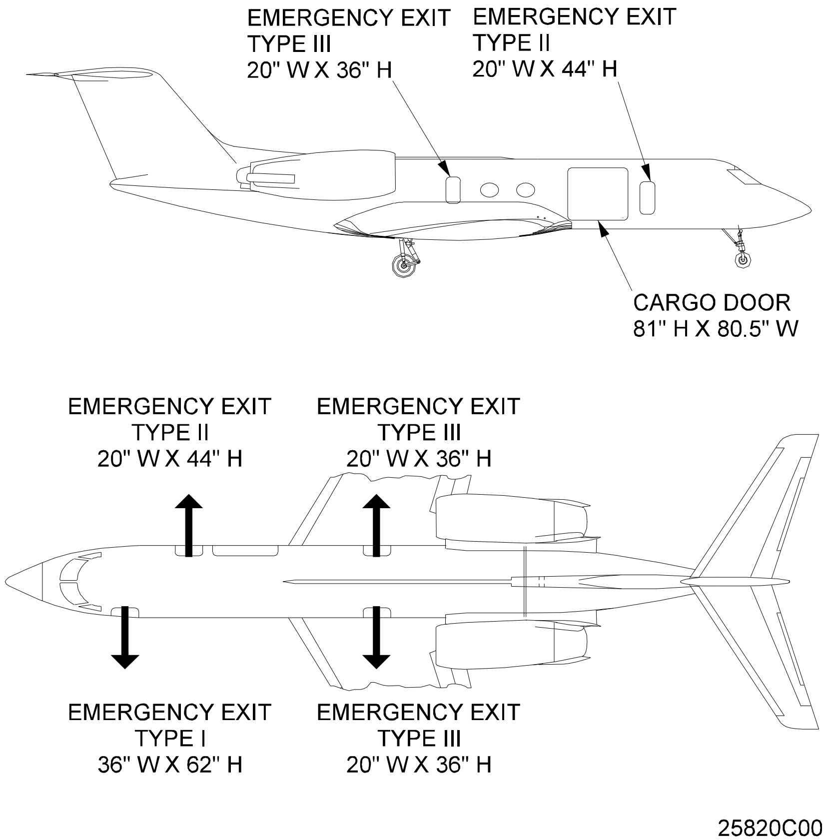
- Lavatory service
- Water service
- Oxygen service
- Single-point refueling
- Forward external switch panel

A forward lavatory service panel is installed as standard equipment on SN 1439 and subsequent. SN 1439 through 1454 have provisions to enable door position monitoring, if desired by the operator. On SN 1455 and subsequent, door position monitoring is enabled as standard.

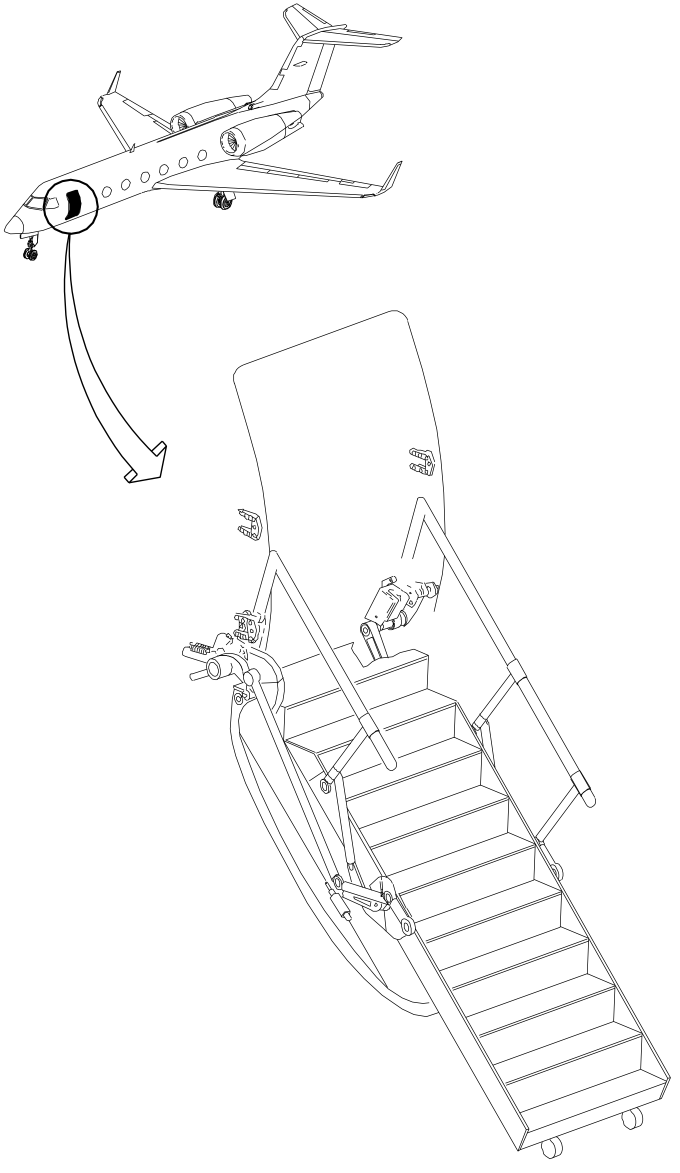


Entrances, Exits and External Doors (Airplanes Not Having Cargo Door)
Figure 1

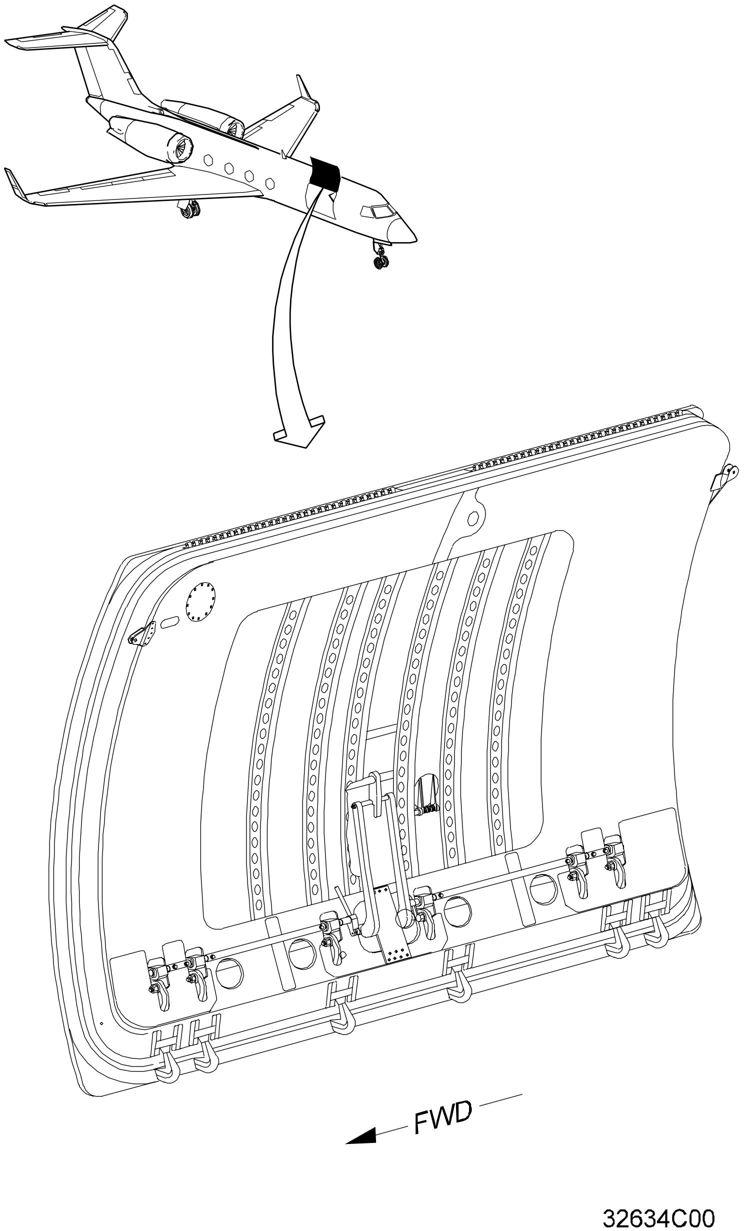
NOTE: DEPICTS AIRCRAFT WITH ASC 213



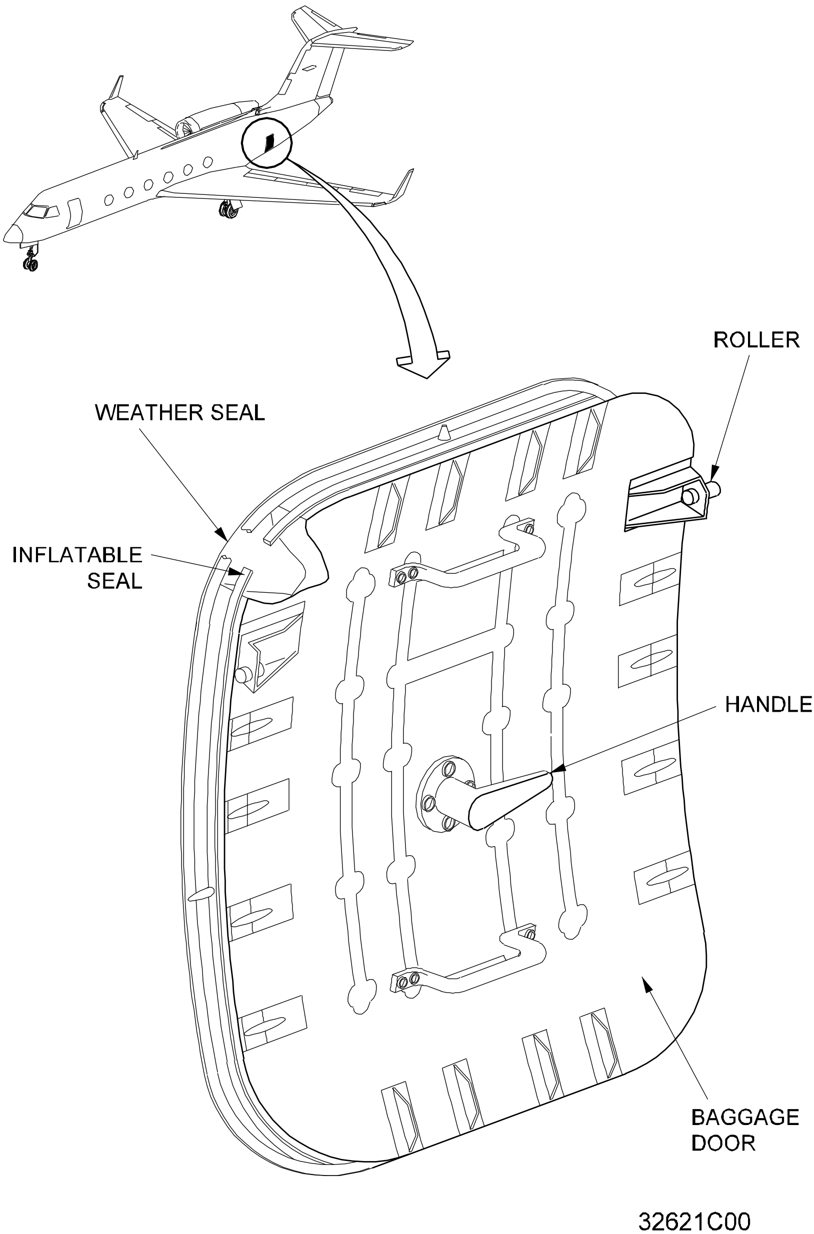
Entrances, Exits and External Doors (Airplanes Having Cargo Door)
Figure 2



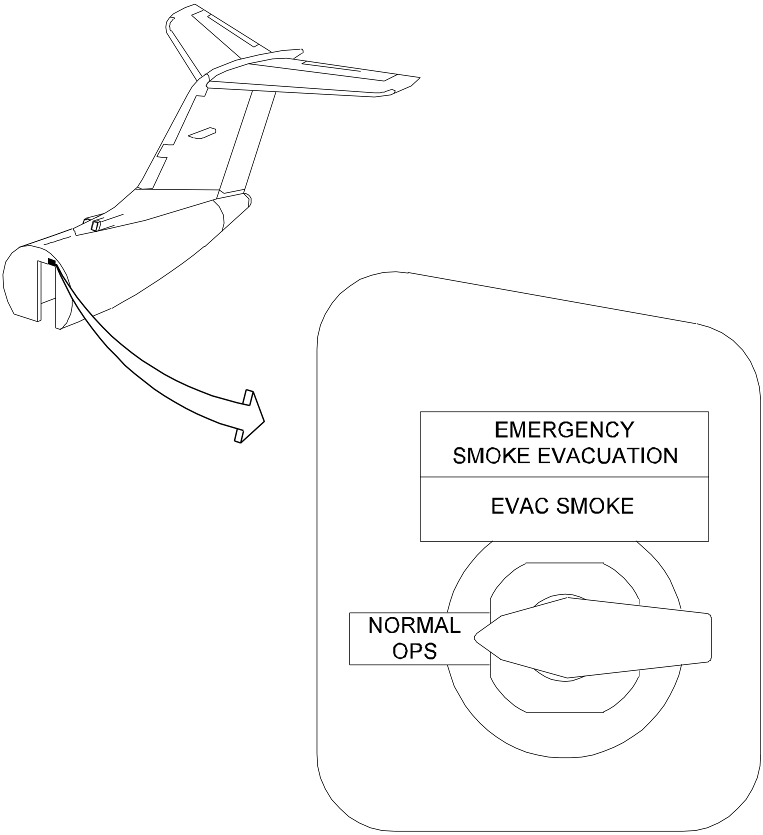
Main Entrance Door Installation
Figure 3



Cargo Door Installation
Figure 4



Baggage Compartment Door Installation
Figure 5



EMERGENCY SMOKE EVACUATION

(SN 1000-1155 (excluding SN 1034) having ASC 157; SN 1034; SN 1156 and subs)

- EVAC SMOKE: Allows deflation of baggage compartment door seal.
- NORMAL OPS: Allows baggage compartment door seal to reinflate.

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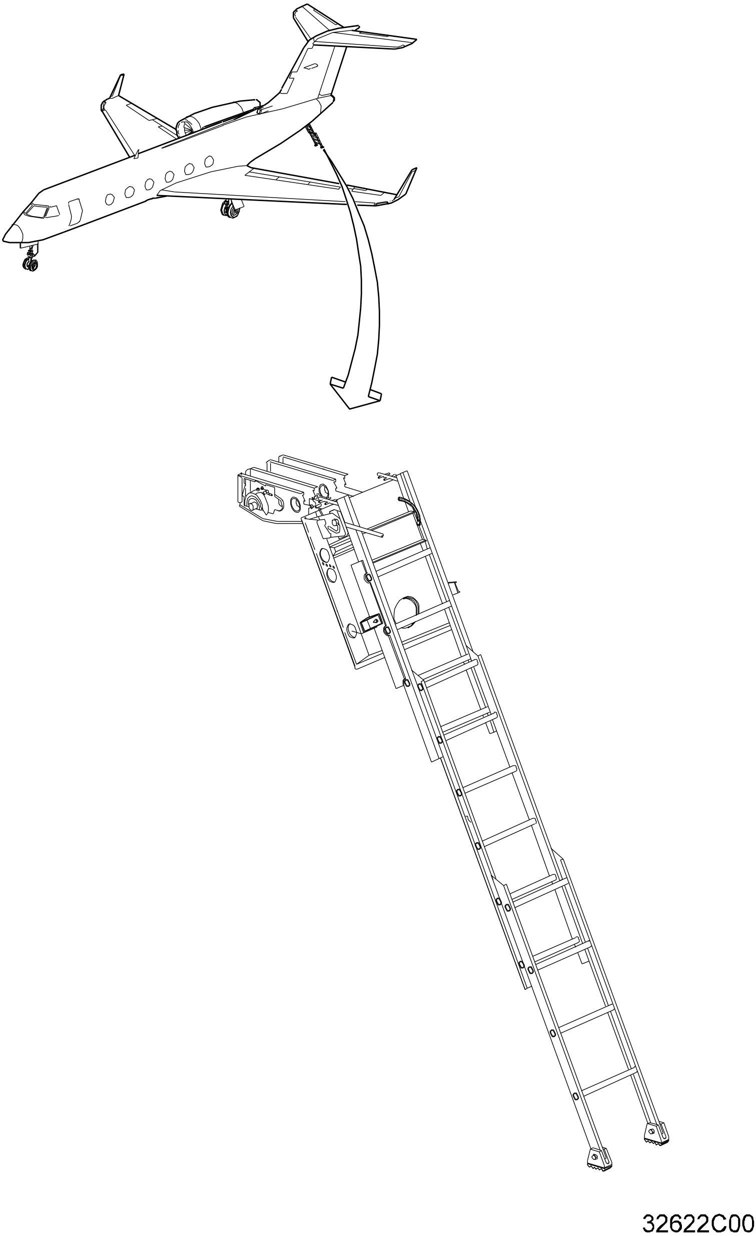
Smoke Evacuation Valve
Figure 6



Tail Compartment Door Installation
Figure 7

2A-52-30: Door Controls and Indications

1. Pressurized Section Door Controls:

A. Main Entrance Door:

The main entrance door is locked by bayonet fittings positioned by a locking handle on the inside of the door or by an outside locking handle. Both the inside and outside handles have secondary latch release handles that secure the locking handles in the locked position.

The main entrance door is normally opened by manually unlocking the bayonet fittings and pushing (from inside) or pulling (from outside) the door over center and letting the door and attached airstair open in a controlled free fall using the weight of the door assembly. See Figure 9 and Figure 10. The opening action of the door is damped by the controlled release of trapped hydraulic fluid. Neither electrical nor hydraulic power are required to open the main entrance door. Normal procedures for opening the main entrance door can be found in Section 09-01-40, Opening and Closing Airplane Doors.

The main entrance door is normally closed using electrical switches located inside an exterior panel or by switches in the cockpit (and on the cockpit door frame for some airplanes) shown in Figure 8 and Figure 10. These switches enable the door close circuitry and control operation of the Auxiliary Hydraulic (AUX) pump during closing. Normal procedures for closing the main entrance door can be found in Section 09-01-40, Opening and Closing Airplane Doors.

If necessary, the main entrance door may be closed or opened manually using alternate procedures. Whenever alternate procedures are used to open or close the main entrance door, the flight crew should ensure the door is never allowed to free fall. If there is no controlled release of hydraulic fluid, damping will not be available and the door could fall with its full weight, resulting in serious injury to personnel and heavy damage to the door and airplane. Alternate procedures for opening the main entrance door, found in Section 09-03-20, Manually Opening and Closing Main Entrance Door, must be carefully followed to ensure maximum safety.

B. Baggage Compartment Door:

The baggage compartment door is opened manually with either the interior or exterior door handle, positioning the integrated bayonet fittings that lock the door. Procedures for opening and closing the baggage compartment door can be found in Section 09-01-40, Opening and Closing Airplane Doors.

C. Optional Cargo Door:

The cargo door, if installed, is controlled with switches on a door control panel, shown in Figure 11. The panel has four switchlights and a three-position control switch.

2. Unpressurized Section Door Controls:

A. Tail Compartment Door:

The tail compartment door is manually opened from the outside with a faired handle, and the attached telescoping stair is extended by hand. Procedures for opening and closing the tail compartment door can be found in Section 09-01-40, Opening and Closing Airplane Doors.

B. **Service Compartment Doors:**

Service compartment doors are opened by pressing latch release buttons integrated into the spring-held latches.

3. **Door Warning System:**

The door warning system monitors the position of the main entrance door, baggage compartment door, cargo door (if installed) and the tail compartment and service doors. A 28V DC circuit will illuminate EICAS, SWLP, or instrument panel light capsule warnings if microswitches are not closed by the locking action of doors. The microswitches are located on:

- The forward upper, top and lower aft bayonet fitting receptacles in the frame of the main entrance door
- The main entrance door latching handle
- The aft bayonet fitting receptacle of the baggage door
- If installed, on the six hook latches of the cargo door and on the cargo door locking handle
- On plunger fittings on the frame of the aft compartment door and service compartment doors

4. **CAS Messages:**

A. **Warning (Red) CAS Messages:**

CAS Message	Cause or Meaning
MAIN DOOR	Main entrance door open or unlocked
BAGGAGE DOOR **(1)**	Baggage door open or unlocked

NOTE(S):

(1) Amber caution message on SN 1000-1389 having ASC 415B, SN 1390 & subs.

B. **Caution (Amber) CAS Messages:**

CAS Message	Cause or Meaning
BAGGAGE DOOR **(1)**	Baggage door open or unlocked

NOTE(S):

(1) SN 1000-1389 having ASC 415B, SN 1390 & subs.

C. **Blue (Advisory) CAS Messages:**

CAS Message	Cause or Meaning
SERVICE DOORS	One or more of the following doors is open: • Tail compartment door Service compartment doors (applicability varies): • Refueling panel • Oxygen service panel • Water service panel • Aft lavatory service panel • Forward external switch panel • Forward lavatory service panel (SN 1455 & subs)

D. Standby Warning Lights Panel (SWLP) Annunciations (SPZ-8000 Equipped Airplanes):

Annunciation	Cause or Meaning
CABIN DOORS	Main entrance and/or baggage door open or unlocked
MAIN DOOR **(1)**	Main entrance door open or unlocked

NOTE(S):

(1) SPZ-8000 equipped airplanes having ASC 415B.

E. Instrument Panel Annunciations:

For airplanes equipped with the optional cargo door, an indicator light capsule is installed on the pilot and copilot instrument panels that illuminates with the text CARGO DOOR whenever the door is open or unlocked. See Figure 12.

5. Circuit Breakers (CBs):

Circuit Breaker Name	CB Panel	Location	Power Source
DOOR CONT / WARN	CPO	B-8	ESS 28V DC
CARGO DOOR SOL	CPO	A-14	ESS 28V DC

6. Limitations:

A. Flight Manual Limitations:

There are no limitations for the door controls and indications system at the time of this revision.

B. System Notes:

(1) Main Entrance Door:

(a) If the main entrance door was manually closed, there will be no restricted damping to oppose the weight of the door during opening. Serious injury to personnel or damage to door and airplane could result if door is allowed to free-fall to the open position.

(b) Never stand underneath the main entrance door when opening the door.

(c) Do not stand or walk on the main entrance door until it is fully extended, locked and in contact with the ramp surface.

(d) To prevent damage to the main entrance door sill area, ensure the bayonet fittings are fully retracted before attempting to close the door.

(2) Optional Cargo Door:

(a) If the optional cargo door is installed, it should not be operated when winds are in excess of forty (40) knots, or if the door is open and mechanically braced, when winds are in excess of fifty (50) knots.

(3) Baggage Compartment Door:

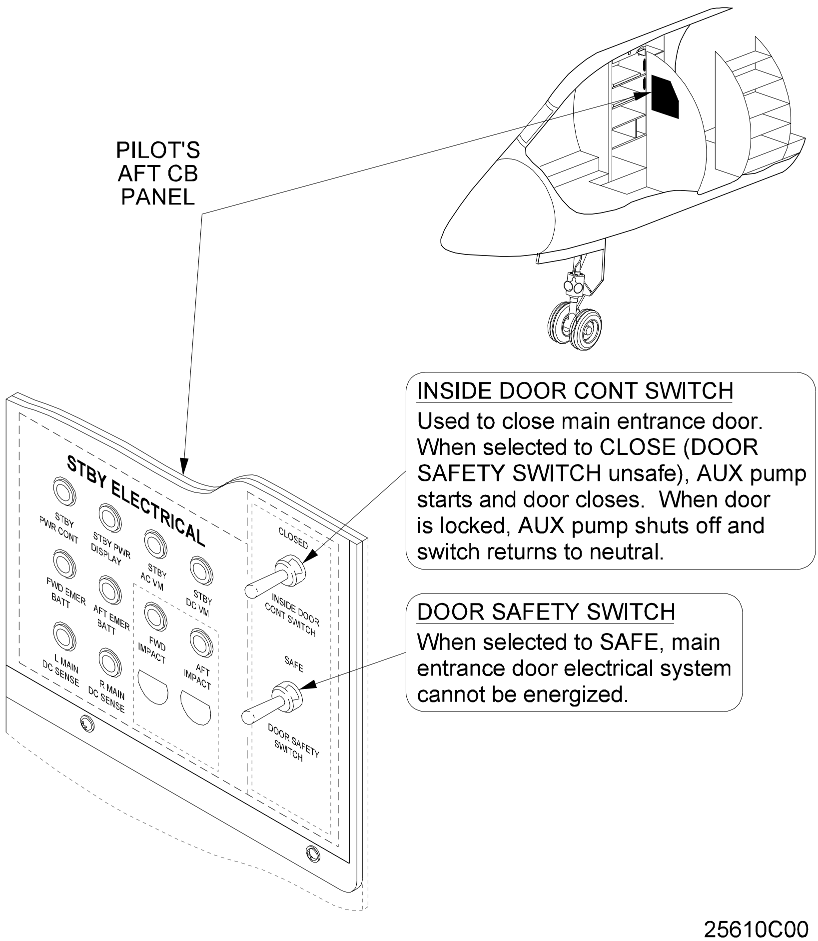
(a) To prevent damage to the baggage compartment door sill area, ensure door handle is rotated to unlock before attempting to close door.

(4) Tail Compartment Door:

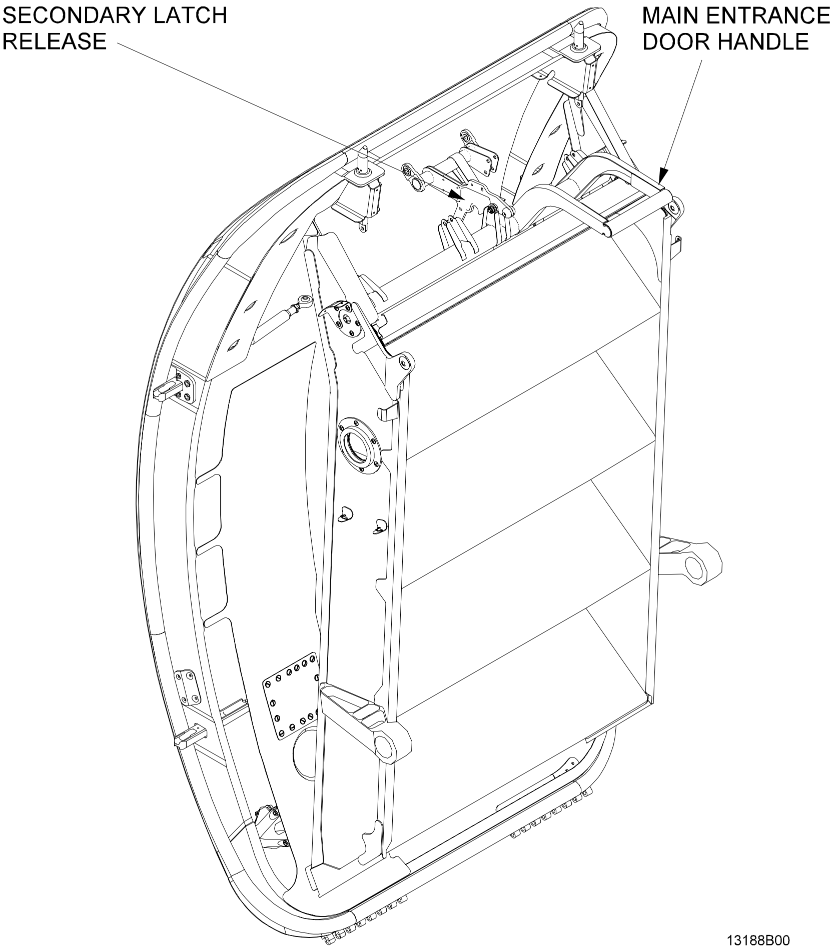
(a) During flight, objects may become dislodged and come to rest on the tail compartment door. When opening the tail compartment door, be alert for objects that may be resting on the door. Always ensure the tail compartment ladder has not become disconnected from door.

If the tail compartment ladder has become disconnected from door, the door may be prevented from lowering once unlocked. If this happens, removing an access panel on the door exterior will allow the flight crew to reach inside the door and position the ladder so that the door may be lowered.

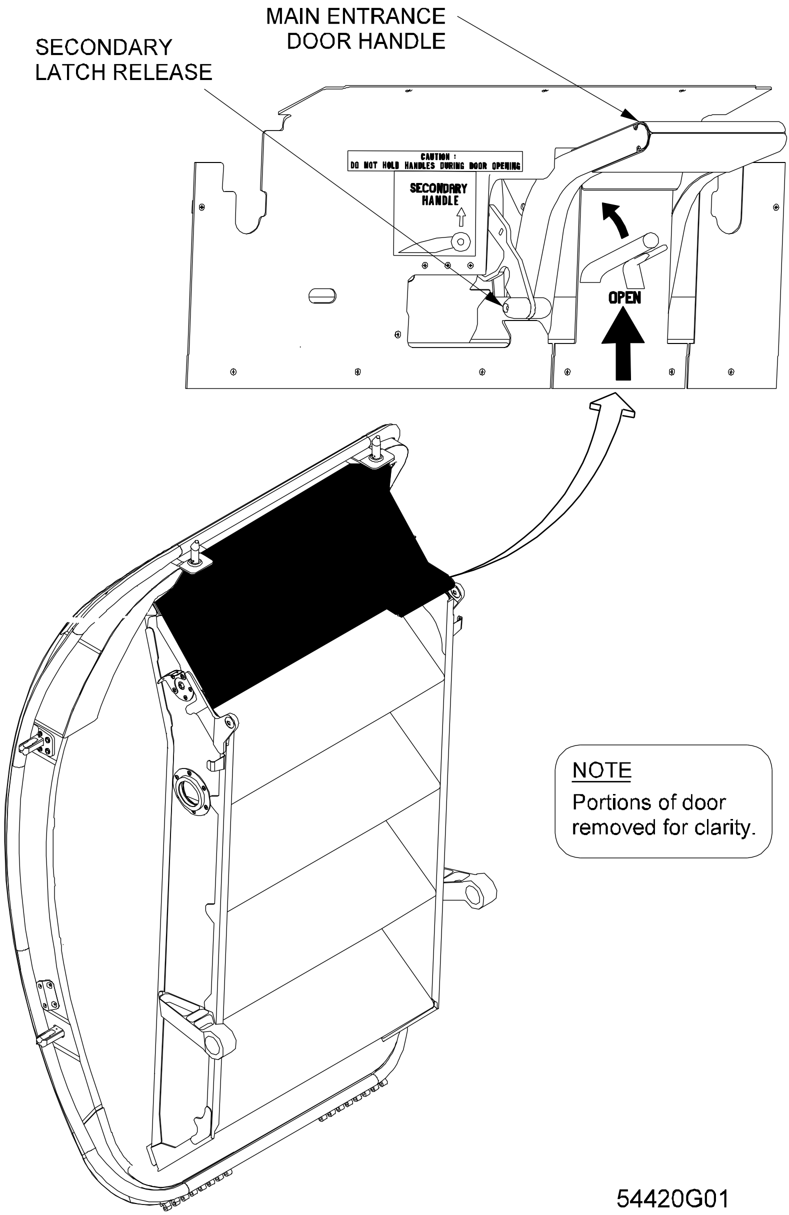
(b) Prior to closing the tail compartment door, ensure the tail compartment ladder is properly secured to the door.



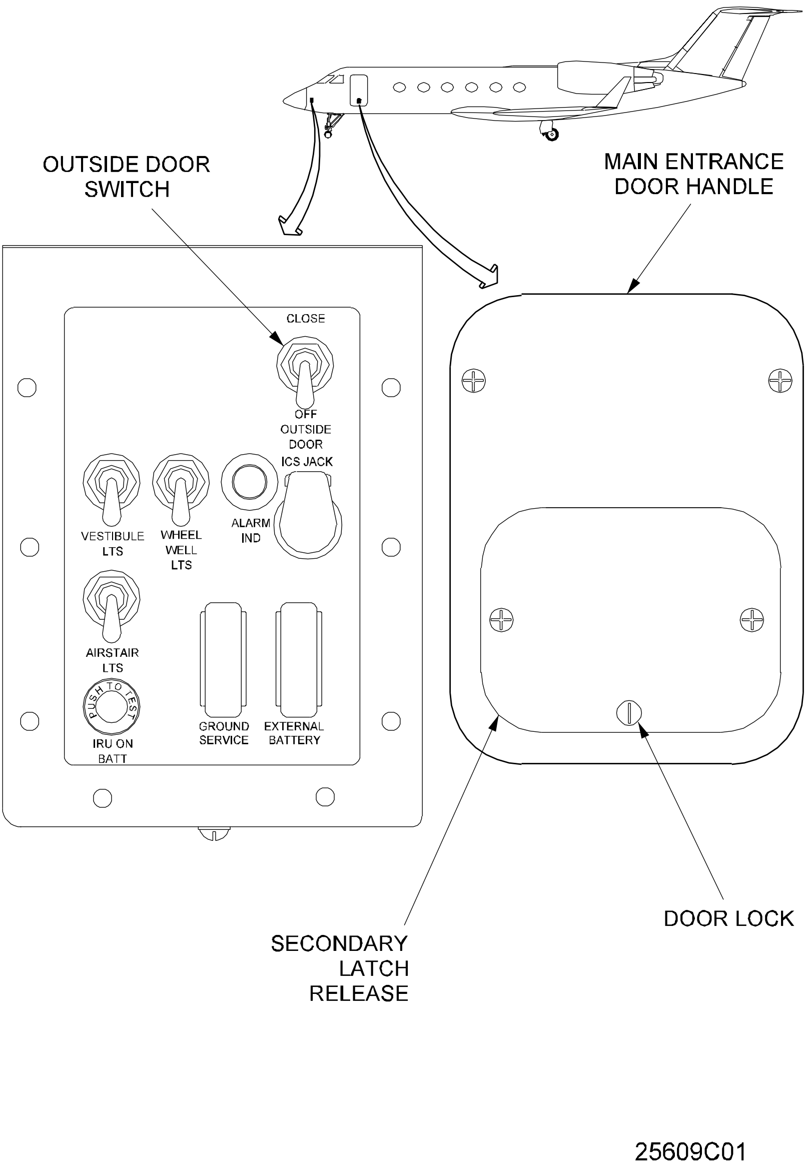
Interior Main Entrance Door Control Switches
Figure 8



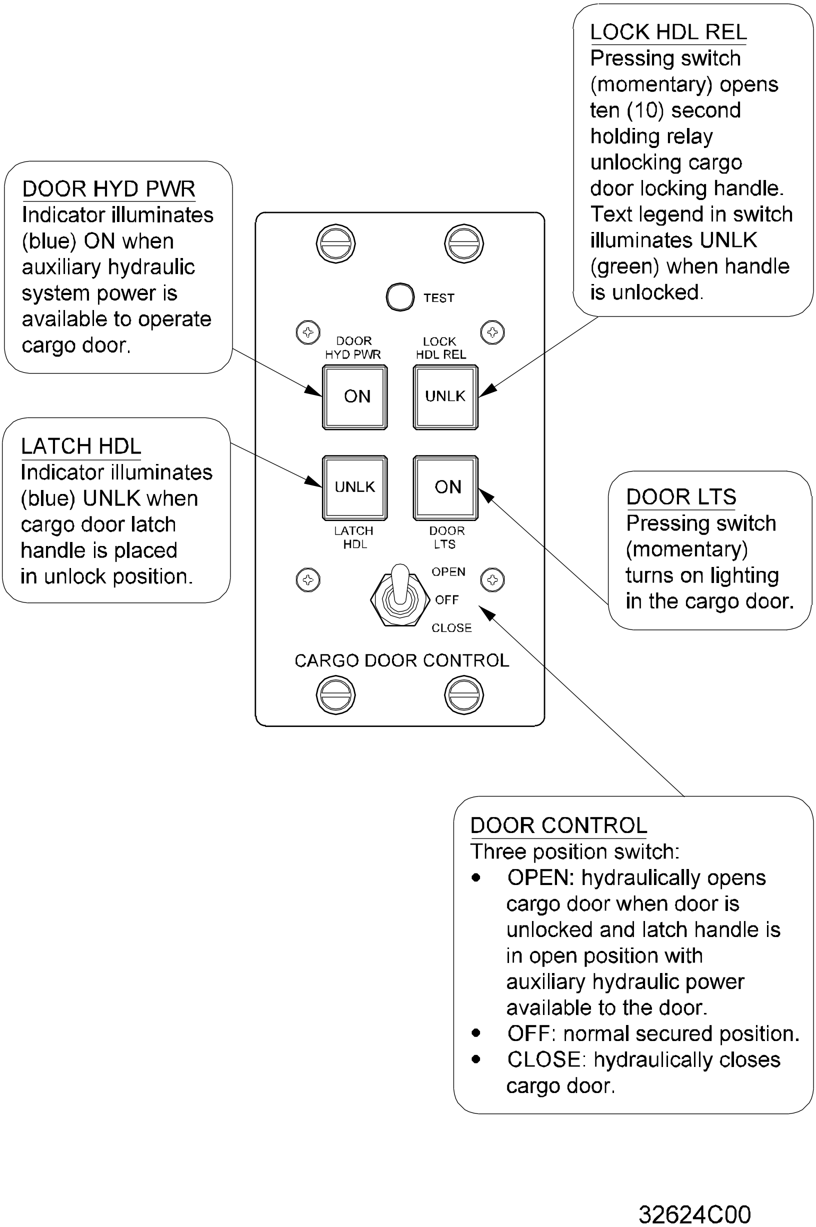
Interior Main Entrance Door Handles (Airplanes Not Having ASC 453A)
Figure 9 (Sheet 1 of 2)



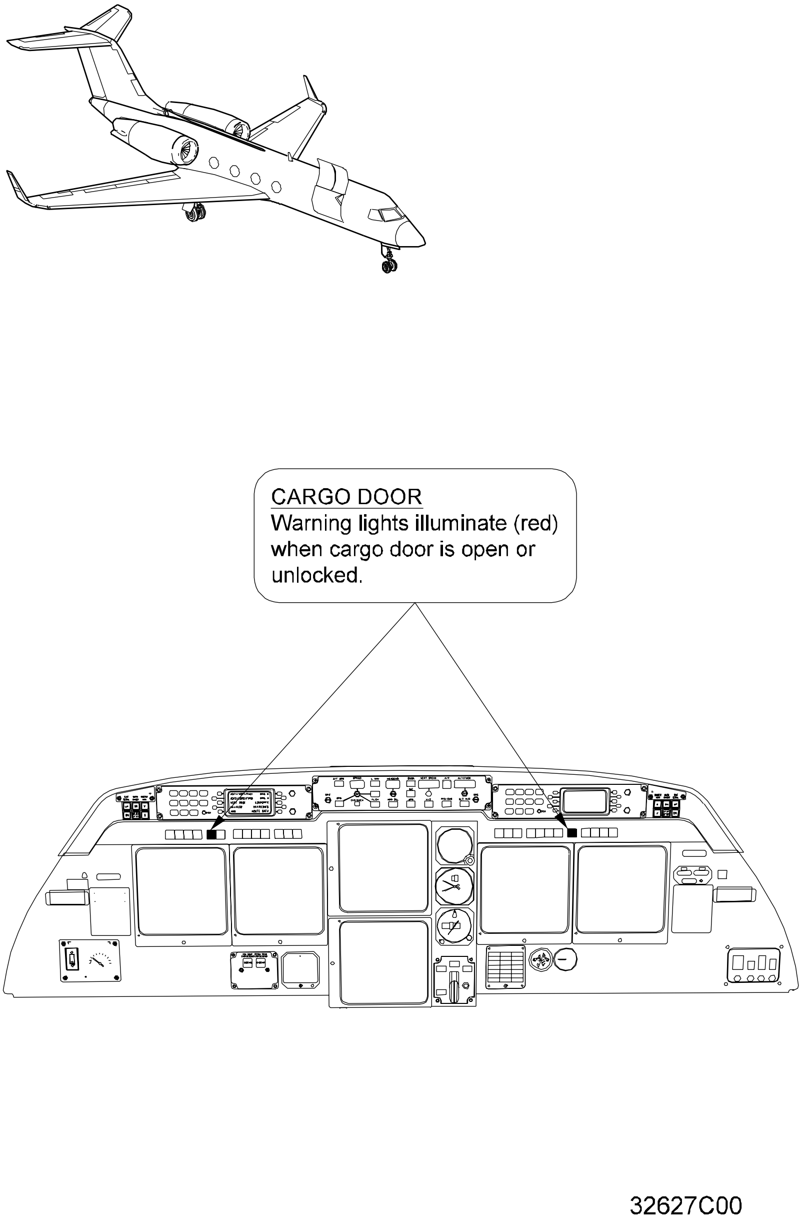
Interior Main Entrance Door Handles (Airplanes Having ASC 453A)
Figure 9 (Sheet 2 of 2)



Exterior Main Entrance Door Handle and Control Switches
Figure 10



Cargo Door Control Panel
Figure 11



Cargo Door Warning Lights on Instrument Panels
Figure 12

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2A-71-10: General Arrangement

1. General Description:

(See Figure 1.)

The Gulfstream IV is powered by two Rolls-Royce Tay Mk611-8 engines, each of which is a twin-spool, axial-flow turbofan. The Tay is a high bypass ratio turbofan, designed to achieve optimum fuel efficiency and component reliability, while complying with noise requirements per FAR Part 36 Stage 3, which reflects the same requirements of ICAO Annex 16 Chapter 3.

The Low Pressure (LP) spool consists of a single stage fan and a three stage Intermediate Pressure (IP) compressor driven by a three stage LP turbine. The High Pressure (HP) spool consists of a twelve stage HP compressor driven by a two stage HP turbine.

The combustion section has ten liners that connect together in an annular chamber. In each liner is a fuel spray nozzle that injects a fine spray of fuel into the liner for ignition.

The fan bypass air stream and LP turbine exhaust are mixed in a twelve lobed forced mixer before discharge to atmosphere through a common propelling nozzle.

The engine is started by an air starter motor which rotates the HP compressor shaft.

A. Engine Data:

(1) Ratings (Takeoff At ISA Sea Level, Static):

- Takeoff Thrust — 13,850 lbs.
- Pressure Ratio — 16.0:1
- Bypass Ratio — 3.10:1

(2) Compressors:

- LP compressor — 1 stage (fan)
- IP — 3 stages
- HP — 12 stages with variable inlet guide vanes and a bleed valve

(3) Dimensions:

- Length — 94.7 inches
- Fan Tip Diameter — 44.0 inches
- Basic Weight (Dry) — 3255 lb

(4) 100% Shaft Speeds:

- LP — 8393 RPM
- HP — 12,484 RPM

B. Subsections Within This Section:

This section is divided into the following subsections:

- 2A-71-20: Nacelle Arrangement
- 2A-71-30: Mechanical Accessories

2. Limitations:

A. Primary Parameter Operating Limits:

The following limitations exist for the Tay Mk611-8 engines installed on the Gulfstream IV:

Condition	LP % RPM	HP % RPM	TGT C°	Time Limit
Maximum Ground Starting TGT	—	—	700	Momentary
Maximum Relighting (Airstart) TGT	—	—	780	Momentary
Maximum Takeoff	95.5	99.7	716 - 800	5 Minutes **(1)**
Maximum Go-Around	95.5	99.7	716 - 800	5 Minutes **(1)**
Maximum Continuous **(2)**	95.5	97.5	715	Unrestricted
Minimum Idle Approach	—	67.0	—	Unrestricted
Minimum Ground Idle	—	46.6	—	Unrestricted
Maximum Reverse	—	88.0	695	**(3)**
Maximum Overspeed	98.3	102.6	—	20 Seconds
Maximum Overtemperature	—	—	801 - 820	20 Seconds

NOTE(S):

(1) The use of takeoff thrust on Go-Around rating is limited to five (5) minutes all engines operating or ten (10) minutes in the event of an engine failure.

(2) Maximum continuous power is not recommended for normal flight operations. Continued use of this power setting may result in reduced engine life.

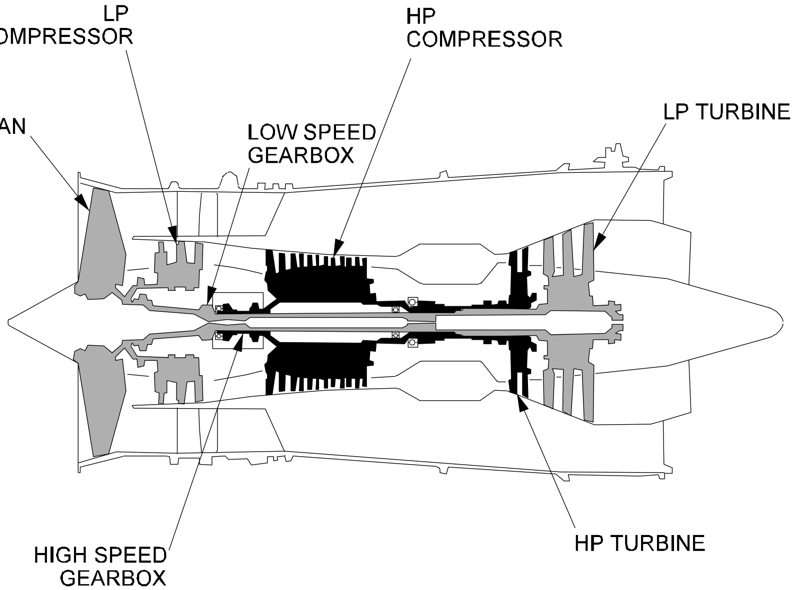
(3) For airplanes SN 1000 thru 1143 without ASC 166: use of thrust reversers is limited to one (1) minute every thirty (30) minutes.

B. Takeoff Power:

For minimum acceptable power settings for takeoff, refer to RATED EPR SETTINGS FOR TAKEOFF THRUST charts in Section 5 of the Airplane Flight Manual. Takeoff EPR must not exceed rated value by more than 0.01.

C. Tailwind Takeoff:

Maximum tailwind component approved for takeoff and landing is 10 knots.



Powerplant General Arrangement
Figure 1

2A-71-20: Nacelle Arrangement

1. General Description:

(See Figure 2.)

The nacelle is designed to:

- House the engine in an aerodynamic casing
- Ventilate the interior of the nacelle
- Provide access to engine components for inspection or servicing
- Collect fluids and deliver them overboard
- Collect core and fan air exhaust gases and propel them rearward
- Assist aircraft speed reduction on the ground
- Provide fire detection and protection within the nacelle

Each nacelle incorporates the following subsystems, units and components:

- Nose Cowl Structure
- Upper and Lower Hinged Cowl Doors
- Fixed Cowl Structure
- Aft Cowl Structure
- Engine Fire Protection System

The nacelle assemblies are dedicated to the left-hand (LH) or right-hand (RH) engine installations and are not interchangeable. The principal material of construction in the fixed and hinged cowling is honeycomb light alloy, except for a steel section of panel assembly outer skin adjacent to the pylon. The nose cowl structure is basically aluminum alloy, with an aft steel bulkhead forming engine forward of the fireshield. The thrust reverser doors and aft cap (aft portion of fairing) are steel construction with the rest of the thrust reverser fairing being of aluminum alloy.

2. Description of Subsystems, Units and Components:

A. Nose Cowl Structure:

(See Figure 3.)

The nose cowl assembly is mounted to the engine air intake casing and forms an annular duct where air is admitted to engine compressors. The structure consists of an inner and outer skin supported by frames. The nose cowl has a piccolo tube mounted inside where hot air is admitted through a series of holes directly from a tap off the engine bleed air manifold for anti-icing purposes. This air is exhausted through holes on the inside of the cowl leading edge to engine air intake. An air inlet scoop is located on the underside of the inlet cowl for alternator cooling purposes. See Section 2A-30-30, Cowl Anti-Ice System, for nose cowl anti-icing.

B. **Upper and Lower Hinged Cowl Doors:**

(See Figure 4.)

WARNING

WHEN WIND SPEED IS ABOVE 15 MILES PER HOUR (13 KNOTS), THE COWL DOORS SHALL NOT BE OPENED OR CLOSED. THIS IS DUE TO THE POSSIBILITY OF THE DOORS BEING BLOWN FREE OF THE CREWMEMBER'S GRASP, WHICH COULD RESULT IN INJURY TO PERSONNEL AND/OR DAMAGE TO THE AIRPLANE . IF THE COWL DOORS MUST BE OPENED IN WINDY CONDITIONS, EVERY ATTEMPT TO MINIMIZE ANY RISK SHALL BE TAKEN, SUCH AS USING ADDITIONAL PERSONNEL AND/OR MOVING THE AIRPLANE SO THAT THE WIND IS BLOCKED. WHEN WIND SPEED IS ABOVE 10 MILES PER HOUR (9 KNOTS), EXERCISE CAUTION WHEN OPENING OR CLOSING THE COWL DOORS.

CAUTION

IF OPENING THE LOWER COWL DOOR OUTSIDE OF THE HANGAR AND UPPER COWL DOOR IS NOT TO BE OPENED, USE A SUITABLE TYING DEVICE WITH AT LEAST 650 LB TENSILE STRENGTH (NO BUNGEE OR OTHER STRETCHABLE CORD) TO SECURE UPPER DOOR IN THE CLOSED POSITION.

CAUTION

DO NOT OPERATE ENGINE ABOVE IDLE WITH COWL DOORS OPEN.

(1) General:

The hinged cowlings consist of upper and lower main access doors, hinged at the upper and lower edges of the fixed cowl respectively. When closed, the doors are secured together with tension latches at the outboard horizontal centerline of the nacelle. Adjustable jury strut rods stowed inside the doors are used to secure the doors in the open position for engine accessibility.

(a) Opening of upper and lower cowl doors:

Due to the mechanical arrangement of the upper and lower cowl doors, the lower cowl door is to be opened first. There are five tension latches securing the lower door to the upper door and are numbered 1 (forward) thru 5 (aft). In addition to these latches, there are also two safety latches; which are both located at 6 o'clock positions at the forward and aft ends of the lower door.

Any time cowl doors are open, ensure doors are secured with front and rear hold open struts. This will allow for utilization of the lower door as a maintenance platform as well as prevent damage to the door and/or fuselage due to high winds or jet blast.

- Ensure area immediately surrounding the upper and lower cowl doors is clear to allow opening of upper and lower cowl doors.
- Position a maintenance platform (adjustable preferred) adjacent to cowl doors. Allow enough vertical and horizontal spacing to prevent the door from contacting the platform.
- Using a flat blade screwdriver or another suitable device, unlatch forward and aft safety latches.
- Depress button on each tension latch and pull firmly to unlock the latches. Do so in the following sequence: No. 1, 2, 4 and 5.
- Lift each of these four latches and secure the safety hooks in theirlocking clips.
- Supporting the weight of lower cowl door, unlatch the remaining No. 3 tension latch and secure its safety hook in holding clip.
- Carefully lower cowl door to the open position.
- Insert the forward and aft jury struts and lock them to the full open position.
- Raise maintenance platform to the approximate vicinity to bottom of the upper cowl door.
- Raise the upper door, insert the forward and aft jury struts and lock them to the full open position.

(b) Closing of upper and lower cowl doors:

NOTE:

If upper cowl door was not opened and is secured to the closed position, remove the securing device at this time.

- Inspect the area inside and outside of the cowl doors to ensure area is clear of tools, maintenance equipment and other objects.
- Manually support upper cowl door. Release and stow front and rear jury struts.
- Lower upper cowl door to the closed position.
- Lower adjustable platform to a suitable height to allow access and clearance of lower cowl door.
- Manually support lower cowl door. Release and stow front and rear jury struts.
- Raise and support lower cowl door in the closed

position and push button on each tension latch to disengage safety hooks in the locking clips.

- Engage and lock lower cowl door to upper cowl door with the No. 3 tension lock by applying firm overcenter pressure.
- Engage and lock the remaining four tension latches in the following order: No. 4, 5, 2 and 1 in a similar manner.
- Verify each of the five latches (and button within that latch) is flush and locked.
- Latch forward and aft safety latches at under side of lower cowl.
- Remove maintenance platform from adjacent area.

(2) Access Doors:

Three access doors are located in lower main door:

- One for engine oil sight gage and oil refill
- One for starter manual shutoff valve
- One for fire access (push to open)

Exhaust ports on lower door include an alternate cooling outlet, a nacelle vent and an LP exhaust air outlet.

C. Fixed Cowl Structure:

(See Figure 5.)

The fixed cowl is attached to the nose cowl and thrust reverser housing. It provides for attachment of upper and lower hinged cowlings. Openings and fluid fittings are also provided for services to and from power plant (such as bleed air duct openings, hydraulic fluid and fuel fittings, etc.).

The engine fire extinguishing distribution nozzles and two of the three engine fire detection elements are fitted at lower portion of fixed cowl.

An access panel is located in the upper forward portion of right hand fixed cowl for anti-ice manual override access.

D. Aft Cowl Structure:

(See Figure 6.)

The aft cowl structure includes the exhaust nozzle and thrust reverser actuator and linkage. It consists of a forward fairing (fixed), an outboard stang beam fairing, an inboard scoop fairing and upper and lower thrust reverser doors.

E. Engine Fire Protection System:

See Section 2A-26-00, Fire Protection, for details on engine fire protection components.

3. Limitations:

A. Flight Manual Limitations:

There are no Flight Manual limitations established for the nacelle at the time of this revision.

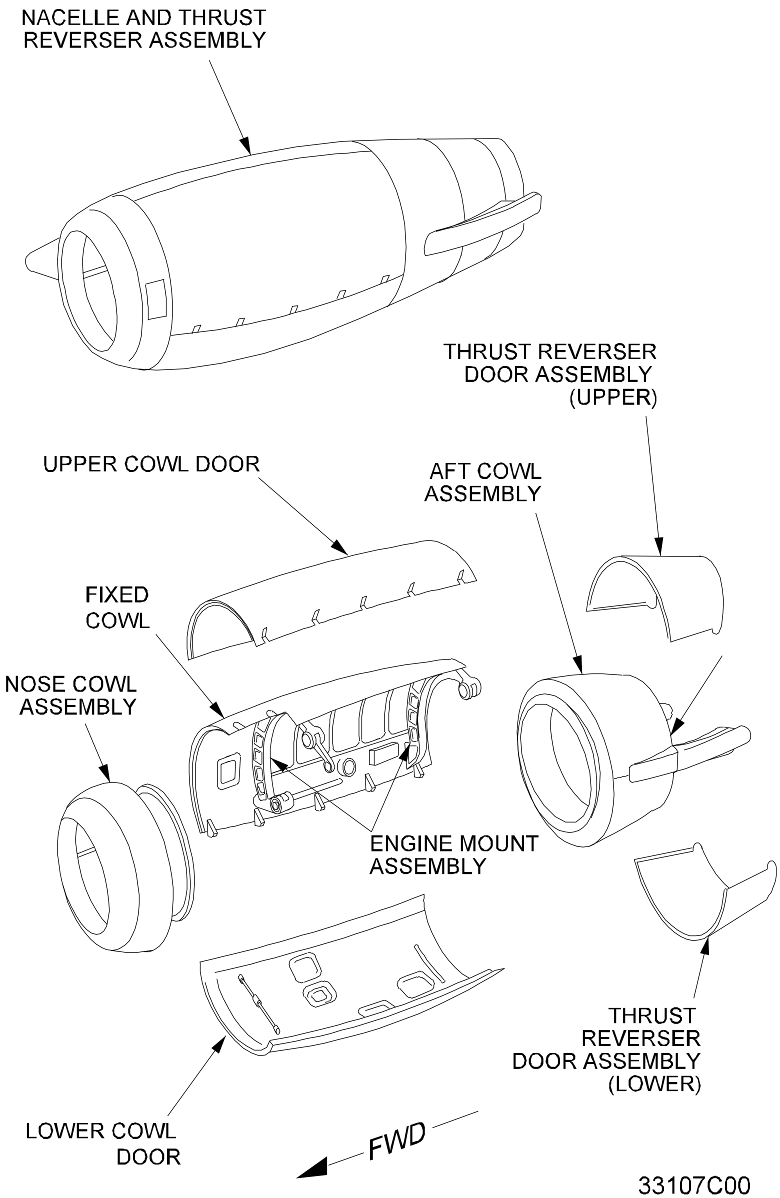
B. Other Limitations:

(1) **Lower Cowl Door Weight Limit:**

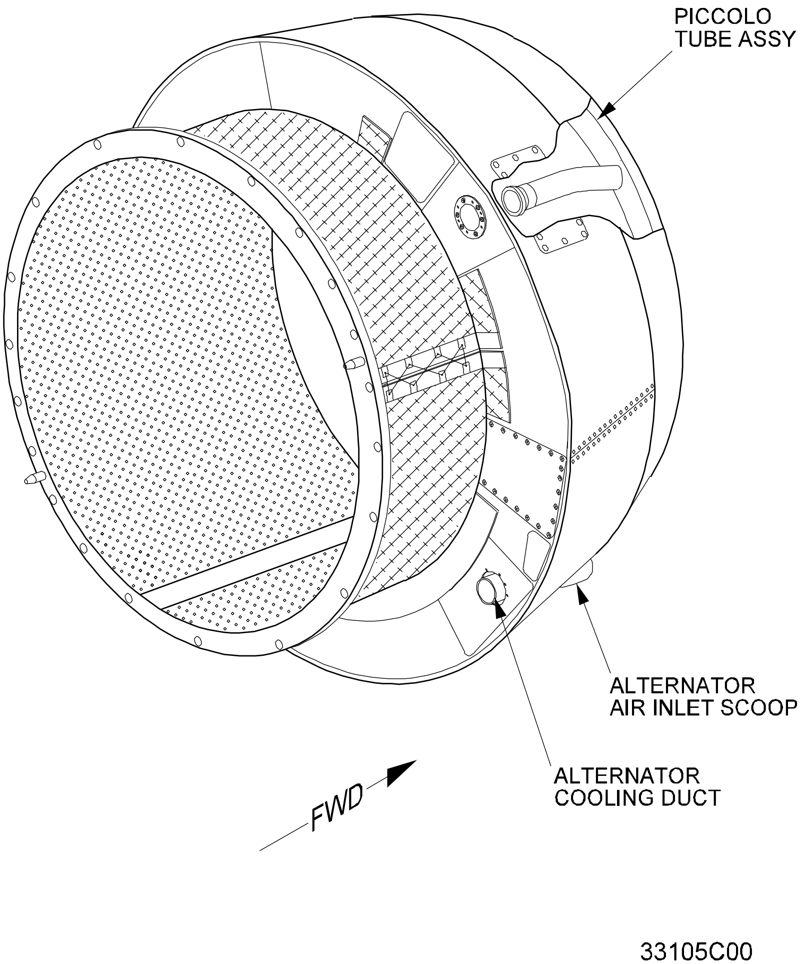
The lower cowl door is designed to support the weight of up to two persons of average weight and a small tool box.

(2) **Maximum Wind Velocity With Cowl Doors Open:**

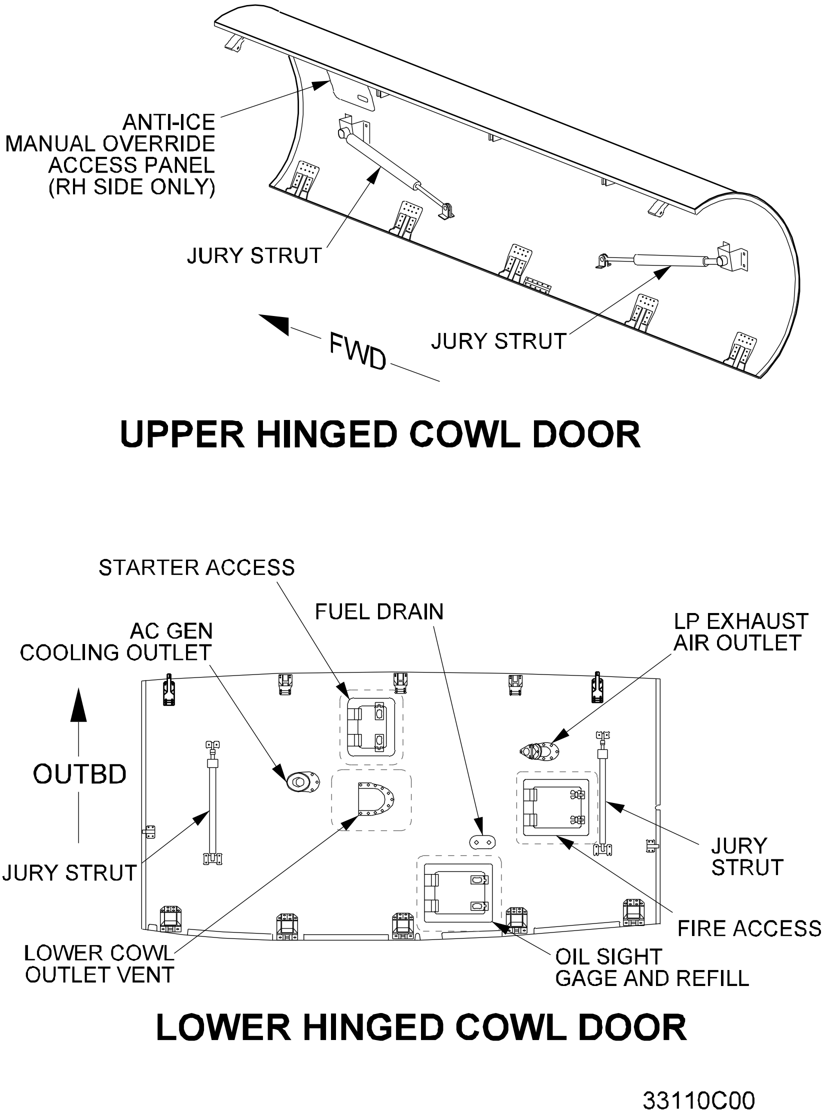
Open cowl doors are designed to withstand wind speeds of up to sixty (60) knots when properly secured with front and rear jury struts in the extended and locked position.



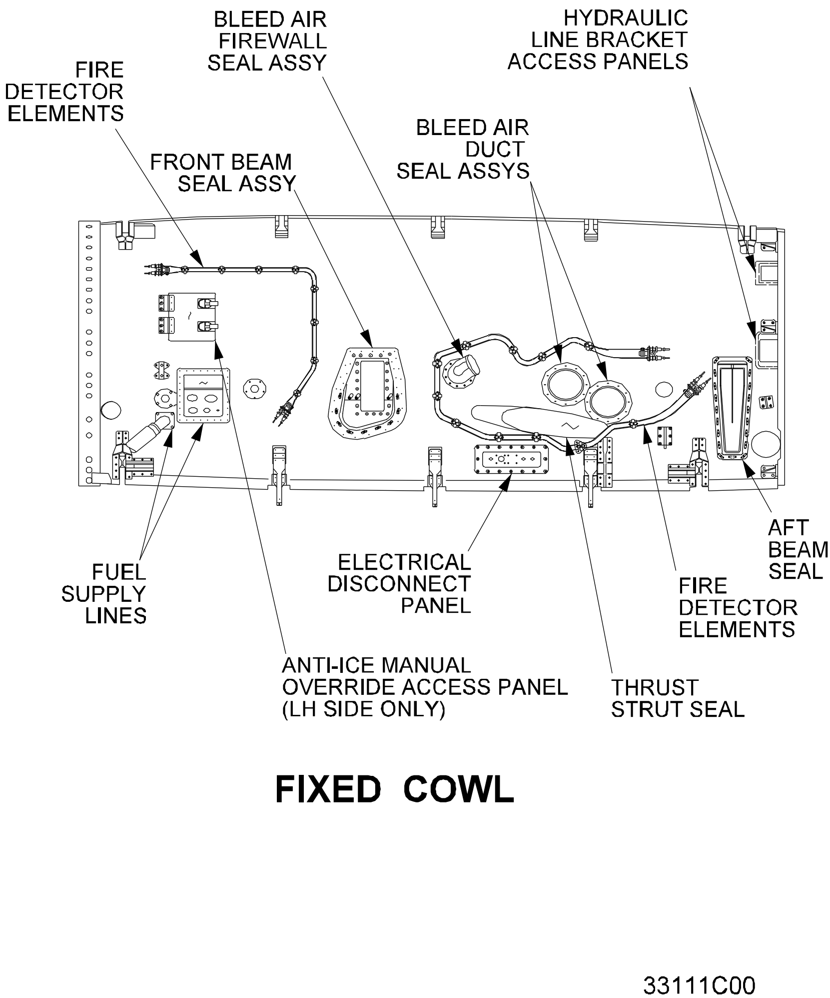
Nacelle Components
Figure 2



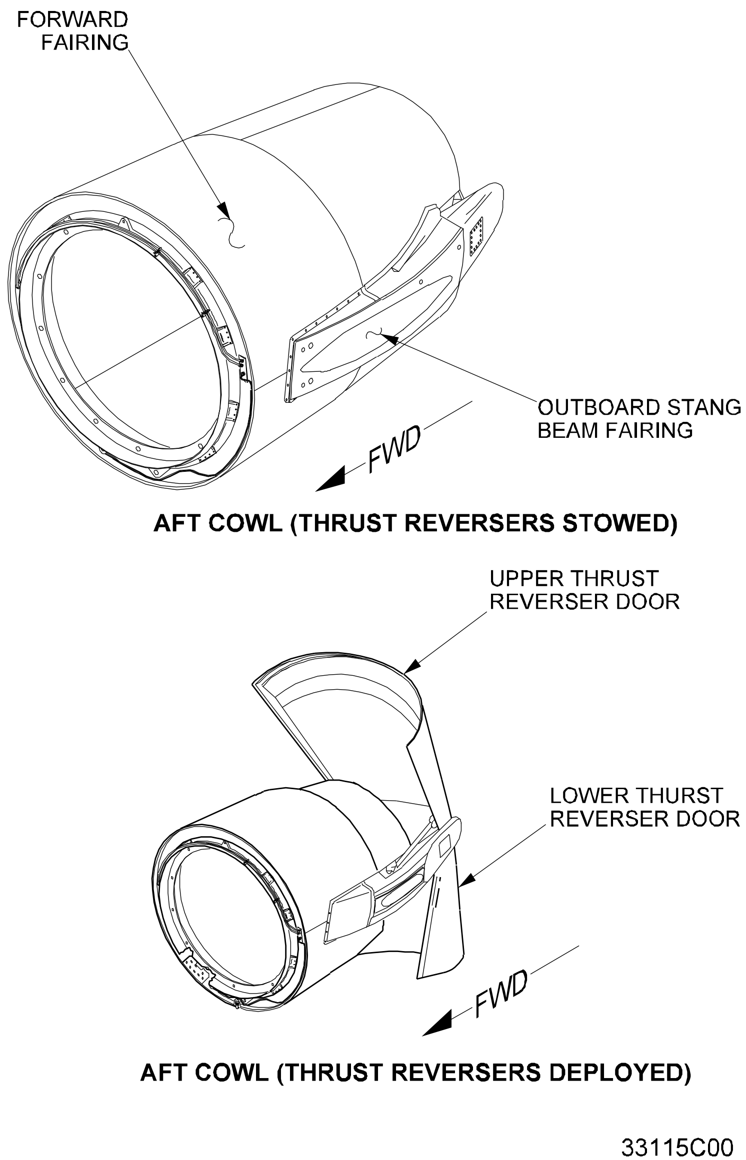
Nose Cowl Structure
Figure 3



Upper and Lower Hinged Cowl Doors
Figure 4



Fixed Cowl Structure
Figure 5



Aft Cowl Structure
Figure 6

2A-71-30: Mechanical Accessories

1. **General Description:**

The purpose of the Mechanical Accessories system is to provide services for the engine and aircraft not directly related to thrust production. An internal gearbox between the LP and HP compressors acts as a power takeoff to drive the external low-speed (LP) and high-speed (HP) gearboxes. The gearbox drive housing contains the front and rear LP compressor shaft bearings and the HP compressor shaft front bearing.

A series of gears within the gearbox transmit LP and HP compressor shaft torque to a pair of driveshafts. These driveshafts extend through the engine's immediate compressor case to drive their respective external gearbox. The LP compressor drives the low-speed gearbox and the HP compressor drives the high-speed gearbox.

A. **Aircraft Service Bleeds:**

- Environmental Control System (ECS)
- Wing Anti-Icing
- Crossbleed Starting
- Precooling of Air for Aircraft Systems

B. **Engine Handling Bleeds:**

- Cowl Anti-Icing
- Engine Stability, Surge Recovery and Flameout Protection

C. **Accessory Gearbox:**

- Power for Gearbox-Mounted Accessories
- Power for Starting and Cranking

2. **Description of Subsystems, Units and Components:**

A. **Aircraft Service Bleeds:**

(1) General:

During normal operations, engine compressor bleed air entering each system (left or right engine) is primarily extracted through the mid-stage (7th stage) check valve. When required (mid-stage pressures too low), mid-stage bleed air is assisted by high-stage (12th stage) bleed air by either augmenting (adding to) mid-stage airflow or by complete mid-stage to high-stage switching (12th stage taking over all pneumatic functions).

(2) Environmental Control System (ECS):

The environmental control system provides for pressurization, heating, cooling, ventilation and the means for reduction of humidity in flight or on the ground. True air conditioning is classified as heating or cooling as necessary to maintain a specific level of temperature within the occupied areas of the aircraft, regardless of the ambient temperatures or the operating conditions. Pressurization is the control over the pressure within the occupied areas.

(3) Wing Anti-Icing:

The wing anti-ice system consists of a leading edge which is fed by

a perforated duct called a piccolo tube. An open air space in the leading edge forms a passage for hot bleed air. The bleed air prevents formation of ice on the wing leading edges. Bleed air tapped from the bleed air manifold flows to the wing leading edges, via anti-ice valves, wing anti-ice ducts and piccolo tubes when the system is in operation.

(4) Crossbleed Starting:

After starting one engine from either APU or an external air supply, the other engine may be started by crossbleeding air from the running engine. Power is advanced on the running engine to achieve a minimum of 25 to 30 PSIG on the bleed air pressure indicator or ENGINE START synoptic page and the remaining normal start procedures are used. Upon completion of crossbleed starting, the engine is returned to idle.

(5) Precooling of Air for Aircraft Systems:

From the bleed air pressure regulator and shutoff valve outlet, regulated bleed air passes through the bleed air precooler heat exchanger located in the pylon of the aircraft, where it is cooled by fan air from the engine. Precooler bleed discharge temperature is controlled to a nominal 400° F by a transfer of heat to the cooler and lower pressure engine fan air that flows through the cooling air passages of the precooler heat exchanger. The cooling air is ducted overboard through the louvers in the lower surface of the engine pylon.

Bleed air from the bleed air precooler heat exchanger outlet enters the crossover manifold ducting in the tail compartment with its temperature and pressure controlled to 400° F and 40 PSIG (for most conditions). The crossover manifold is tapped to distribute bleed air to the various systems.

B. Engine Handling Bleeds:

(1) Cowl Anti-Icing:

The nose cowl assembly mounted on the engine air intake consists of a piccolo tube mounted inside where air is admitted directly from a tap off of the engine bleed air manifold. For anti-icing purposes, hot air is supplied to the nose cowl leading edge duct (top of cowl) and piccolo tube by a feed pipe coming from the bleed air manifold. A series of holes in the piccolo assembly allows hot air to flow around the insides of the nose cowl. This air is exhausted through holes on the inside of the cowl leading edge to the engine air intake.

The anti-icing air supply is controlled by a shutoff and pressure regulating valve, mounted on each engine. The valve in turn, is

controlled by an ON-OFF switch, one for each side, located on the cockpit overhead panel and labeled L COWL and R COWL.

NOTE:

Ice accumulation on the front of the engine fan and spinner can disrupt and restrict airflow into the engine. Large build ups of ice could increase the risk of structural and impact damage. The fan spinner uses a soft rubber point that distorts during engine operation to ensure ice accretion is shed centrifugally before it builds up. The fan blades use centrifugal force to shed ice and prevent accumulation during operation. This use of centrifugal force is known as the passive form of ice protection.

(2) Engine Stability, Surge Recovery and Flameout Protection:

The HP compressor control system allows for the stability of airflow in all ranges of RPM through the use of Variable Inlet Guide Vanes (VIGV's), a 7th stage bleed air valve and the Airflow Control Regulator (AFCR). The VIGV's and the 7th stage bleed air valve are operated together by the AFCR, which operates in response to input based on HP compressor inlet temperature and HP rpm.

At a low RPM, the VIGV's are at their maximum or "closed" angle (relative to engine airflow). The bleed valve, which is in the open position at this time, allows the bleed air to escape into the bypass duct and exit the engine through the fan air exhaust.

As engine RPM increases (approximately 70 to 80% HP RPM), the AFCR initiates movement of the VIGV's toward their "open" position, creating a minimized airflow angle of attack and closes the bleed air valve. Closing the valve channels 7th stage bleed air into the manifold the surrounds the HP compressor case. The VIGV's increase the angle of incidence (with decreased airflow) on the front stage of the HP compressor as engine RPM decreases. This prevents the flow of air back towards the front of the compressor which can create blade stall and compressor surge, as a result of the unstable airflow.

C. Accessory Gearbox:

(See Figure 8.)

(1) Low-Speed Gearbox:

Also referred to as the left gearbox, it is not considered a module and is mounted on the left side of the Intermediate Pressure Compressor (IPC) case. The gearbox is driven by the LP spool through the intermediate gearbox and has mounting provisions for the following:

- LP Shaft Governor
- LP Tach Generator

(2) High-Speed Gearbox:

Also referred to as the right gearbox, it is mounted on the lower right side of the IPC case. The gearbox is driven by a shaft from the

internal gearbox located between the IP and HP compressors and transfers power from the HP spool to the gearbox. It has mounting provisions for the following:

- Engine Alternator
- Centrifugal Breather Outlet
- Hydraulic Pumps
- LP Fuel Pump
- HP Fuel Pump
- HP RPM Tach Generator
- Fuel Flow Regulator
- Starter Motor
- Airflow Control Signal Transmitter

(3) Power For Starting And Cranking:

The high-speed gearbox transmits power from the starter motor to the engine during normal start/crank procedures.

3. Controls and Indications:

The Mechanical Accessories system has no controls and indications as a stand-alone system. See the following sections for specific controls and indications:

- Section 2A-21-00, Air Conditioning
- Section 2A-30-00, Ice and Rain Protection
- Section 2A-36-00, Pneumatics
- Section 2A-80-00, Engine Starting

4. Limitations:

There are no limitations established for the Mechanical Accessories system at the time of this revision.

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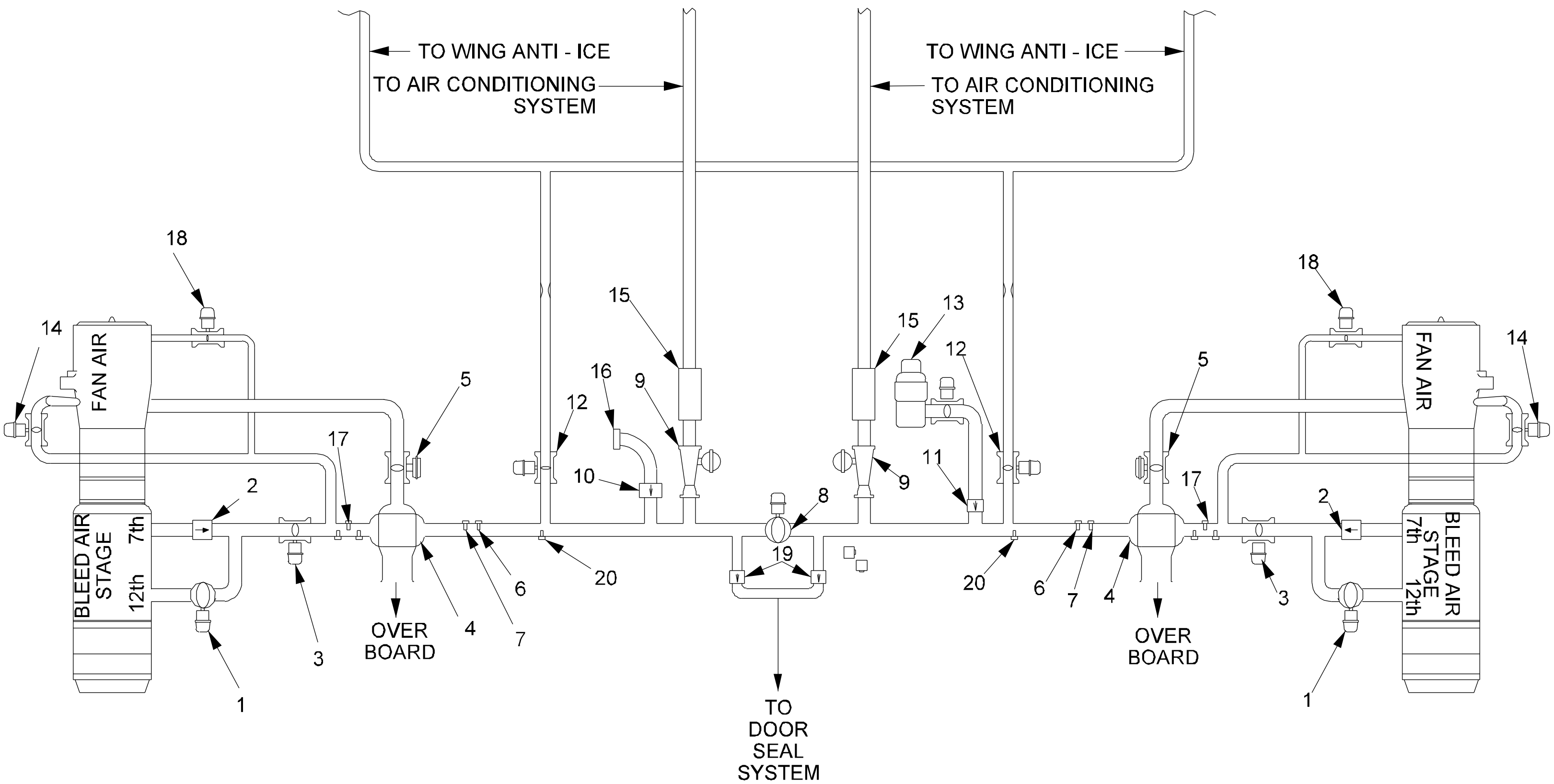

ITEM NO	DESCRIPTION	ITEM NO	DESCRIPTION
1	HP BLEED AIR PRESSURE REG & S/O VALVE	11	AUX POWER UNIT AIR CHECK VALVE
2	LP BLEED AIR CHECK VALVE	12	WING ANTI-ICE PRESSURE REG S/O & TEMPERATURE CONTROL VALVE
3	BLEED AIR PRESSURE REG & S/O VALVE	13	AUXILIARY POWER UNIT (APU)
4	BLEED AIR PRECOOLER HEAT EXCHANGER	14	AIR TURBINE STARTER & BLEED CONTROL VALVE
5	AIR MODULATING PRECOOLER FAN VALVE	15	CABIN/COCKPIT OZONE FILTER
6	PRECOOLER OUTLET TEMPERATURE SENSOR	16	EXTERNAL AIR FILTER
7	PRECOOLER TEMPERATURE CONTROL ANTICIPATOR SENSOR	17	COWL ANTI-ICE DUCT OVERHEAT (675 F) SWITCH
8	BLEED AIR ISOLATION S/O VALVE	18	COWL ANTI-ICE PRESSURE REGULATOR & SHUTOFF VALVE
9	AIR CONDITIONING SYSTEM SHUTOFF & FLOW CONTROL VALVE	19	CHECK BLEED AIR DOOR SEAL SYSTEM VALVE
10	EXTERNAL AIR CHECK VALVE	20	BLEED AIR OVERTEMP (550 F) SWITCH

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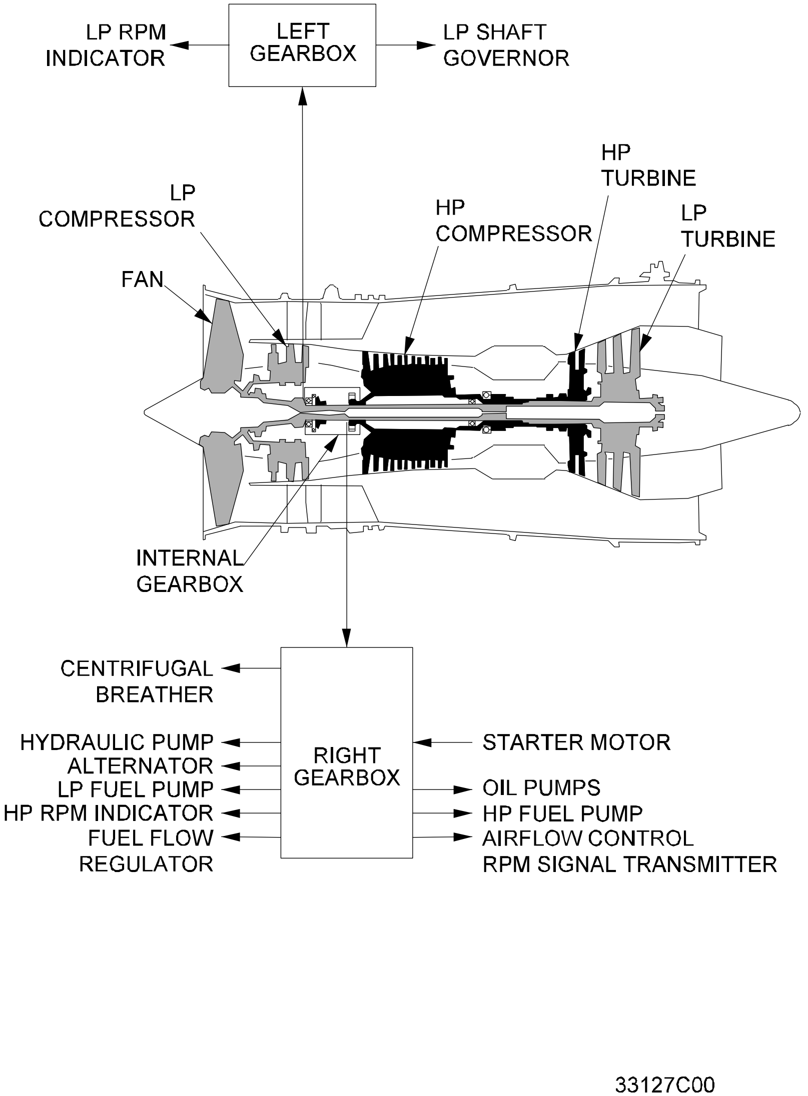
Bleeds System Simplified
Block Diagram
Figure 7



Accessory Gearbox Layout
Figure 8

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2A-73-10: Engine Fuel System

1. General Description:

The engine fuel system is a mechanical, all-speed governing system which controls fuel flow automatically to maintain a selected High Pressure (HP) speed and provide rapid, surge-free acceleration and deceleration control. A solenoid-operated, high idle setting ensures acceptable acceleration is available during final approach.

Fuel is supplied from the aircraft fuel system to an engine driven Low Pressure (LP) fuel pump which delivers it through the engine fuel cooled oil cooler, filter and flowmeter and to an engine driven variable output HP fuel pump. From the HP pump, fuel is delivered to a fuel regulator, which meters it into two separate flows to the spray nozzles. The main flow passes through the N1 shaft governor to the HP fuel cock, while the primary flow passes directly to the HP fuel cock. Both flows are then distributed respectively to the primary and main spray nozzles.

The HP fuel shutoff valve fuel cock has three positions: SHUT, START and OPEN. Only two of these positions, SHUT and OPEN, are utilized while operating the engine. The START position, although not selected, permits a supplementary fuel flow to the main spray nozzles to facilitate easier starting under certain conditions.

The N1 shaft governor prevents the LP compressor shaft from exceeding its limitation by trimming the fuel flow from the HP fuel pump.

2. Subsystems, Units and Components:

(See Figure 1.)

The engine fuel system consists of the following subsystems, units and components:

- Low pressure fuel pump
- Oil cooler
- Fuel filter
- High pressure fuel pump
- Low pressure shaft governor
- High pressure fuel shutoff valve
- Fuel tubes and manifolds
- Fuel nozzles

3. Description of Subsystems, Units and Components:

(See Figure 1.)

A. Low Pressure Fuel Pump:

The LP fuel pump attaches to the front face of the high-speed (HS) gearbox. It is sometimes known as the LP backing pump. The pump keeps the fuel pressure at the inlet to the HP fuel pump at a value high enough to prevent cavitation. The LP fuel pump has a splined shaft at its front end that engages a driving shaft in the HS gearbox. The basic design of the LP fuel pump is a housing with a fuel inlet and outlet port. A drains connector allows for fuel to go to the engine drains tank.

B. Oil Cooler:

The LP fuel pump delivers the fuel from the aircraft supply to the oil cooler. The LP fuel decreases the temperature of the oil in the cooler before the fuel goes to the fuel filter.

C. Fuel Filter:

The 10 micron LP fuel filter attaches to the left-hand side of the intermediate case. The basic design of the fuel filter is a housing which has a paper element in the inner chamber. The element removes unwanted matter from the fuel that comes from the fuel-cooled oil cooler and prevents foreign material from passing on to the HP pump.

D. High Pressure Fuel Pump:

The HP fuel pump attaches to the rear face of the high-speed gearbox. A shaft gear in the gearbox turns the drive shaft in the front of the fuel pump. The fuel pump receives LP fuel from the fuel filter and flowmeter. The pump then supplies the fuel, at a high pressure, to the Fuel Flow Regulator (FFR).

The HP fuel pump has a rotating assembly which has seven inclined hollow cylinders. The cylinders attach to an adjustable cam plate. The cam plate angle can therefore change to adjust the stroke of the pistons. A servo system controls the output of fuel to the FFR.

E. Low Pressure Shaft Governor:

The LP shaft governor limits the fuel flow to the fuel spray nozzle in some operating conditions. This ensures that the LP compressor shaft does not exceed its maximum speed limit.

F. High Pressure Fuel Shutoff Valve:

The HP fuel shutoff valve is a plunger-type valve operated by a rack and pinion. It attaches to the left-hand side of the HS gearbox with four bolts. In the full-flow position, the pilot flow and main flow of fuel go to the fuel manifolds. In the closed position the plunger closes off the flow of fuel to the manifolds, in this position the fuel, together with the servo-fuel for the HP fuel pump, goes to the LP outlet. The valve connects, through its operating lever and a control linkage, to a manual control in the crew compartment (see Figure 2).

G. Fuel Tubes and Manifolds:

Fuel tubes allow fuel to go from the inlet connector at the LP fuel pump through the subsequent units to the fuel spray nozzles. The tubes are made of stainless steel with ferrules welded on to their ends. The grooves in these ferrules are for rubber seal rings which prevent leakage of fuel when the tubes are installed. There are two fuel manifolds on the engine: the main and pilot fuel manifold. Short tubes, which have union nuts at each end, connect the manifolds to the fuel spray nozzles.

H. Fuel Nozzles:

Ten fuel spray nozzles supply a continuous spray of atomized fuel particles to the combustion chambers. Each fuel spray nozzle has two systems that supply fuel: a pilot system and a main system. The systems operate independently and give the correct combustion conditions for the full range of engine operation.

4. Controls and Indications:

(See Figure 3.)

NOTE:

A detailed description of the Engine Instruments and Crew Alerting System (EICAS) messages can be found in Section 5 of Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV.

Mounted on the fuel filter housing assembly are the following two pressure switches:

A. Fuel Pressure Low Warning Switch:

When a low pressure condition occurs, this 15 psi switch will illuminate the red L-R FUEL PRESS warning indication on the EICAS and/or the Standby Warning panel.

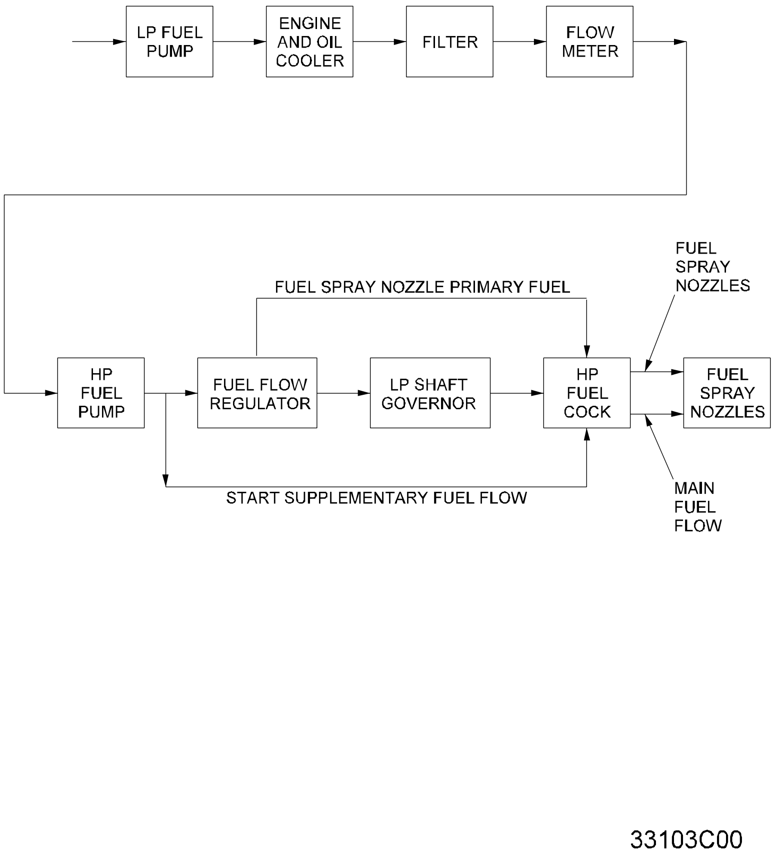
B. Fuel Filter Differential Pressure Switch:

When the fuel filter becomes obstructed, this 7 psid switch senses the difference in the fuel pressures upstream and downstream of the filter element and will illuminate the L-R FUEL FILTER warning indication on the EICAS and/or the Standby Warning panel.

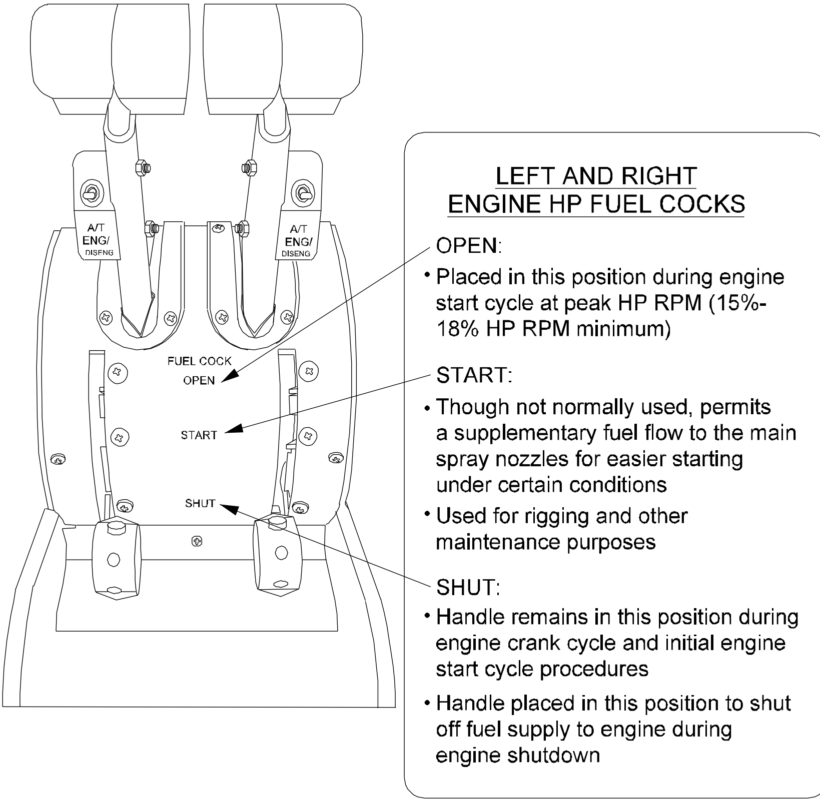
5. Limitations:

The maximum and minimum engine fuel temperatures are as follows:

- Maximum: 90°C (Fuel temperature up to 120°C for maximum of fifteen (15) minutes is permissible)
- Minimum: -40°C



Engine Fuel System Simplified Block Diagram
Figure 1



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HP Fuel Cock Positions
Figure 2

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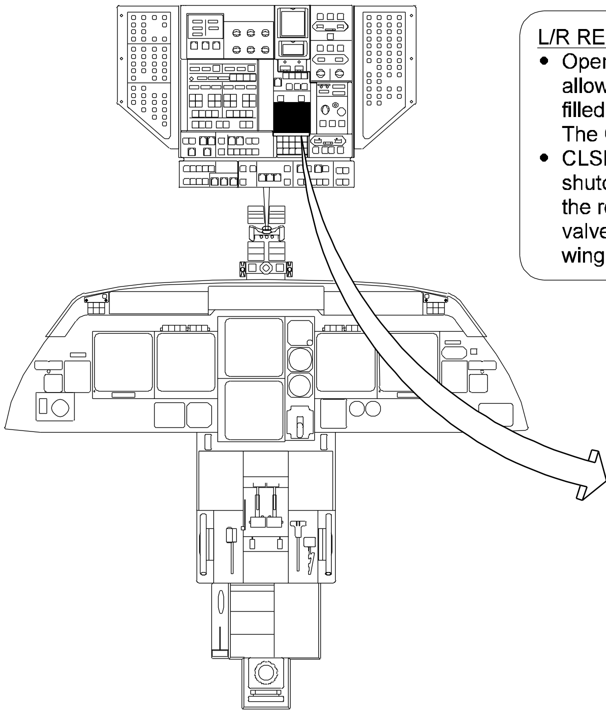
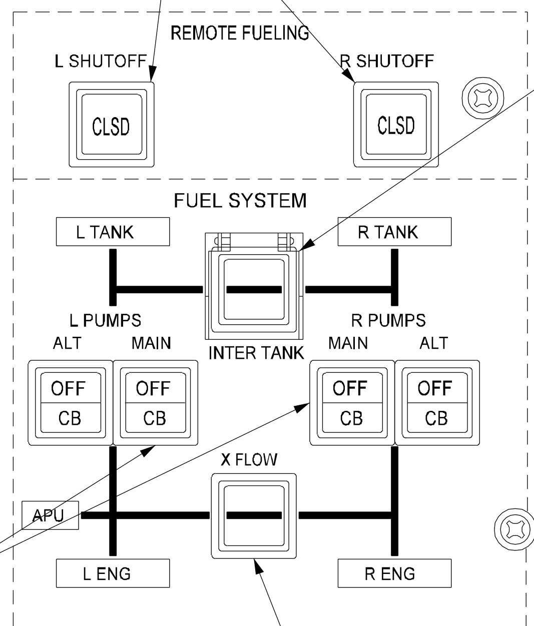


L/R REMOTE FUELING SHUT OFF
- Open (normal position): pressure fueling is allowed in the respective wing tanks until filled or fuel supply is shut off by other means. The CLSD switch legend is not illuminated.
- CLSD: supplies 28V DC to the respective fuel shutoff solenoid valve.The valve opens and the resultant back pressure against the pilot valve float system stops fuel flow to the respective wing tank. The CLSD switch legend is illuminated.

INTER TANK
- Closed (normal position): intertank valve (in right hopper) is closed. Bar in switch is extinguished.
- Open: intertank valve opens, combining left and right hoppers. Bar in switch is illuminated and a blue FUEL INT TNK OPEN advisory message is displayed on CAS.



L (or R) MAIN (or ALT) PUMPS
- On (normal position): respective boost pump operates. Amber OFF legend in switch is extinguished.
- OFF: respective boost pump is shut off. Amber OFF legend in switch is illuminated.
- CB: respective boost pump's power circuit breaker is open. Amber CB legend in switch is illuminated.

X FLOW
- Closed (normal position): crossflow valve (in left hopper) is closed. Bar in switch is extinguished.
- Open: crossflow valve opens, allowing fuel from fuel pump manifold pressure chamber on one side to flow to the chamber on the other side, enabling both engines to be operated from one tank system. Bar in switch is illuminated and a blue FUEL X FLOW OPEN advisory message is displayed on CAS.

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Engine Fuel System
Controls and Indications
Figure 3

2A-74-10: Engine Ignition System

1. General:

A. Description:

The ignition system, together with the cranking system, starts the engine on the ground and independently in flight (to re-start the engine).

The function of the ignition system is to supply ignition to the fuel and air mixture in the discharge of a series of sparks across the electrodes of the igniter plugs (the igniters are in combustion liners No. 4 and No. 8 in each engine). Combustion then passes between each liner through interconnectors to complete the combustion cycle. The aircraft electrical system supplies 28V DC to the high-energy ignition units, control is by the start master switch and the individual start selector switches.

B. Operation:

During the start sequence (MASTER START and ENG START switchlights to the ON position), the engine's start relay energizes to supply 28V DC power from the Essential DC bus to energize the respective ignition relay. The ignition relay then closes to supply 28V DC bus to the ignition unit(s).

NOTE:

On aircraft SN 1000 thru 1143 with ASC 151 and 1144 thru subsequent, only one ignition unit and igniter plug operate during ground starting, for example:

- Left Engine — No. 1 Ignition Unit
- Right Engine — No. 2 Ignition Unit
- With the ignition unit(s) energized, a green IGN annunciation appears next to the engine's HP RPM display.

As engine RPM reaches approximately 38 to 45%, the starter's centrifugal cutout switch opens to de-energize the start and ignition relays. The ignition unit(s) de-energize and the IGN annunciation clears.

Lifting the switch guard and pressing an AIR START IGN switchlight supplies 28V DC directly from the DC bus to the corresponding ignition units. With an AIR START IGN switchlight in on, the switchlight's blue ON caption illuminates and the green IGN annunciation appears next to the engine's HP RPM display on EICAS. When the ignition units are receiving power through the AIR START IGN switchlight they operate continuously.

C. Subsystems, Units and Components:

(See Figure 2.)

The Engine Ignition system is composed of the following units and components:

- Ignition Relay
- High Energy Transistorized Ignition Units (2 Per Engine)
- High Tension Leads (2 Per Engine)

- Igniter Plugs (2 Per Engine)

2. **Description of Subsystems, Units and Components:**

   A. **Ignition Relay:**

   During ground starting, the ignition relay is energized when the starter relay and pneumatic starter are energized. When energized, the relay provides power from the Essential DC bus through the #1 IGN and #2 IGN circuit breakers to the high energy ignition units. Each ignition unit is provided with DC power from a separate circuit breaker.

   B. **High Energy Transistorized Ignition Units:**

   The ignition units change a low voltage DC electrical supply into a High Energy (HE.) output. They can be energized independently of the engine starting system if a re-start in flight is required. The components of the unit are housed in a light alloy case. This case also carries a Light Duty (LD) receptacle to connect it to the aircraft electrical supply and a HE receptacle to connect it to the igniters.

   C. **High Tension Leads:**

   A high tension lead connects the ignition unit output to its igniter plug.

   D. **High Energy Igniter Plugs:**

   The engine has two HE igniter plugs, installed in the diffuser case at numbers 4 and 8 combustion liner positions. The HE ignition unit supplies power to the terminal connector of the plug. This electrical energy causes a high intensity spark across the electrodes of each plug and supplies ignition to the fuel and air mixture in the combustion chamber.

3. **Controls and Indications:**

   (See Figure 1.)

NOTE:

A description of the Engine Instruments and Crew Alerting System (EICAS) can be found in Section 5 of Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV.

   A. **Circuit Breakers:**

   The Engine Ignition system is protected by the following circuit breakers:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
L/R #1 IGN	P	I-8, I-9 **(1)**	ESS 28V DC Bus
L/R #2 IGN	P	J-8, J-9 **(1)**	ESS 28V DC Bus

NOTE(S):

(1) Depending on effectivity.

4. **Limitations:**

   A. **Flight Manual Limitations:**

   (1) For airplanes SN 1000 thru 1249 without ASC 304:

   The duty cycle time for continuous (airstart) ignition without ASC 304 is five (5) minutes ON and thirty (30) minutes OFF for cooling. There

is no limitation on the ignition when used in a thirty (30) seconds ON, thirty (30) seconds OFF cycle.

(2) For airplanes SN 1250 and subs and SN 1000 thru 1249 with ASC 304:

There is no duty cycle time limitation for continuous (airstart) ignition with ASC 304 installed.

B. System Notes:

Although selection of continuous ignition is not time-limited, it will reduce overall igniter life. It should be turned off as soon as the condition has returned to normal.

It is recommended that the Air Start Ignition be selected ON for landing on a runway with standing water, slush, or snow.

If volcanic ash is encountered, however, it is recommended that Airstart Ignition be selected OFF. Operation of igniters when flying through volcanic ash may create a “glassed over” condition and render them inoperative.

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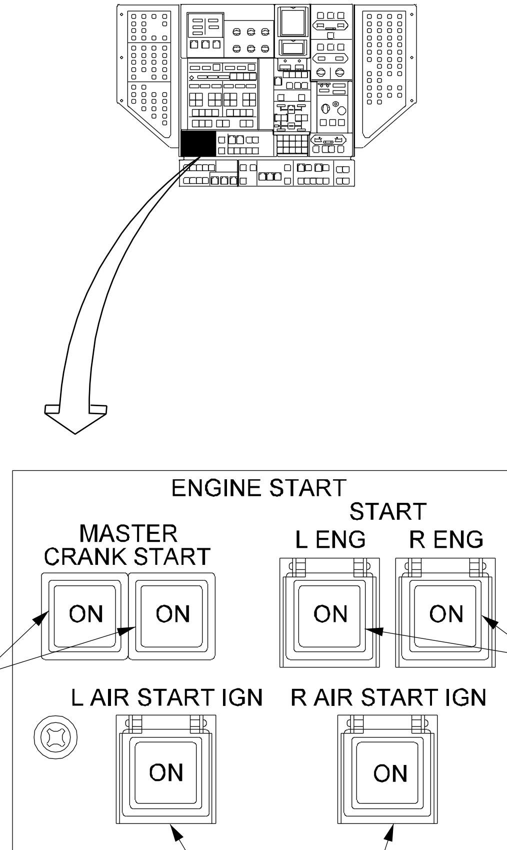

MASTER CRANK

OFF:
- Blue ON legend is extinguished
- De-energizes start relay
- Automatically closes isolation valve (on ground only)

ON:
- Blue ON legend illuminates
- Allows the START switches to energize the start relay without energizing the auto-ignition relay
- Allows for cranking of engine without ignition or starting of engine
- Automatically opens isolation valve (on ground only)

MASTER START

OFF:
- Blue ON legend is extinguished
- De-energizes start and auto-ignition relay
- Automatically closes isolation valve (on ground only)

ON:
- Blue ON legend illuminates
- Allows the START switches to energize the start relay and the auto-ignition relay
- Allows for cranking of engine without ignition or starting of engine
- Supplies ignition when starter is selected
- Automatically opens isolation valve (on ground only)

NOTE: For aircraft 1000 through 1155 (excluding 1034) having ASC 135, aircraft 1034, and aircraft 1156 and subsequent, selection of the MASTER CRANK or MASTER START switch closes the LEFT ECS PACK valve (on ground only). The valve will automatically reopen when the MASTER CRANK or MASTER START switch is deselected.

L/R ENG START

OFF:
- Blue ON legend is extinguished
- De-energizes start relay
  - Can be used to manually terminate power that energizes start relay (in lieu of automatic power termination at 42% HP RPM)

ON:
- Blue ON legend illuminates
- Start relay energized

NOTE: For aircraft 1000 through 1155 (excluding 1034) having ASC 135, aircraft 1034, and aircraft 1156 and subsequent, selection of the L ENG START or R ENG START switch closes the RIGHT ECS PACK valve (on ground only). The valve will automatically reopen when the start valve closes.

L/R AIRSTART IGN

OFF:
- Blue ON legend is extinguished
- De-energizes ignition units

ON:
- Blue ON legend illuminates
- Energizes ignition units directly from the Essential DC bus through the ignition unit CBs (this does not indicate ignition unit is operational)

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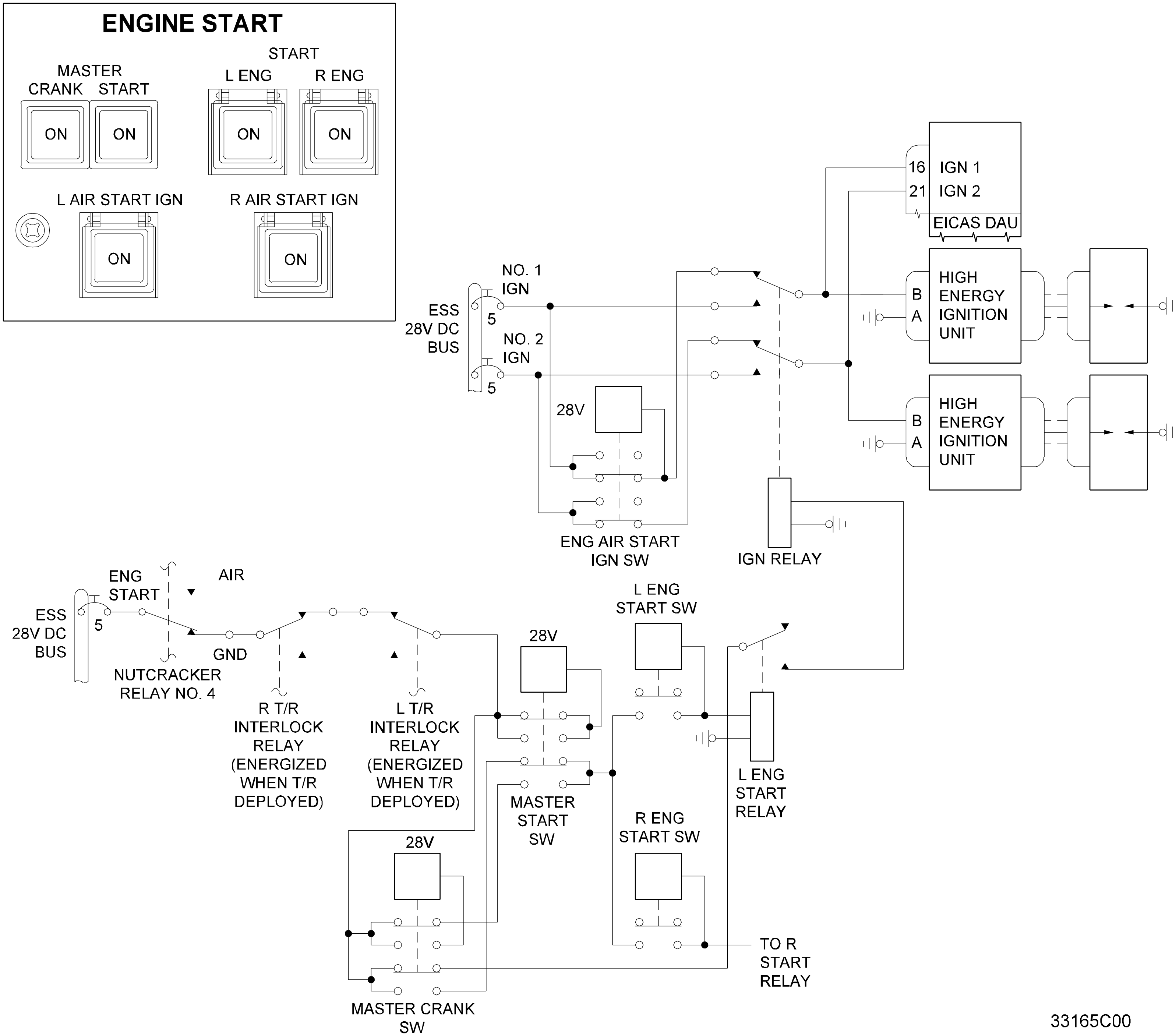
Engine Ignition System
Controls
Figure 1



Engine Ignition System
Simplified Block Diagram
Figure 2

2A-76-10: General

The engine controls system governs engine operation through all phases of flight. This is accomplished by both automatic and manual engine controls.

The engine controls system is composed ot the following subsystems:

- 2A-76-20: Electronic Engine Control System
- 2A-76-30: Engine Thrust Management System

2A-76-20: Electronic Engine Control System

1. General:

A. General Description:

The Tay engine control system, in addition to using the standard mechanical cable and push/pull rod systems, has three automatic controls, as follows:

- P_3 Limiter
- LP Governor
- Top Temperature Controller

2. Description of Subsystems Units and Components:

A. P_3 Limiter:

The P_3 limiter controls the maximum internal engine pressure to prevent over-boosting of the engine. The system is utilized when 238 PSI (maximum pressure) is exceeded.

B. LP Governor:

The LP governor controls the maximum LP spool speed by limiting the fuel flow to the fuel spray nozzle. It activates at approximately 94.6% LP RPM, and attempts to maintain the speed below 95.5% RPM (LP overspeed).

As the speed of the LP compressor shaft approaches the set value, the force caused by the centrifugal governor fly-weights moves the metering plunger against the balance spring which decreases the fuel flow through the governor until the plunger has a balanced condition. This also causes a decrease in pressure across the governor which causes the HP fuel pump to decrease the fuel quantity supplied. This prevents any increase in engine speed. When there is a reduction in the LP shaft speed, the LP governor drops off line and normal engine speed control resumes.

In addition to the LP governor operating as a power limiter or overspeed governor, it may also operate during full power conditions in the event of fuel flow regulator failure. During flight, the governor may also operate if the power settings are set at less than maximum. This is due to the increase of the speed differential between the HP and LP compressors with the increase of altitude and the fall in temperature.

C. **Top Temperature Controller:**

NOTE:

Not applicable for aircraft 1000-1319 (with ASC 394 incorporated) or aircraft 1320 and subsequent. The top temperature controller was removed during the above ASC and omitted in production thereafter.

The Top Temperature Controller (TTC) controls the maximum TGT of the engine during high-power operating conditions. The TTC activates at approximately 800° to 805° C and attempts to maintain the TGT below 820° C (maximum overtemperature).

If the temperature control switch in the crew compartment is set to ON, an increase in TGT more than the pre-set limit causes the amplifier to energize the motor in the actuator. The motor turns the output shaft which operates the throttle control mechanism to decrease the quantity of fuel from the HP fuel pump. This decreases the TGT.

When the engine goes to the over-temperature condition (more than the amplifier datum temperature), the TTC actuator is moved in the appropriate direction to decrease the fuel supply.

NOTE:

Should manual control of the engine become necessary, an external ON-OFF switch can be selected to OFF. In this condition, the TTC actuator is moved to the no-trim stop. In this position, a full range of control at the throttle levers is available.

3. **Flight Manual Limitations:**

There are no Flight Manual limitations established for the electronic engine control system at the time of this revision.

2A-76-30: Engine Thrust Management System

1. **General Description:**

The engine thrust management system provides a means of setting and controlling idle thrust, forward thrust and reverse thrust.

Engine thrust is indicated as a measurement of the ratio of fan pressure to intake total pressure. It is referred to as Engine Pressure Ratio (EPR). The air data computer supplies intake pressure information from four probes in the bypass duct. These signals are sent to a transducer, adjusted to a common EPR/thrust value for the engines and displayed on the EICAS (annunciator page). The engines are set to a predetermined EPR for takeoff, climb and cruise.

A. **Subsystems, Units and Components:**

The engine thrust management system is composed of the following subsystems, units and components:

- Power Levers
- Thrust Reverser Levers
- Airflow Control System

- Autothrottle
- Engine Synchronizer

2. Description of Subsystems, Units and Components:

A. Power Levers:

The power lever assembly consists of left and right power levers for setting forward thrust. The power levers are mechanically connected to the input lever on the fuel flow regulator to permit manual selection of fuel flow and engine RPM.

B. Thrust Reverser Levers:

The thrust reverser lever assembly for each engine is mounted in the cockpit control pedestal. It consists of a power lever (forward thrust lever) and a smaller lever called the reverse thrust lever, mounted and pivoting on the upper portion of the throttle lever. Both levers operate a bellcrank common to both levers through an interconnecting link assembly. The bellcrank, in turn, is connected to a cable sector wheel (at the cockpit floor line) by a push-pull rod. This makes it possible to utilize the same cable run, push-pull rods, and bellcranks when selecting engine speed. To prevent one lever from being moved when not desired, an interlocking mechanism is provided. This consists of a roller attached to the interconnecting link assembly riding in a contoured slot. The design permits moving the power lever from its IDLE position toward the maximum forward thrust position without moving the reverse lever. The roller in the slot is locked from following the contour of the power lever track. Only when the power lever is moved back to its IDLE position, can the reverse lever be moved in its upward/aft direction. This, in turn, moves the bellcrank in the same direction, thereby preventing it from being moved due to the roller being in the slot. Moving the reverse thrust lever from its stow (IDLE) position towards its maximum reverse thrust position will allow for an increase in HP compressor speed.

C. Airflow Control System:

The airflow control system is designed to prevent compressor stalls or surges during stable and transitioning compressor speeds. The system includes variable Inlet Guide Vanes (IGVs) located forward of the first stage of the HP compressor and a bleed strap surrounding the 7th stage of the HP compressor.

The varying positions of the IGVs ensure correct airflow to the first stage of the compressor. The bleed strap prevents “choking” of the HP compressor rear stage. This is accomplished by opening and dumping compressor air into the bypass and increasing airflow through the HP compressor front stages at the same time, thus preventing stalls.

The variable guide vanes and bleed valve are controlled by an airflow regulator and actuator. This unit responds to changes in EPR which is a function of HP compressor inlet temperature T_{26} and HP RPM. The actuator operates the inlet guide vanes and bleed valve through signals from the T_{26} and HP RPM.

D. Autothrottle:

(See Figure 1.)

NOTE:

On aircraft SN 1214 and subsequent, engine synchronization is a function of the autothrottle system. With the autothrottle engaged and ENGINE SYNC selected, The autothrottle will sync the engines when engine power is at limit EPR for CLB, CRZ or MCT ratings. With engines at limit EPR, the autothrottle will reduce RPM on the “high” engine to match either LP or HP RPM. The other engine will remain at the limit EPR.

(1) The PZ-800 performance computer aids the pilot in determining and selecting the optimum airspeed/engine setting for any given flight condition. The performance computer functions as an autothrottle computer to directly control aircraft throttle settings and allows the pilot to optimize thrust management.

The autothrottle uses the selected EPR rating as an upper limit for control in all modes of operation. Automatic rating selection is available, where the PZ-800 performance computer chooses the engine rating based on the phase of flight. Also, minimum engine limits are observed to avoid the nonlinear flat response area near idle throttle settings.

(2) The following is a brief description of how the autothrottle system operates in LP, HP and EPR synchronization:

- With autothrottles engaged, engine synchronizer functions are performed through performance computers, not the synchronizer system.
- When FLCH mode is selected or directed by climb power or idle (as appropriate) and autothrottles are engaged, engines are synchronized to EPR by performance computers regardless of engine synchronizer selection. If engine synchronizer is selected ON in this mode, the appropriate lights will be illuminated even though LP and HP synchronizers are inhibited. When not in FLCH mode with autothrottles and engine synchronizer selected to ON, engines are synced to LP or HP by performance computers through switch selection.
- When not in FLCH mode and engine synchronizer is not selected, engines will default to EPR synchronizer with autothrottles selected ON.

(3) Autothrottle Control:

There are two types of autothrottle control modes:

Speed:

- Cruise
- Pitch hold, climb or descent

- Vertical speed, climb or descent
- V path

Power:

- FLCH climb or descent
- FLCH takeoff
- FLCH go-around

When autothrottle is engaged in a speed mode, performance computers will equalize EPR unless LP or HP synchronizer is selected ON.

When autothrottle is engaged in a power mode, performance computers will equalize EPR regardless of LP or HP synchronize selection.

E. Engine Synchronizer:

(See Figure 2.)

NOTE:

Prior to activating the sync system, the engines should be manually synchronized by using the power levers and the engine RPM indicators.

- Aircraft 1000 thru 1213 excluding 1183:

  The engine synchronization system provides a means to automatically match the LP or HP RPM of the left (slave) engine to the right (master) engine over a limited, predetermined range. This limited range prevents the left engine from losing more than a fixed amount of RPM should the right engine be throttled back, shut down or otherwise lose RPM. The engine synchronizing system consists of an amplifier which is supplied with RPM signals from the HP and LP tach generators. These signals are computed and an output signal is sent to an actuator on the left engine. The actuator, in turn, moves the RPM/FUEL lever on the fuel regulator to either increase or decrease the RPM of the left engine to match the right engine. The left engine power lever is not affected by actuator motion.

- Aircraft 1183, 1214 and Subsequent:

  Engine synchronization is a function of the autothrottle system. When FLCH mode is selected or directed by climb power or idle, as appropriate, and autothrottles are engaged, engines are synchronized to EPR by performance computers regardless of engine synchronizer selection. If engine synchronizer is selected ON in this mode, the appropriate lights will be illuminated even though LP and HP synchronizers are inhibited. LP or HP sync will only be active when the autothrottle is in the Speed Control Mode, e.g., CRUISE, V/S, VPATH or PITCH HOLD mode.

3. Controls and Indications:

A. Circuit Breakers (CBs):

The engine thrust management system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
L SEC LOCK	P	G-7	ESS DC Bus

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
R SEC LOCK	P	H-7	ESS DC Bus
L T/REV CONTROL	P	I-7	ESS DC Bus
R T/REV CONTROL	P	J-7	ESS DC Bus
L T/R EMER STOW	P	K-7 or L-7 **(1)**	ESS DC Bus
R T/R EMER STOW	P	K-7 or L-7 **(1)**	ESS DC Bus
ENG SYNC	P	K-9	R MAIN DC Bus
L EPR 115V	CP	A-9	ESS AC Bus
R EPR 115V	CP	B-9	ESS AC Bus
A/T SERVO #1	CPO	C-9	L MAIN DC
A/T SERVO #2	CPO	D-9	R MAIN DC

NOTE(S):

(1) Depending on effectivity.

B. Crew Alerting System (CAS) Messages:

The HP SYNC or LP SYNC annunciator lights are available on the Engine Instruments Display page (EICAS). Illumination of either annunciator indicates which tach generators (HP or LP) are currently being used.

Caution (Amber) Messages:

CAS Message:	Cause or Meaning:
AT OFF	Autothrottle disconnected

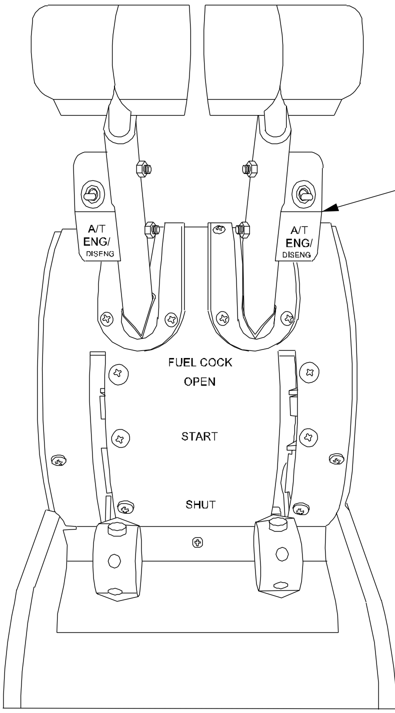
Advisory (Blue) Messages:

CAS Message:	Cause or Meaning:
AT ENGAGE INHIBIT	Attempt is made to engage autothrottle under any of the following conditions: • A/T disconnect button active • A/T not armed on flight guidance control panel • Both engines not running • EPR below 1.17 • Isolation valve open
AT 1-2 FAIL	Indicated autothrottle has failed, autothrottle will disconnect.
AT NOT IN HOLD	Airplane speed has exceeded 60 KCAS with autothrottle engaged on takeoff and autothrottle servos are not in hold.

4. Limitations:

A. Flight Manual Limitations:

Engine synchronizer must be OFF for takeoff and landing.



A/T ENG/DISENG

ENG (ENGAGED):

- Default SYNC is EPR
- HP or LP SYNC active only if autothrottle is in a speed control mode
- During takeoff, enters HOLD mode at 60 knots and remains until 400 feet AGL
- All engine SYNC functions (HP, LP EPR) are performed through the autothrottle computers
- When not in FLCH mode and engine synchronizer is not selected, engines will default to EPR synchronizer

DISENG (DISENGAGED):

- Disengages autothrottle system allowing for manual operation

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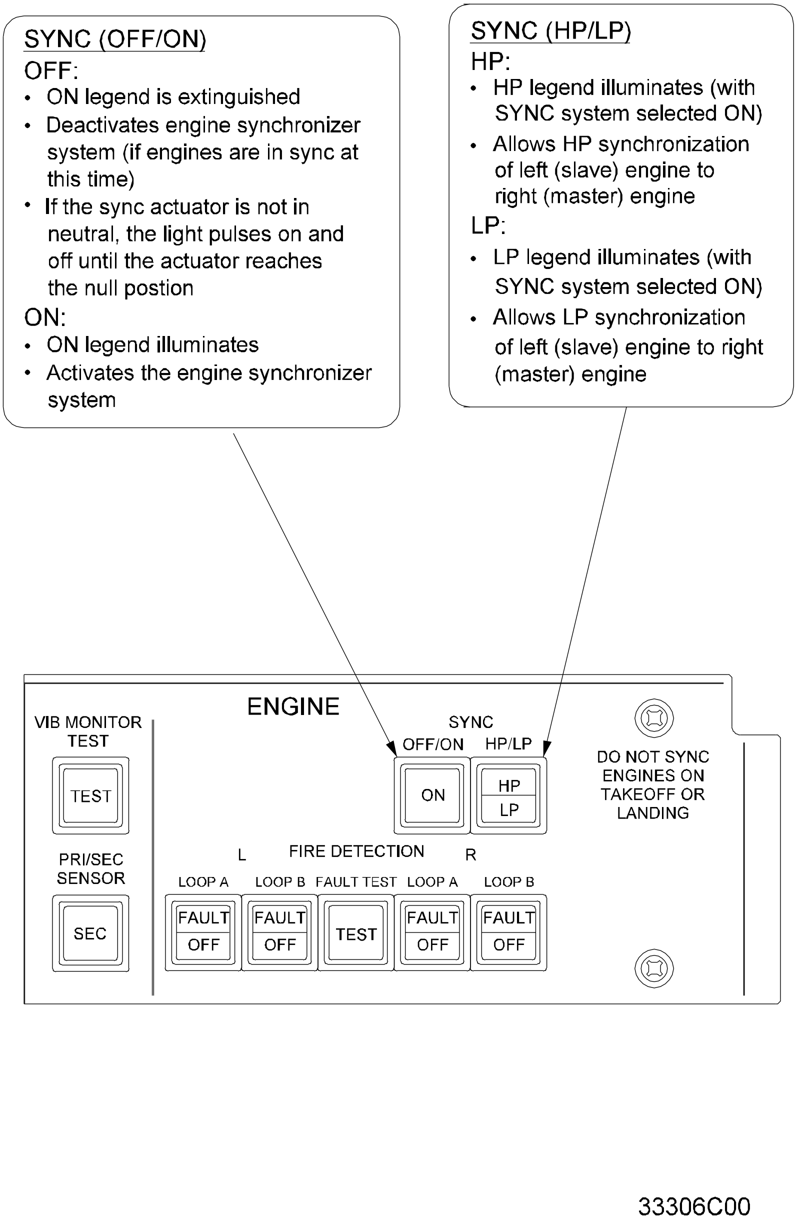
Autothrottle Control
Figure 1



Engine Sync Control Panel
Figure 2

2A-77-10: Engine Indication System

1. General:

The Engine Indication system generates data from the engines to provide the flight crew with operating parameter indications and alerts classified as advisory, caution or warning.

Refer to the system descriptions listed as follows during the study of this section:

- 2A-31-50: Central Display System
- Engine Instruments and Crew Alerting System (EICAS)

NOTE:

A description of the Engine Instruments and Crew Alerting System (EICAS) with specific information on displays of the Engine Parametric data and/or Engine Discrete Status data can be found in Section 5 of Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV.

A. Subsystems, Units and Components:

The Engine Indication system is composed of the following subsystems:

- Engine Pressure Ratio (EPR) Indication System
- Engine Speed Indicating System
- Turbine Gas Temperature (TGT) Indication System
- Engine Vibration Monitor (EVM) Indication System
- Engine Fuel Indicating System
- Engine Oil Indication System
- Standby Engine Instruments (Digital) Indication System

2. Description of Subsystems, Units and Components:

A. Engine Pressure Ratio (EPR) Indication System:

Engine pressure ratio (EPR) indication is displayed on the center instrument panel through the Engine Instrument and Crew Alerting System (EICAS) and the standby engine instruments indicator. These indicators provide an indication of engine power in the form of ratio of engine fan duct total pressure to intake total pressure which is commonly known as EPR.

The EPR transmitter has a single pressure transducer to measure engine fan duct pressure. Each EPR transmitter receives free stream total pressure via an ARINC 429 data bus from each of the two air data computers aboard the aircraft. It outputs EPR data, fan duct pressure, flight line fault codes and shop fault codes on two ARINC 429 buses to the data acquisition unit for EICAS EPR display and to the standby engine instruments indicator for EPR digital readout.

B. Engine Speed Indicating System:

(1) Low Pressure (LP) Speed:

Engine LP turbine RPM indication is displayed on the center instrument panel through the EICAS and standby engine instruments indicator. The LP tach generator is installed to and driven by the low speed gearbox providing signals directly to the appropriate EICAS Data Acquisition Unit (DAU) and standby engine instrument panel input. These signals give an indication of LP compressor shaft speed in percent.

LP tachometer signals are also provided to the EVM system for processing and displaying turbine vibration and to the engine synchronizer system for synchronizing LP turbines when desired.

Output of each HP tach generator is fed directly into the EICAS. Signals from the right engine HP tach generator are sent to EICAS DAU #2 and signals from left engine HP tach generator are sent to EICAS DAU #1.

Each tachometer generator also sends signals to the standby engine instrument signal conditioner to provide a digital display on the standby engine instruments indicator when the indicator switch is placed in MAN or AUTO mode when EICAS is not operational.

(2) High Pressure (HP) Speed:

Engine HP turbine RPM indication is displayed on the center instrument panel through the EICAS and the standby engine instruments indicator. The HP tach generator is installed to and driven by the high speed gearbox providing signals directly to the appropriate EICAS DAU and standby engine instruments indicator input. These signals give an indication of HP compressor shaft speed in percent.

HP tachometer signal inputs are also provided to the EVM system for processing and displaying turbine vibration and to the engine synchronizer system for synchronizing HP turbines when desired.

Output of each HP tach generator is fed directly into the EICAS. Signals from the right engine HP tach generator are sent to EICAS DAU #2 and signals from left engine HP tach generator are sent to EICAS DAU #1.

Each tachometer generator also sends signals to the standby engine instrument signal conditioner to provide a digital display on the standby engine instruments indicator when the indicator switch is placed in MAN or AUTO mode when EICAS is not operational.

C. Turbine Gas Temperature (TGT) Indicating System:

The TGT indication is displayed on the center instrument panel on the EICAS and standby engine instruments indicator. Nine dual element thermocouples are mounted equally spaced around the periphery of the forward section of the turbine outer case. The thermocouples are connected in parallel and coupled to a set of terminals in the junction box located at bottom of each engine. The thermocouples are used for both top temperature control and TGT indication. The thermocouples are located in the LP turbine stage one nozzle guide vanes and protrude into the hot exhaust gas stream providing a signal for TGT indication. Due to temperature scatter which exists within an engine, thermocouples will sense different temperatures. The temperature indicated is the average of

the nine positions.

Output from each engine thermocouple junction box is fed directly into the EICAS. Signals from left engine junction box are sent to EICAS DAU #1 and signals from right engine junction box are sent to EICAS DAU #2.

Each junction box also sends signals to the associated top temperature control amplifier and standby engine instrument signal conditioner to provide a digital display on the standby engine instruments indicator when the indicator switch is placed in MAN or AUTO mode when EICAS is not operational.

D. Engine Vibration Monitoring (EVM) Indication System:

(1) General:

The purpose of the vibration monitoring system is to provide crew with a means of continuous monitoring of balance of rotating assemblies in engine in order to detect a possible internal failure which could result in an engine failure. The EVM system is capable of early detection of such failures which can greatly reduce cost of engine overhaul.

(2) Operation:

The system consists of two sensors (primary and secondary) mounted side by side on the intermediate compressor of engine. The sensors are positioned to pick up vibration emanating from both LP and HP rotating assemblies. The two units provide system redundancy and provide for a second measurement point. The sensor signals are processed by a Signal Conditioning Unit (SCU) in the forward port radio rack behind the main entrance door. Each of the four sensors (two per engine) are conditioned independently in the SCU, providing signal redundancy from the sensor through the first stage of the SCU. One signal per engine (primary or secondary sensor) is inputted into parallel tracking filters which use HP and LP tach signals to follow the individual spools. This results in two EICAS display readings per engine from one sensor: LP imbalance and HP imbalance. In the event of higher than normal readings, the secondary sensor can be selected to verify the high reading; therefore, there will be four readouts (two per engine) which present the degree of rotor imbalance for each spool. Each engine will have two readouts from one sensor and the aircrew may select the alternate sensor with the SENSOR-PRI/SEC switch located on the cockpit overhead panel. The SCU output is 0-3 volts DC corresponding to 0-3 inches per second of vibration in velocity units. The EICAS will digitize these four signals and display each in digital form.

The system also consists of a VIB MONITOR - TEST switch located on the cockpit overhead panel. When depressed a signal of appropriate amplitude and frequency is applied simultaneously to all charge amplifier inputs for display by EICAS indicators. This test function will input two units of vibration to each sensor at 120 Hz and

should read 2.0 IPS on the display on all four readouts. For 2.0 IPS display to be valid, test must be completed prior to engine start.

NOTE:

If test mode is inadvertently activated when engines are operating, existing display will show an increase of approximately 2.0 IPS. The press-to-test function and display are not considered valid in this condition.

E. Engine Fuel Indication System:

(1) Engine Fuel Filter Warning:

L FUEL FILTER and R FUEL FILTER messages are provided to the EICAS to inform the aircrew that fuel filter blockage is taking place and that the filter element should be checked. On airplanes 1000 - 1252 not having SPZ-8400, L FUEL FILTER and R FUEL FILTER warning lights (located on standby warning lights panel) will also alert the aircrew to this blockage condition if standby warning light panel switch is placed in MAN mode or in AUTO mode when EICAS is not operational. These two warning systems are controlled by their respective fuel differential pressure switches mounted on each engine low pressure fuel filter. The differential pressure switch for each side is subjected to both fuel inlet and outlet pressures and a variation between the two pressures in excess of 7 ±0.3 PSI will complete an electrical circuit to appropriate warning indication in cockpit. The circuit will remain completed until differential pressure decreases to a value approximately 1 PSI below that at which circuit was completed.

Power from the essential 28V DC bus comes through the WARN LTS PWR #14 and WARN LTS PWR #15 circuit breakers to the left and right side fuel differential pressure switches, respectively. As long as a differential pressure of less than 17 ±0.3 PSI exists across the filter circuit to the warning system (EICAS and on airplanes 1000 - 1252 not having SPZ-8400, standby warning lights) the switch will be open. When a filter blockage occurs and a differential pressure of more than 7 ±0.3 PSI exists across filter, the switch will close. A circuit will then exist to EICAS to provide L or R FUEL FILTER readout from the essential 28V DC bus. On airplanes 1000 - 1252 not having SPZ-8400, a circuit will also exist to standby warning lights panel L FUEL FILTER or R FUEL FILTER light if standby warning lights panel switch is in MAN or AUTO mode and EICAS is not operational. The circuit to these warning systems will remain until differential pressure across the filter decreases to a value approximately 1 PSI below the actuating point.

(2) Engine Fuel Flow Indication:

Fuel flow indication is displayed by the EICAS and the standby engine indicator. The fuel flow transmitters are located on left side of each engine in areas of warmer fuel temperatures, thereby eliminating the need for specialized de-icing equipment. Each transmitter provides an electrical signal that is directly proportional to mass flow rate of fuel and transmits this signal to the EICAS and the standby engine indicator.

Output of each transmitter is fed directly into the EICAS with signals from the right engine fuel flow transmitter being sent to EICAS DAU #2.

Each transmitter also sends signals to the standby engine instrument signal conditioner to provide a digital display on the standby engine indicator when the indicator panel switch is placed in MAN or AUTO mode when EICAS is not operational.

(3) Engine Fuel Low Pressure Indication:

R FUEL PRESS LOW and L FUEL PRESS LOW readouts are provided through the EICAS to inform the aircrew of an engine fuel low pressure condition. On airplanes having SPZ-8000, R FUEL PRESS and L FUEL PRESS warning lights (located on the standby warning lights panel) will also alert the aircrew to this low pressure condition if the standby warning lights panel switch is placed in MAN or AUTO mode when EICAS is not operational. These two warning systems are controlled by their respective pressure switches mounted on low the pressure fuel filter and by outputs from the fuel boost pump logic unit. The low fuel pressure switch monitors outlet pressure at the fuel filter. If fuel pressure falls below 15 ±0.40 PSI, an electrical circuit is completed to the appropriate warning indicator in the cockpit. Power for this system comes from the essential 28V DC bus.

(4) Fuel Temperature Indication:

A fuel temperature bulb installed in the low pressure fuel outlet of the engine oil cooler gives an indication of engine fuel temperature through the EICAS. The temperature bulb contains a platinum element housed in a tube which is connected to an electrical receptacle. This receptacle then connects to the indicator circuit.

The resistance value of the bulb element changes with an increase or decrease in fuel temperature. A proportionate change in current is then transmitted to the EICAS with signals from the right engine fuel temperature bulb sent to the DAU #2 and signals from the left engine fuel temperature bulb sent to DAU #1.

F. Engine Oil Indication System:

(1) Engine Oil Temperature:

Engine oil temperature indication is displayed on the center instrument panel on the EICAS. An oil temperature bulb is located on the outlet side of each engine oil cooler to transmit signals to the EICAS. The temperature bulb contains a platinum element housed in a tube which is connected to an electrical receptacle which connects to the indicator circuit.

The resistance value of the bulb element changes with an increase or decrease in oil temperature. A proportional change in current is then transmitted to the EICAS with signals from right engine oil temperature bulb sent to DAU #2 and signals from left engine oil temperature bulb sent to DAU #1.

(2) Engine Oil Pressure:

Oil pressure indication is displayed by the EICAS. Each engine is equipped with an oil pressure transmitter mounted on rear of the oil

tank. The transmitters provide electrical signals to a dual signal conditioner located in the left hand radio rack, top shelf. From here, signal outputs are routed to EICAS.

The engine oil pressure transmitter consists of a twin stator armature connected to a bellows housed in a cylindrical case. Engine oil is admitted to the bellows chamber. Variations in the pressure of the oil system causes the bellows to expand or contract. This acts upon and causes movement of the armature assembly thereby causing changes in the ratio of the voltage in the two legs of the transmitter. The dual signal conditioner then outputs a linear DC analog signal to the EICAS, proportional to the ratio of the voltage from the transmitter.

Power for the signal conditioners is derived from the Essential #2 26V AC transformer bus via the R ENG OIL PRESS circuit breaker and from the Essential #1 26V AC transformer bus via the L ENG OIL PRESS circuit breaker. Both circuit breakers are located on the copilot's circuit breaker panel. The signal conditioner output is a linear analog voltage of 0-10V DC representing 0-75 PSI into a load of 10K ohms, with an accuracy of ±2% over the full range of 0-75 PSI. The output is routed to the EICAS DAU #2 for right engine indication and DAU #1 for left engine indication.

(3) Engine Low Oil Pressure Warning:

R OIL PRESS LOW and L OIL PRESS LOW readouts are provided through the EICAS to inform the aircrew of an engine oil low pressure condition. On airplanes having SPZ-8000, R OIL PRESS and L OIL PRESS warning lights (located on standby warning lights panel) will also alert crew to this low pressure condition if standby warning lights panel switch is placed in MAN or AUTO mode when EICAS is not operational. These two warning systems are controlled by their respective pressure switches mounted on rear of the oil tank.

The engine low oil pressure warning switch consists of a diaphragm, a push rod and electrical contacts. Holes in the mounting flange allow oil to enter from the diaphragm chamber. At and above the switch setting of 15 ±0.40 PSI, oil pressure is sufficient to move diaphragm and push rod arrangement holding the electrical contacts of the switch open. When oil pressure drops, the switch closes.

Power from essential 28V DC bus comes through WARN LTS PWR #14 and WARN LTS PWR #15 circuit breaker to right and left oil pressure warning switches, respectively. Both circuit breakers are located on the pilot's circuit breaker panel. When oil pressure falls below 15 PSI and the switch closes, a circuit will then exist to EICAS to provide a L or R OIL PRESS LOW readout from the Essential 28V DC bus. On airplanes having SPZ-8000, a circuit will also exist to standby warning lights panel L OIL PRESS or R OIL PRESS light, if

standby warning lights panel switch is in MAN or AUTO mode and EICAS is not operational.

NOTE:

For Flight Data Recorder System (FDRS), airplanes 1183 and Subsequent, a signal path branches from right and left oil pressure warning switches. This signal controls the on an off power to FDRS.

(4) Engine Oil Filter Warning:

An engine oil filter Differential Pressure Indicator (DPI) assembly installed at the rear face of the engine oil tank contains two separate indicator assemblies, a sensing switch assembly and a filter maintenance indicator (pop-out indicator) assembly. If there is a blockage in the engine oil filter, the switch assembly sends an electrical signal to the EICAS, providing L OIL FILT BPASS and R OIL FILT BPASS warning displays.

If the oil filter becomes blocked, pressure of the high pressure oil (from upstream side of the engine oil filter) will increase and pressure of the low pressure oil (from the oil filters on the downstream side) will decrease. This causes a piston/magnet in the assembly to compress its spring and move away from an actuating lever which operates a microswitch completing a circuit to the EICAS. When pressure drop across filter is more than 30 ±4.5 PSI, the actuating lever is free to operate the microswitch. The assembly has a thermal lockout leaf which ensures that the sensing switch does not operate below 29°C.

G. Standby Engine Instruments (Digital) Indication System:

Engine pressure ratio, LP tachometer, HP tachometer, fuel flow and turbine gas temperature indications for the left and right engine are displayed on the standby engine instruments indicator located on the pilot's instrument panel. Signals from each indication system are transmitted to a signal conditioner where they are converted to a digital readout for display on the standby engine instruments indicator. The instrument digital display brightness is controlled by STBY INSTR control knob on the copilot's cockpit lighting control panel. Clockwise rotation of knob increases light intensity and counterclockwise rotation decreases intensity.

Power for operation is derived from Emergency 28V DC bus through the R STBY ENG INSTR and L STBY ENG INSTR circuit breakers, both located on copilot's circuit breaker panel.

Signals from each indication system (EPR, LP tach, HP tach, fuel flow and TGT) are transmitted to the standby engine instrument signal conditioner to provide a digital display on the standby engine instruments indicator when

the indicator switch is placed in MAN or AUTO mode when the EICAS is not operational.

NOTE:

The digital readouts on the standby engine instruments indicator are tested by operation of WARN LT-TEST switch on pilot's or copilot's lighting control panel. When this switch is depressed, left and right digital readouts of 1888 and decimals are displayed for EPR, LP tach, HP tach and TGT and an 8888 readout is displayed for fuel flow.

3. Controls and Indications:

A. Circuit Breakers (CBs):

The Engine Indication system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
ENG VIB MONITOR	CP	B-8	L MAIN AC Bus
L ENG OIL PRESS	CP	A-12	ESS AC Bus
R ENG OIL PRESS	CP	B-12	ESS AC Bus
L STBY ENG INSTR	CP	A-11	AFT E-BAT Bus
R STBY ENG INSTR	CP	B-11	R MAIN DC Bus
WARN LTS PWR #14	P	I-4 or I-5 **(1)**	ESS DC Bus
WARN LTS PWR #15	P	I-5 or J-5 **(1)**	ESS DC Bus

NOTE(S):

(1) Depending on effectivity.

B. Crew Alerting System (CAS) Messages:

Warning (Red) Messages:

CAS Message:	SWLP Indication:	Cause or Meaning:
L FUEL FILTER	L FUEL FILTER	Left fuel filter clogging.
R FUEL FILTER	R FUEL FILTER	Right fuel filter clogging.
L FUEL PRESS LOW	L FUEL PRESS	Left fuel pressure at inlet to high pressure fuel pump is less than 15 PSI or both fuel boost pumps on left side have been turned off with crossflow valve CLOSED.
R FUEL PRESS LOW	R FUEL PRESS	Right fuel pressure at inlet to high pressure fuel pump is less than 15 PSI or both fuel boost pumps on right side have been turned off with crossflow valve CLOSED.
L OIL PRESS LOW	L OIL PRESS	Left oil pressure below 16 PSI.
R OIL PRESS LOW	R OIL PRESS	Right oil pressure below 16 PSI.

Caution (Amber) Messages:

CAS Message:	Cause or Meaning:
L FUEL TEMP HOT	Left engine fuel temperature above 90°C or below 0°C.
R FUEL TEMP HOT	Right engine fuel temperature above 90°C or below 0°C.

Advisory (Blue) Messages:

CAS Message:	Cause or Meaning:
L OIL FILT BPASS	Left engine oil filter has clogged.
R OIL FILT BPASS	Right engine oil filter has clogged.

4. Limitations:

A. Flight Manual Limitations:

There are no limitations established for the Engine Indication system at the time of this revision.

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2A-78-10: General

1. **General Description:**

The purpose of the engine exhaust system is to continue the aerodynamic external surface of the engine and to collect hot and cold stream gas flows. The gases are then propelled through an acoustically lined exhaust unit designed to reduce noise.

A Thrust Reverser (TR) unit is incorporated in the exhaust unit to assist aircraft speed reduction on the ground. Two hydraulically actuated pivot doors, which are supplied with hydraulic pressure from the aircraft hydraulic systems, move into and block the combined fan and core airflow when the TR is deployed.

In the forward thrust configuration, the two pivot doors are closed and locked in position by four door locks. Two primary locks are mounted in the side beams. A secondary lock is integral with the hydraulic actuator and a tertiary lock is mounted on the rear face of the front structure torque box.

A manual restow system is incorporated in the TR design and is actuated by the flight crew by use of a manual stow switch. On airplanes Serial Number (SN) 1000 through 1435 hot having Part 2 of Aircraft Service Change (ASC) 418, the manual restow function energizes a solenoid in two dual solenoid TR selector control valves. The valves are then commanded to route stowing hydraulic pressure to both door actuators. On airplanes SN 1436 and subsequent and airplanes SN 1000 through 1435 having Part 2 of Aircraft Service Change (ASC) 418, the manual restow function de-energizes a solenoid in two single solenoid TR selector control valves. With the solenoids de-energized, stowing hydraulic pressure is routed to both door actuators.

2. **Thrust Reverser Lockout:**

If a TR system malfunction is suspected or if a failure prevents normal function of either or both TRs (including a no deploy condition), the airplane **cannot** be dispatched until the condition is corrected. Gulfstream recommends contacting Gulfstream Technical Operations, the Operator's Area Field Service Representative or the Operator's Home Base of Operations if a TR system malfunction is suspected or a failure is detected.

Thrust reverser lockout procedures for the Gulfstream IV airplane, although available to increase the level of dispatch capability, are a complex procedure. An improperly locked down TR may cause extensive damage to the TR system, spurious UNLOCK or DEPLOY warnings to be displayed to the flight crew or, in the worst case, actual inadvertent deployment of a TR door. Therefore, consideration should be given to allowing TR lockout procedures to be performed only by properly certificated Airframe and Powerplant Technicians in accordance with Chapter 78 of the latest approved version of the Gulfstream Aerospace GIV Aircraft Maintenance Manual.

With one reverser inoperative, the flight crew must be aware of possible asymmetric braking effects if the operative reverser is deployed on landing. If desired, lockdown of a properly operating TR can be accomplished to avoid possible asymmetric braking effects. However, Gulfstream does not recommend this in the event that the operational TR is needed for any reason during period the single malfunctioning TR is locked down.

The following references should be consulted if a TR system malfunction is

suspected or a failure is detected:

- GIV Airplane Flight Manual, Abnormal Procedures, Section 3-24-00, Thrust Reverser System
- GIV Airplane Flight Manual, Emergency Procedures, Section 4-24-00, Thrust Reverser System
- GIV Aircraft Maintenance Manual, Chapter 78, Engine Exhaust
- GIV Master Minimum Equipment List, System 78
- GIV Operating Manual, Limitations, Section 01-78-10, Reverse Thrust
- GIV Operating Manual, Abnormal/Emergency Procedures, Section 05-24-00, Thrust Reverser System

3. Subsystems, Units and Components:

For the purposes of this discussion, the engine exhaust system is divided into the following subsystems:

- 2A-78-20: Thrust Reverser System

2A-78-20: Thrust Reverser System

1. General:

(See Figure 4.)

A. Description:

The engine exhaust (aft cowl) system directs the exhaust gases from the engine turbine section aft and into the atmosphere. The Thrust Reverser (TR) exhaust unit for the Gulfstream IV is an airframe furnished component and is therefore not considered a part of the engine. It is intended for ground use only and provides a means of decelerating the aircraft during landing roll. When the reverser assembly is in the stowed position, it forms the engine exhaust nozzle and the aft most portion of the nacelle fairing.

B. Operation:

Between touchdown and 70 KCAS, reverse thrust may be used to shorten landing roll distance. The thrust reverser system is armed upon touchdown (with both nutcrackers in the ground mode or wheel speed sensors detecting acceleration) and the power lever for specific reverser is at IDLE. Only when the indication system on the instrument panel indicates REV ARM can reverse thrust be utilized. Reverse thrust is obtained by pulling up and aft on the reverse thrust levers. When the reverser doors unlock, the REV UNLOCK light will illuminate and when the doors are fully deployed, REV DPLY light will illuminate. Increased reverse thrust is obtained by further aft movement of reverse levers.

(1) Arming System:

When the power lever mounted in the cockpit control pedestal is in the idle position (left power lever electrical switch also in idle position), the reverse thrust lever can be moved in its upward / aft direction for deployment.

Providing that the following conditions are met, 28V DC essential bus power is transmitted via the L TR CONT circuit breaker to the left TR arm indicator, resulting in the REV ARM light on the pilot’s instrument panel to illuminate:

- Left power switch in the idle position
- Left fire handle in the normal position
- T/REV EMER STOW switch not depressed (and on airplanes SN 1000 - 1143 excluding SN 1034 not having Aircraft Service Change (ASC) 166 Nutcracker No. 3 relay energized [on ground configuration]) or on airplanes SN 1000 - 1143 excluding 1034 having ASC 166 and airplanes SN 1034, 1144 and subsequent
- Either nutcracker No. 3 relay energized (on ground configuration) or TR wheel speed relay energized (wheel speed above 65 MPH)

(2) Deployment:

When reverse thrust lever is moved in its upward / aft direction, a sector wheel and push-pull rod in the cockpit pedestal is caused to move forward to activate the left TR lever switch into the deploy position. Power from the 28V DC essential bus is also routed to energize the left secondary lock relay. When this relay is energized, a circuit is completed via the L SEC LOCK circuit breaker and the closed contacts of the left secondary lock relay to energize the secondary lock actuator. With the secondary lock actuator energized, the circuit (coming through L T/REV CONT circuit breaker) is also completed to energize solenoid No. 1 of the left TR selector control valve. This routes hydraulic pressure to the deploy side of the TR actuator, resulting in the unlocking of the TR actuator primary lock and deploying the doors.

NOTE:

On airplanes SN 1000 - 1143 excluding SN 1034 having ASC 166 and airplanes SN 1034, 1144 and subsequent, the left secondary lock time delay relay is energized after 5 seconds. Power is no longer routed to left secondary lock relay and to secondary lock actuator, but is routed directly to solenoid No. 1 of the left TR selector control valve to maintain hydraulic pressure to the deploy side of the actuator.

- REV UNLK indication:

  The left thrust reverser REV UNLK light will illuminate when the secondary lock actuator retracts (unlocks), either the upper or lower door limit switch is activated, or the primary lock switch (part of TR actuator) is closed (unlocked position). When any or all of these conditions are present, power is transmitted from the warning lights power system to energize the left TR interlock relay. When activated, this relay completes a circuit through its contacts to the left TR unlock indicator, causing the REV UNLK light to illuminate. In addition, when any of the conditions for the REV UNLK light illumination occurs, the Engine Instrument and Crew Advisory System (EICAS) and standby warning light indications are also activated if either Nutcracker No. 1 relay is not energized

(air configuration) or power lever No. 2 T/O relay is energized.

- REV DPLY indication:

  The left thrust reverser REV DPLY light on the pilot's instrument panel will illuminate when TR doors are fully deployed. In this condition, power is transmitted from the warning lights power system through closed contacts of the deploy limit switch to the left TR deploy indicator which causes the REV DPLY light to illuminate.

(3) Stowing:

When reverse thrust lever in cockpit is moved to the stow position, the left TR lever switch opens. This de-energizes the left secondary lock relay, thereby opening circuits to de-energize solenoid No. 1 of the left TR selector control valve and to shut off hydraulic pressure to the deploy side of TR hydraulic actuator. At the same time, circuits are closed to energize solenoid No. 2 of the left TR selector control valve to allow hydraulic pressure to the stow side of TR hydraulic actuator. The REV DPLY light will extinguish when doors begin to transition towards stow position. When TR doors are stowed and locked, the REV UNLK light will extinguish also.

(4) Emergency Stowing:

(See Figure 1.)

If an in-flight failure causes a REV UNLK light and REV UNLOCKED message to be displayed on the EICAS, the T/REV EMER STOW switch, located on the center pedestal, can be depressed. Although the results of depressing the T/REV EMER STOW switch are the same, the sequence of events varies with airplane effectivity as follows:

Airplanes SN 1000 through 1435 not having Part 2 of ASC 418: Depressing the T/REV EMER STOW switch completes a circuit from the essential 28V DC bus via the T/R EMER STOW circuit breaker (airplanes SN 1000 - 1182 not having ASC 210) or L T/R EMER STOW and R T/R EMER STOW circuit breakers (airplanes SN 1000 - 1182 having ASC 210 and SN 1183 and subsequent) to energize the stow solenoid of both TR selector control valves. This in turn allows hydraulic pressure to the stow side of both TR hydraulic actuators.

Airplanes SN 1436 and subsequent and airplanes SN 1000 through 1435 having Part 2 of ASC 418: These airplanes have the existing dual solenoid TR selector control valves replaced with single solenoid TR selector control valves. The single solenoid TR selector control valve employs a new spool and sleeve assembly that is spring biased and hydraulically powered to the stow position. The new valve spool and sleeve assembly is automatically piloted to the stow position when hydraulic power is available, thus eliminating the need for the stow solenoid. Depressing the T/REV EMER STOW switch removes all electrical power to both TR selector control valves. Spring pressure and hydraulic pressure then move the spool to the stow position, allowing hydraulic pressure to the stow side of both TR hydraulic actuators. The T/R EMER STOW circuit breaker

(airplanes SN 1000 - 1182 not having ASC 210) or L T/R EMER STOW and R T/R EMER STOW circuit breakers (airplanes SN 1000 - 1182 having ASC 210 and SN 1183 and subsequent) are removed on these airplanes.

C. Subsystems, Units and Components:

The thrust reverser system is composed of the following subsystems, units and components:

(1) Safety Devices, Consisting of the Following:

- Primary lock
- Secondary lock
- Mechanical feedback system

(2) Major Components, Consisting of the Following:

- Selector control valve
- Hydraulic actuator
- Secondary lock actuator

2. Description of Subsystems, Units and Components:

A. Thrust Reverser System Safety Devices:

During normal TR system operation, the following safety devices are utilized to protect against inadvertent operation of a TR, a jammed condition and/or improper operation of the system:

(1) Primary Lock:

(See Figure 2.)

The hydraulic actuator includes a built-in locking device which is engaged by a spring force when the actuator is fully extended (TR stowed position). This lock disengages hydraulically when deploy pressure is applied to the actuator.

(2) Secondary Lock:

(See Figure 2.)

A secondary latch located in the outboard stang beam locks both doors in the stowed position. It is spring loaded to the locked position and is unlocked by a solenoid actuator which is energized via a deploy command. On airplanes having ASC 18 (CAA requirements), airplanes SN 1000 - 1143 excluding SN 1034 having ASC 166 and SN 1034, 1144 and subsequent, the solenoid actuator is de-energized when doors deploy with a 5 second time delay relay.

(3) Mechanical Feedback System:

A mechanical feedback system automatically retards the engine power lever from a forward position to a near idle position should a reverser inadvertently deploy. This system also serves as a throttle interlock in that it restricts the reverse thrust lever from moving into reverse power range until the reverser moves from fully stowed to fully deployed position. In addition, the feedback limits reverse power and forward power during reverser transients. This feature protects the system in the event of a jammed reverser during landing or improper reverser operation during an aborted landing (after reverser has been deployed).

B. Thrust Reverser System Major Components:

(1) Selector Control Valve:

(See Figure 3.)

(a) All GIV airplanes:

Two TR selector control valves, located in the tail compartment at FS 682, port hydraulic fluid to the hydraulic actuator for deploying or stowing the TRs. One valve controls left TR door operation and the other valve controls right TR door operation.

(b) Airplanes SN 1000 through 1435 not having Part 2 of ASC 418:

Each TR selector control valve is a four-way, three-position unit that uses two solenoids to deploy and stow the TR. When the deploy solenoid is energized, flow from the pressure port is routed to the deploy port and the stow port is routed to hydraulic return. When the stow solenoid is energized, flow from the pressure port is routed to the stow port and the deploy port is routed to hydraulic return. In the neutral position (both solenoids de-energized), both ports are routed to hydraulic return.

The energized solenoid mechanically controls system pressure to the appropriate end of the spool which then actuates internal routing of the flow-through valve. The center position of spool in sleeve is maintained through opposing springs which position stops at this point. The respective stop-spring is overcome to allow the spool to move to an operating position as commanded.

(c) Airplanes SN 1436 and subsequent and airplanes SN 1000 through 1435 having Part 2 of ASC 418:

Each TR selector control valve is a four-way, three-position unit that uses a single solenoid to deploy and stow the TR. The solenoid is spring biased and hydraulically powered to the stow position. When the solenoid is energized, flow from the pressure port is routed to the deploy port and the stow port is routed to hydraulic return. When the solenoid is de-energized, flow from the pressure port is routed to the stow port and the deploy port is routed to hydraulic return.

The single solenoid TR selector control valve employs a new spool and sleeve assembly. The new spool and sleeve assembly is automatically piloted to the stow position when hydraulic power is available, thus eliminating the need for the stow solenoid.

(2) Hydraulic Actuator:

(See Figure 2.)

The TR hydraulic actuator is used to operate the TR doors. Each actuator is mounted on the outboard side of each engine between the bellcrank and outboard stang beam.

The actuator operates on the aircraft hydraulic system using an

operating pressure of 3000 PSI and a minimum flow of 6 GPM. It uses a mechanical lock that consists of four latches driven by a spring-loaded lock plunger. The normal actuator position is with piston rod extended, TR stowed and mechanical lock engaged.

In the stowed position, the four latches (each riding on a ball bearing) are seated against the lock plunger which retains them in the extended position. When extended, the latches are seated against the end of the piston chamber. Deploy hydraulic pressure (less than 1000 psi) will cause the lock plunger to act against a retaining spring. When the lock plunger moves away from the piston, it allows the latches to retract into the piston. As the piston retracts, residual hydraulic pressure in the STOW side of the actuator is ported to return through the STOW port.

Hydraulic pressure applied through the STOW port forces the piston to the extend position. The residual hydraulic pressure in the deploy side of the actuator is ported to return through the DEPLOY port. At the end of its stroke, the piston bottoms against the plate at the cylinder's end fitting. At the same time, the lock plunger (which is held against the other side of the plate by retaining spring wedges between the latches and piston rod surface) forces the latches to extend from the piston into the locked position. The end circumference of the plunger is sloped, enabling it to wedge between the latches, ball bearings and piston rod surface.

The lock indicator switch is operated by a hydraulic plunger in the switch housing. When the actuator or piston is extended and locked, the plunger is restrained within the housing by one of the latches. A ball bearing between the side of the hydraulic plunger and the spring-loaded switch plunger keeps the switch plunger depressed. When the actuator piston is unlocked, hydraulic pressure moves a hydraulic plunger inward towards the cylinder. The ball bearing seats inside a groove in the plunger, which allows the switch plunger to extend. Extension or retraction of the switch plunger determines which set of contacts in the switch is open or closed.

(3) Secondary Lock Actuator:

(See Figure 2.)

The secondary lock actuator is solenoid operated to provide electro-mechanical control of the secondary lock for the TR. The actuator is mounted at the outboard stang beam.

The solenoid is spring-loaded to the extended position and retracts when 28V DC power is applied to a solenoid coil. A bank of three microswitches is contained within the actuator. Switch A indicates shaft position (unlock), switch B controls current to the TR selector control valve and switch C is not used but can be used to replace switch B (if necessary).

3. Controls and Indications:

A. Circuit Breakers (CBs):

The thrust reverser system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
L SEC LOCK	P	G-7	ESS DC Bus
R SEC LOCK	P	H-7	ESS DC Bus
L T/REV CONTROL	P	I-7	ESS DC Bus
R T/REV CONTROL	P	J-7	ESS DC Bus
T/R EMER STOW **(1)**	P	K-7	ESS DC Bus
L T/R EMER STOW **(2)**	P	L-7	ESS DC Bus
R T/R EMER STOW **(2)**	P	K-7	ESS DC Bus

NOTE(S):

(1) Airplanes SN 1000 - 1182 not having ASC 210 or Part 2 of ASC 418.

(2) Airplanes SN 1000 through 1182 having ASC 210 but not having Part 2 of ASC 418; SN 1183 through 1435 not having Part 2 of ASC 418.

B. Crew Alerting System (CAS) Messages:

Warning (Red) Messages and Annunciations:

CAS Message:	SWLP Indication:	Cause or Meaning:
L REV UNLOCK	L REV UNLOCKED	Left thrust reverser has become unlocked.
R REV UNLOCK	R REV UNLOCKED	Right thrust reverser has become unlocked.

Annunciation:	Cause or Meaning:
White REV UNLOCK light above DU 2 or 5.	Thrust reverser unlocked.
White REV DPLY light above DU 2 or 5.	Thrust reverser deployed.

Advisory (Blue) Annunciations:

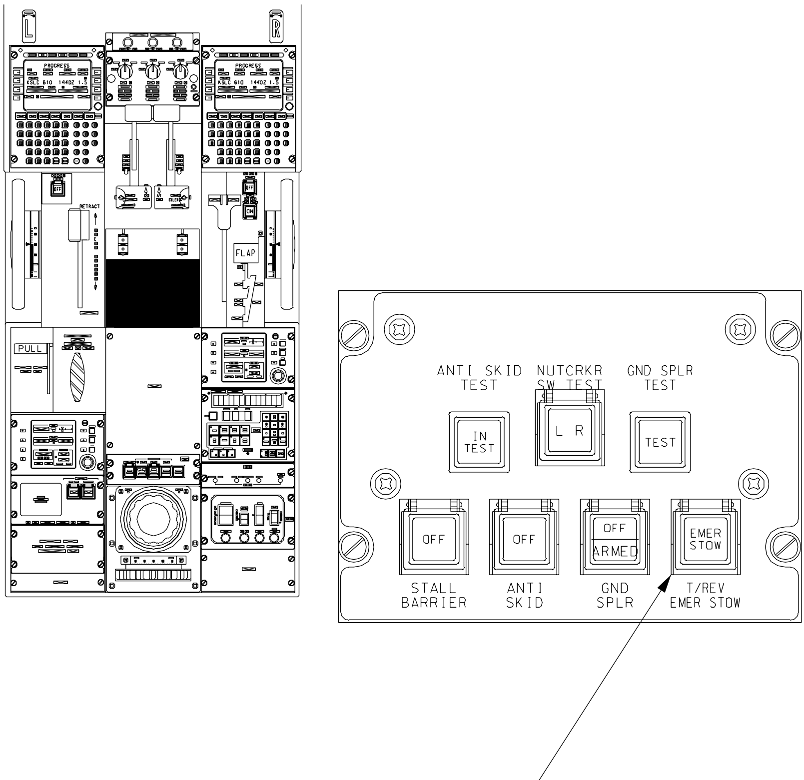
Annunciation:	Cause or Meaning:
REV ARM light (green) illuminated above DU 2 or 5.	Parameters satisfied for thrust reverser deployment.

4. Limitations:

A. Flight Manual Limitations:

Cancellation of reverse thrust should be initiated by 70 KCAS so as to be at reverse idle by normal taxi speed. Thrust reverser door extension at taxi speeds is permitted if ASC 166 is incorporated and thrust reversers are kept low.

For airplanes SN 1000 through 1143 without ASC 166, use of thrust reversers is limited to one (1) minute every thirty (30) minutes.

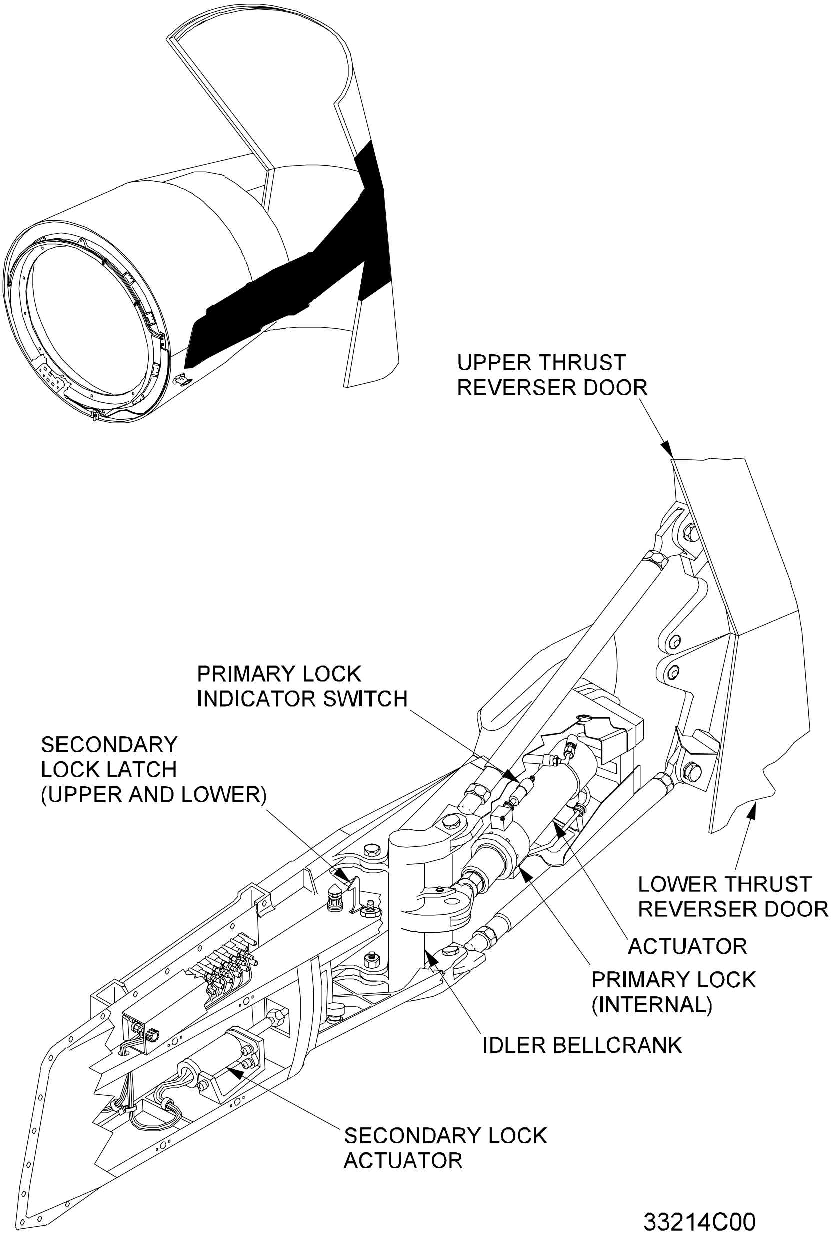


T/REV EMER STOW

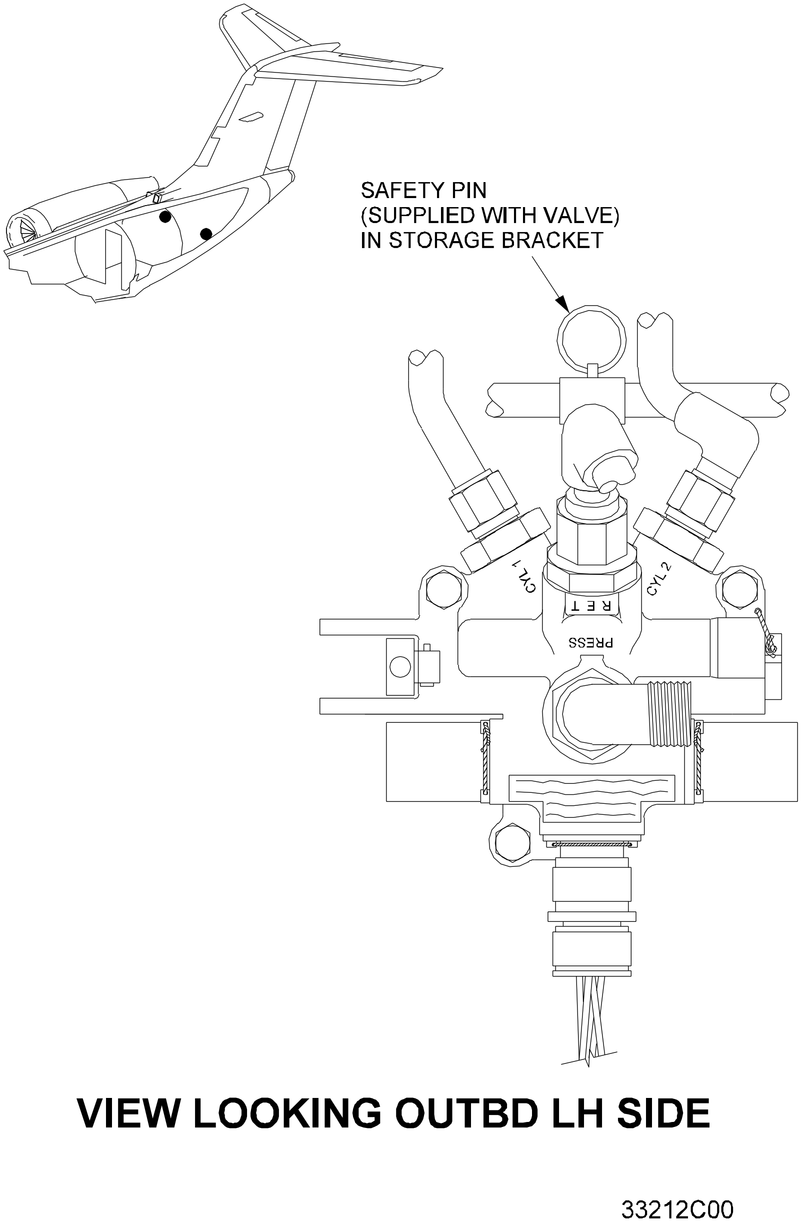
- Depressed if an inflight failure of either or both thrust reversers is detected (indicated by a white REV UNLK light and a red REV UNLOCKED message displayed on the EICAS)
- Directs hydraulic pressure to the stow side of the actuator only
- Disables all other thrust reverser controls

33428C00

Thrust Reverser Emergency Stow Switch
Figure 1



Thrust Reverser Components
Figure 2



Thrust Reverser Selector Control Valve
Figure 3

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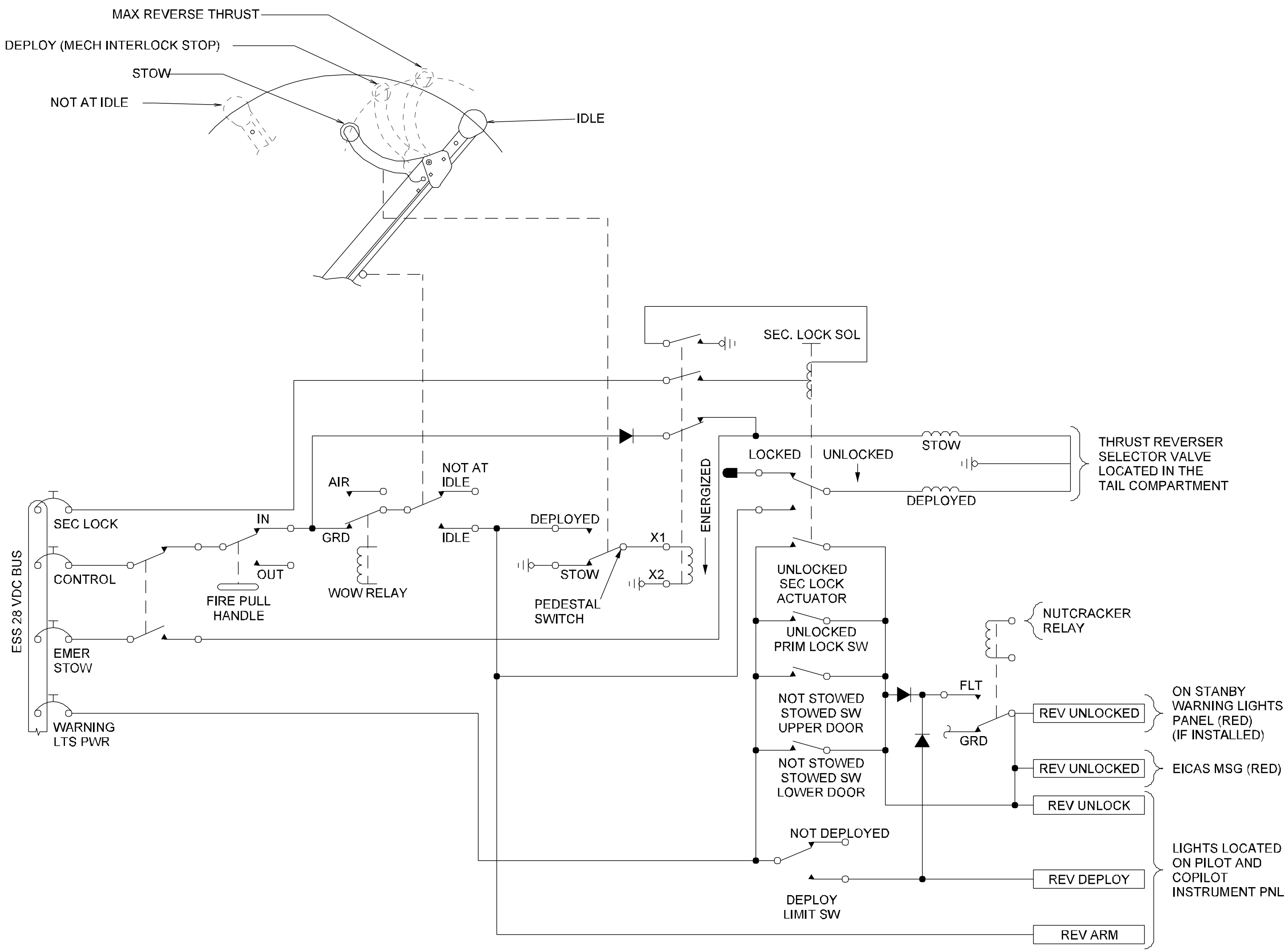
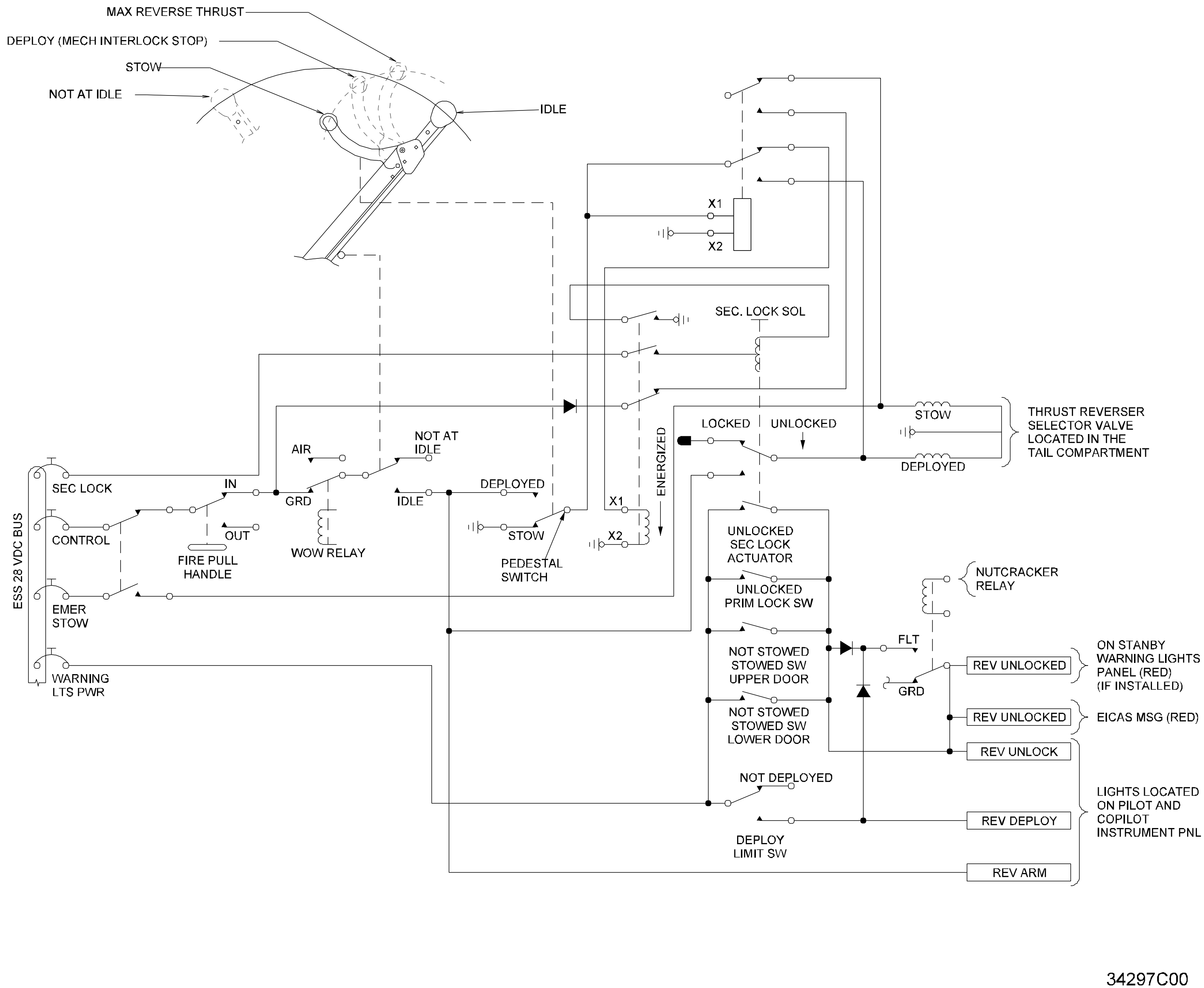


Thrust Reverser Simplified Schematic (Airplanes SN 1000 - 1143 Excluding SN 1034 Not Having ASC 166 / 166A)
Figure 4



Thrust Reverser Simplified Schematic (Airplanes Having ASC 18, Airplanes SN 1000 - 1143 Excluding SN 1034 Having ASC 166 / 166A and Airplanes SN 1034, 1144 and Subsequent)
Figure 5

2A-79-10: Engine Oil System

1. **General:**

   A. **Description:**

   The oil system is a self-contained, full flow re-circulatory type that gives lubricating and cooling oil flow to the engine bearings, gears and splines throughout the starting, stopping and operational sequence.

   B. **Operation:**

   Oil is drawn from the oil tank by a pressure pump. The pump feeds the engine via an oil cooler and a pressure oil filter, each of which contain a bypass valve. A relief valve incorporated in the pressure filter ensures that oil pressure does not exceed a pre-determined value. An oil filter differential pressure indicator assembly (mounted at the rear of the oil tank) indicates a blockage in the oil filter.

   Oil is returned to the tank by five scavenge pumps and discharged on to a de-aerator tray. The oil tank vents into the high-speed gearbox which exhausts through a centrifugal breather.

   Feed oil filtration is given by a pressure oil filter, thread and gauze filters. Return oil filtration is given by gauze strainers and scavenge filters.

   C. **Subsystems, Units and Components:**

   - Engine Oil Tank
   - Oil Pressure Transmitter
   - Low Oil Pressure Switch
   - Oil Pump Assembly
   - Oil Filter Assembly
   - Fuel Cooled Oil Cooler (FCOC)
   - Engine Oil Remote Replenishing System

2. **Description of Subsystems, Units and Components:**

   A. **Engine Oil Tank:**

   (See Figure 2.)

   The oil supply for engine lubrication is in an oil tank mounted below the oil cooler on the accessory suspension bracket.

   The capacity of each engine oil tank is as follows:

   - Left Engine: 6.87 Liters
   - Right Engine: 7.3 Liters

   A calibrated sight glass, installed on the left-hand wall, shows the amount of oil necessary to fill the tank to its correct level. The FULL mark shows the oil capacity of the tank. The sight glass has two graduated scales, one for the left-hand engine (colored red) and one for the right-hand engine (colored green).

   The oil supply to the pressure pump is drawn from the tank by suction induced through the rotation of the pump gears. The oil flows from the bottom of the tank through the supply tube connector on the right-hand wall

of the tank. The strainer in the supply connector prevents any foreign matter contamination in the oil system.

Return oil from the engine main bearings and internal gearbox scavenge pumps goes into the tank by way of a common tube from the oil pump. Return oil from the high-speed gearbox scavenge pump goes into the tank by a separate tube. Both of these tubes discharge the scavenge oil, as an oil/air mixture, onto the de-aerator tray. The oil flows along the tray which allows the entrapped air to vent through a tube into the high-speed gearbox.

B. Oil Pressure Transmitter:

(See Figure 2.)

The oil filter outlet pressure is sensed by the oil pressure transmitter. The oil pressure transmitter converts oil pressure into an electrical signal and sends it to a dual-signal conditioner. From there, signal outputs are routed to the EICAS through their respective DAU. Power for the signal conditioner is derived from the Essential No. 1, 26V AC transformer bus via the L ENG OIL PRESS circuit breaker and the Essential No. 2, 26V AC transformer bus via the R ENG OIL PRESS circuit breaker.

(1) The engine oil pressure parameters (on ground and during takeoff) are as follows:

- On Ground = Weight On Wheels (WOW)
- Takeoff = Up To Four Minute Period Following WOW Transition

HP RPM Ranges (%)	Oil Pressure (PSIG)	
	White Indication	Red Indication
< 76.0	> 16	< 15
78.4 ≥ HP > 76.0	> 17	< 16
80.0 ≥ HP > 78.4	> 18	< 17
> 80	> 30	< 29

(2) The engine oil pressure parameters (air mode) are as follows:

- Air Operation = Greater Than Four Minutes Following WOW Transition

HP RPM Ranges (%)	Oil Pressure (PSIG)	
	White Indication	Red Indication
< 76.0	> 16	< 15
78.4 ≥ HP > 76.0	> 17	< 16
80.8 ≥ HP > 78.4	> 18	< 17
83.2 ≥ HP > 80.8	> 19	< 18
85.6 ≥ HP > 83.2	> 20	< 19
88.0 ≥ HP > 85.6	> 21	< 20
90.4 ≥ HP > 88.0	> 22	< 21
92.8 ≥ HP > 90.4	> 23	< 22
95.2 ≥ HP > 92.8	> 24	< 23
> 95.2	> 25	< 24

C. Low Oil Pressure Switch:

(See Figure 2.)

The low oil pressure switch is a 15 ±0.4 PSI switch that activates the low oil pressure warning system if the pressure falls below its set parameter.

When the oil pressure is less than the lowest acceptable limit, the contacts stay closed. This causes the low pressure warning indicator in the crew compartment to illuminate. Oil fills the diaphragm chamber through holes in the circular base plate of the switch assembly. As the pressure increases, the diaphragm causes a pushrod to move to the moving contact. When the pressure is more than the setting limit the contacts open and the warning indicator in the crew compartment extinguishes.

The electrical contacts attach to the bottom of the terminal block. A setting adjustment assembly is attached to the terminal block molding. This assembly permits the contacts to operate at any specified oil pressure within the setting range.

The top cover assembly contains a flame trap. This flame trap will stop the spread of fire if the contacts arc when oil vapor is in the switch.

D. Oil Pump Assembly:

(See Figure 4.)

The oil pump unit, driven by the right gearbox, consists of six internal oil pumps, consisting of one oil pressure pump and five oil scavenge pumps. These internal pumps affect the following components:

- LP Turbine Bearing
- Internal Gearbox
- Thrust Bearings
- HP Turbine Bearing
- Right Gearbox

(1) Oil Pressure Pump:

The oil pressure pump supplies pressurized engine oil to the components (listed above). The pump is a gear-driven positive-displacement pump and incorporates a 200 PSI pump pressure relief valve. If the pump output exceeds 200 PSI, the excess pressure is returned back to the pump inlet.

(2) Oil Scavenge Pumps:

The oil scavenge pumps return engine oil from the engine components (listed above) back to the oil tank. The return oil output from four of the pumps (all except the right gearbox return oil) is directed into a common oil return tube to the tank.

The right scavenge pump return oil is pumped through a separate tube to prevent oil from siphoning from the oil tank to the right gearbox when the engine is not operating.

E. Oil Filter Assembly:

(See Figure 2.)

The oil filter assembly mounts on the front wall of the engine oil tank. The oil flows from the oil cooler into the filter housing, flowing from the outside to the inside of the element. If the pressure difference across the element is

more than 50 PSI, the bypass valve opens to let the oil bypass the element. The system relief valve is subjected to the pressure downstream of the element. If this pressure is more than 50 PSI, the valve opens to allow the oil flow back into the tank below the de-aerator tray.

An oil filter differential pressure indicator assembly (mounted on the rear wall of the oil tank) will indicate if there is a blockage in the pressure oil filter. When the differential pressure across the filter is more than 18 ±3.0 PSID, the indicator button compression spring forces a red button to extend (approximately 0.187 inch) from the body of the indicator. When the differential pressure exceeds 30 ±4.5 PSID, the differential pressure switch closes, which illuminates a blue L-R OIL FILT BPASS message on the EICAS.

After passing through the element, the oil is directed through two tube connectors. One tube conveys oil to a two-way connector on the bypass duct, from where the oil passes to the turbine bearings and central feed oil tube. The other tube moves oil to the high-speed gearbox.

F. Fuel Cooled Oil Cooler:

(See Figure 3.)

Oil from the pressure pump passes into the rear of the oil cooler. It flows around the outside of the tubes and is directed by the baffle plates to pass numerous times through the matrix assembly. On reaching the forward end of the cooler, the oil passes to the pressure oil filter for distribution throughout the system. The fuel, which flows through the tubes, is of a moderately low temperature and extracts a proportion of the heat from the oil.

At low temperatures, the viscosity of the oil is high, restricting its flow around the tubes. When the pressure drop reaches 50 PSI, the bypass valve opens to allow oil to pass directly to the outlet connection. This ensures adequate oil supply for engine lubrication.

G. Engine Oil Remote Replenishing System:

(See Figure 1.)

The engine oil can be replenished from a self-contained engine oil reservoir located in the aft equipment bay on the left side. The system consists of a 16 pint reservoir, oil pump, left and right solenoid valves and associated plumbing.

The replenishing pump receives electrical power from the 28V DC essential bus through the ENGINE OILER circuit breaker. The pump is also controlled by the tail compartment door switch, an engine oiler switch located near the reservoir and an engine oiler switch located on each engine. The switches on each engine are spring-loaded to OFF. When selected ON, each switch activates the replenishing pump and opens the solenoid valve for its respective engine.

3. Controls and Indications:

NOTE:

A detailed description of the Engine Instruments and Crew Alerting System (EICAS) can be found in Section 5 of Honeywell's SPZ-8000 (or SPZ-8400) Digital Automatic Flight Control System Pilot's Manual for the Gulfstream IV.

A. Circuit Breakers (CBs):

The engine oil system is protected by the following circuit breakers:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
ENGINE OILER	P	L-9 or G-10 **(1)**	ESS DC Bus
L ENG OIL PRESS	CP	A-12	ESS AC Bus
R ENG OIL PRESS	CP	B-12	ESS AC Bus

NOTE(S):

(1) Depending on effectivity.

B. Crew Alerting System (CAS) Messages:

Warning (Red) Messages:

CAS Message:	SWLP Indication:	Cause or Meaning:
L-R OIL PRESS LOW	L-R OIL PRESS	Indicated oil pressure below 16 PSI

Advisory (Blue) Messages:

CAS Message:	Cause or Meaning
L-R OIL FILT BPASS	Engine oil filter has clogged

4. Limitations:

A. Flight Manual Limitations:

(1) Oil Inlet Temperature:

- Maximum: 105°C (Oil temperature up to 120°C for maximum of fifteen (15) minutes is permissible)
- Minimum for Starting: -40°C
- Minimum for opening power lever: -30°C

NOTE:

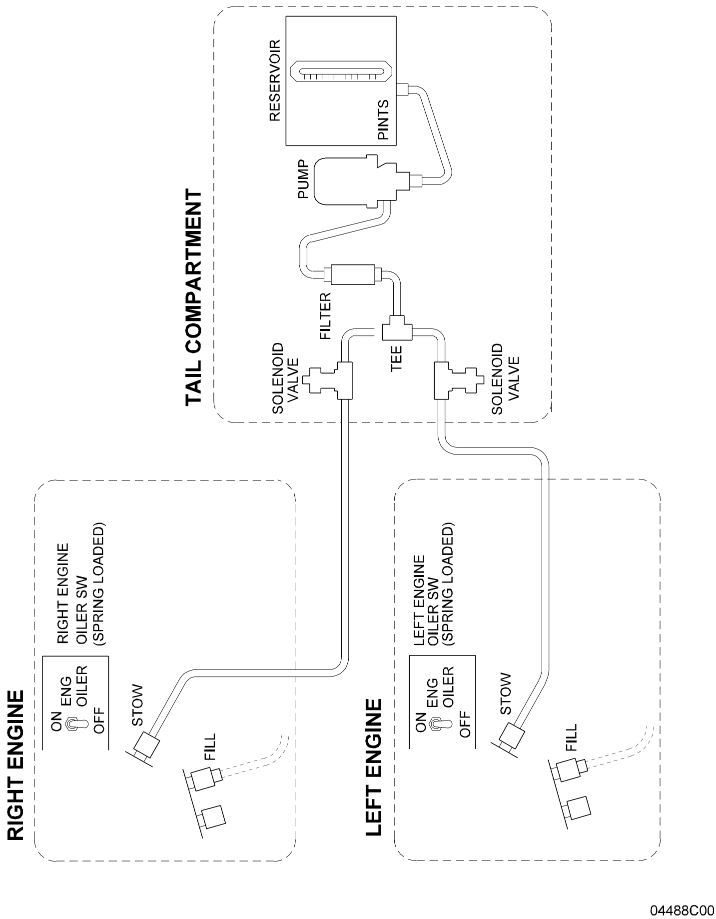
External heating will be required to raise oil temperature to -40°C for cold weather starting. If oil temperature is less than -30°C, the engine should be idled until at least -30°C temperature is reached.

(2) Oil Pressure:

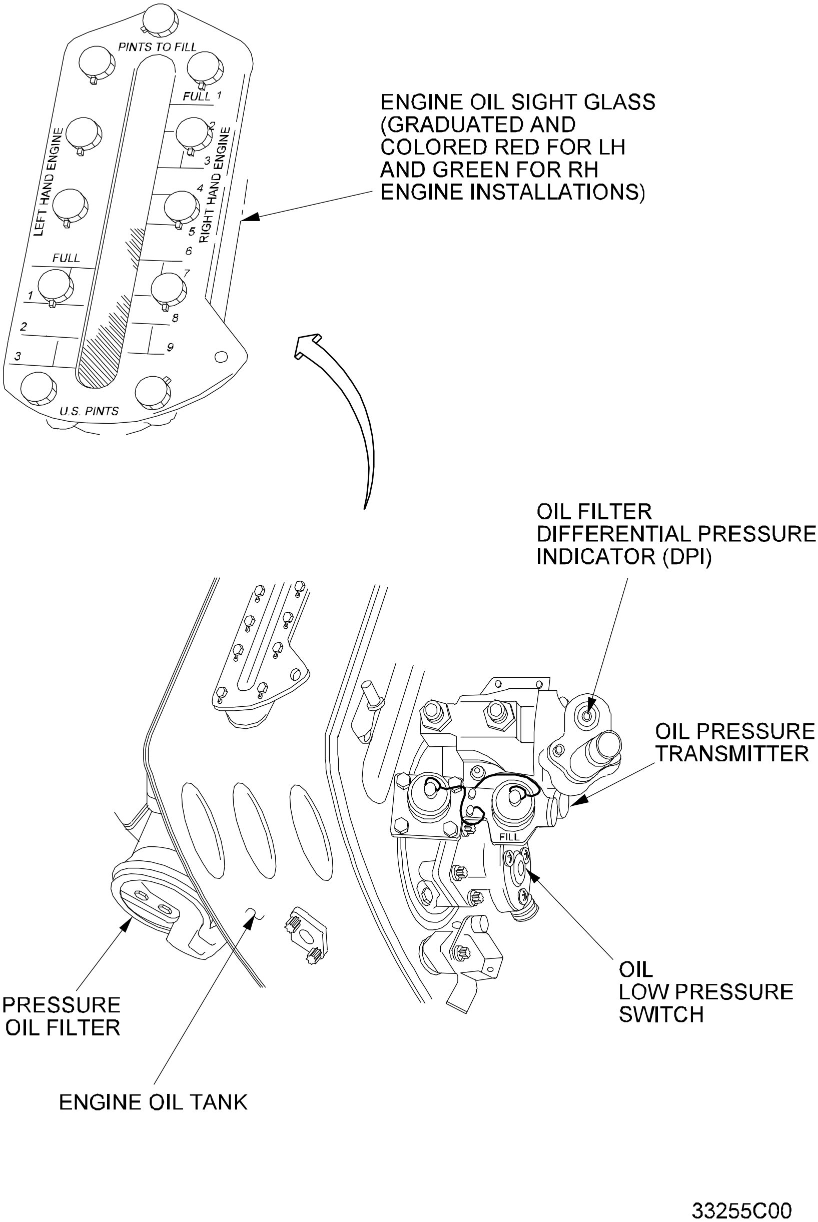
- Minimum at Idle: 16 PSI
- Minimum acceptable for takeoff (at takeoff power): 30 PSI

Minimum to complete flight:

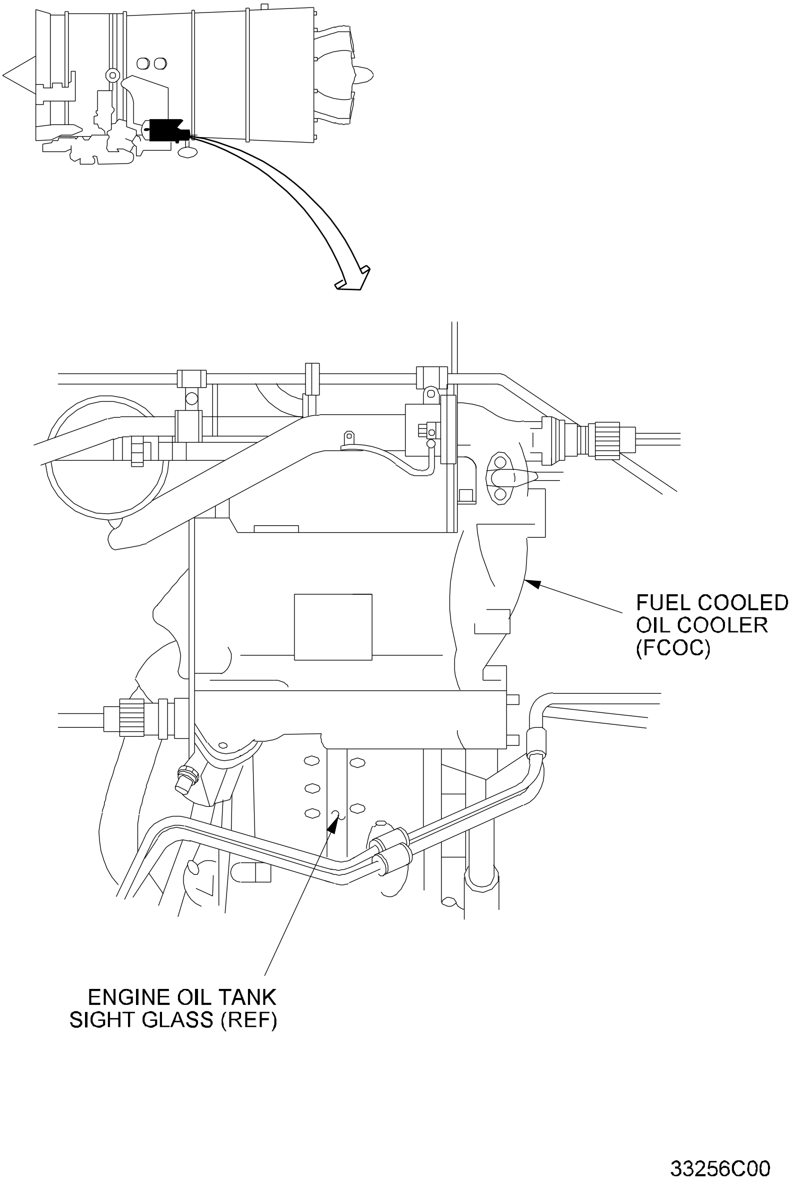
- At maximum continuous (97.5% HP): 25 PSI
- 92% HP RPM: 22 PSI
- 84% HP RPM: 19 PSI
- 76% HP RPM or lower: 16 PSI



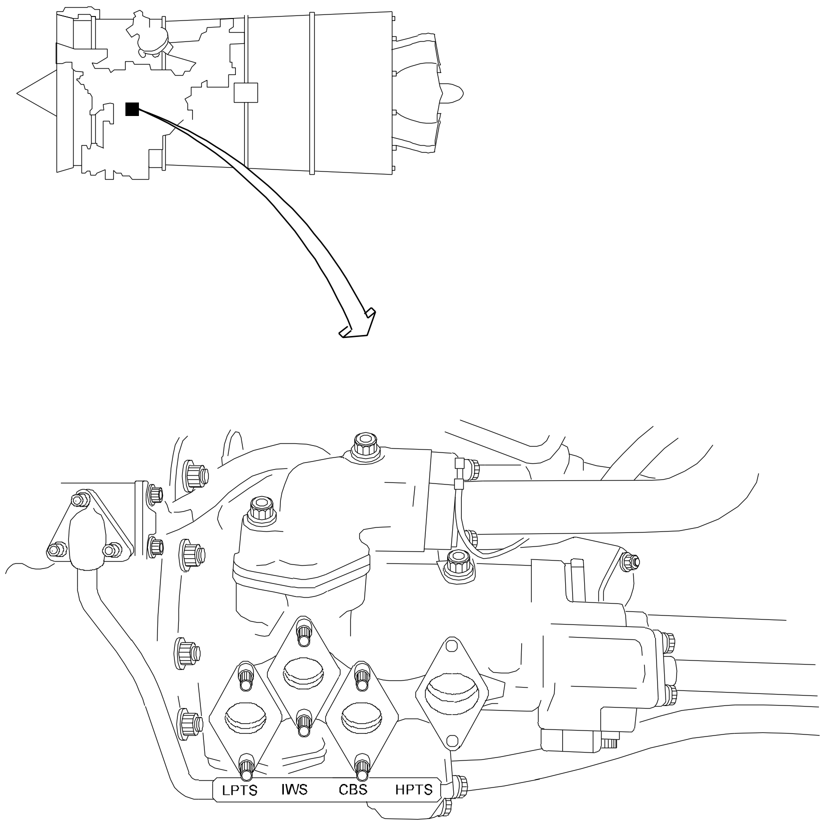
Engine Oil Remote Replenishing System Simplified Diagram
Figure 1



Engine Oil Tank and Associated Components
Figure 2



Fuel Cooled Oil Cooler
Figure 3



Oil Pump Assembly
Figure 4

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2A-80-10: Engine Starting System

1. **General:**

A. **Description:**

The engine starting system can receive pneumatic starting air from any of the following sources:

- Auxiliary Power Unit (APU)
- An already running engine (crossbleed)
- An external air source (i.e., engine start cart)

Engine starting can be accomplished on the ground (ground starts) or in the air (air starts), with or without the starter motor. The air used for starting is ducted by means of the bleed air manifold from any of the three sources listed above. The cranking operation is accomplished by a pneumatic starter mounted on the right hand (HP) gearbox. The motion of the starter (rotation) engages a series of gears and a drive shaft. The drive shaft is connected to the HP spool of the engine, causing it to turn. The starter rotates the engine up to approximately 15% HP RPM, at which point fuel is introduced with ignition from the ignitors within the two liners. The starter now assists engine spool-up to approximately 38 to 45% HP RPM.

Each engine ignition system incorporates two transistorized high energy ignition units. They are operated automatically when the engine is being started to deliver high voltage and amperage to the igniter plugs. Each igniter plug is served by a high-energy ignition unit. The plugs, when fired, ignite the fuel/air mixture and the resultant flame passes through interconnectors located between each liner to complete the flame propagation. Cranking the engine over without ignition can be accomplished by selective switching by the crew. Ignition may also be used independently of the engine starting system if a relight in flight is required.

NOTE:

For additional information pertaining to the engine ignition system, see Section 2A-74-10, Engine Ignition System.

Electrical power for ground starting and ignition power is obtained from the Essential 28V DC bus. Ignition indication is displayed through the Engine Instrument and Crew Advisory System (EICAS) located on the center console.

B. **Operation:**

The engine starting system provides the flight crew with a means of using supplied pneumatic power (starting air) for the following tasks:

(1) Normal Engine Starts (Ground): See Section 03-03-20, Starting Engines.

(2) Alternate Engine Starts (Ground):

The five types of alternate engine starts available on the ground are as follows:

- Cold start: see 03-08-10, Cold Start
- External air start: see 03-08-20, External Air Start
- Crossbleed start: see 03-08-30, Crossbleed Start
- Engine cranking cycle: see 03-08-40, Engine Cranking Cycle
- Engine start (battery power only): see 03-08-50, Engine Start - Battery Power Only

(3) Engine Airstarts:

The two types of engine airstarts are as follows:

- Immediate: see 05-10-20, Airstart - Immediate
- Normal: see 05-10-30, Airstart - Normal

(4) Starter-Assisted Air Starts:

Engine starting mode inflight within the starter-assist region of the airstart envelope in which the flight crew controls the start sequence.

(5) Wet And Dry Cranks:

Engine operating mode in which the engine is motored but without igniters ON. Wet cranking allows fuel to the engine whereas dry cranking does not. See Section 03-08-40: Engine Cranking Cycle, for engine cranking procedures.

C. Subsystems, Units and Components:

The engine starting system is composed of the following subsystems, units and components:

- Pneumatic Starter
- Shuttoff Differential Pressure Regulator

NOTE:

The pneumatic starter and the shutoff differential pressure indicator are combined together to form the starter motor assembly.

2. Description of Subsystems, Units and Components:

A. Pneumatic Starter:

The pneumatic starter is electrically controlled and pneumatically operated. It consists of a single-stage turbine rotor, a 16:1 reduction gear, output shaft assembly exhaust housing and gear housing assembly.

Starter motor operation is controlled by the following four switches located on the ENGINE START panel on the cockpit overhead panel (see Figure 1 and Figure 2):

- MASTER (two switches, CRANK and START, both with ON / OFF positions)
- START (L ENG and R ENG)

B. Shuttoff Differential Pressure Regulator:

The shutoff differential pressure regulator consists of a pneumatic actuator, a filter, a switcher solenoid valve assembly, a metering valve assembly, servo assembly, overpressure limiter, butterfly valve, valve body assembly and actuator assembly. The regulator functions both as an air pressure

regulator and as an air flow shutoff valve.

Control air enters the valve body assembly through the upstream pressure sensing tube. The air passes through the filter to the metering valve assembly and to the switcher solenoid valve assembly. With the solenoid de-energized, control air passes through the switcher valve and is transmitted to the pneumatic actuator closing chamber. The force of the control air pressure acting on the pneumatic actuator piston supplements the force of the piston spring to maintain the butterfly valve to the closed position, preventing the flow of air.

As the inlet air pressure rises above the predetermined value, the air pressure acting on the piston of the metering valve assembly overcomes the force of the spring, closes the valve, and regulates the pressure to the switcher solenoid valve assembly.

When the solenoid is energized, control air passes through the switcher solenoid valve assembly and is transmitted to the pneumatic actuator opening chamber at a rate determined by the pneumatic actuator chamber orifice. The pneumatic actuator closing chamber is vented to the atmosphere through the switcher solenoid valve assembly. As the butterfly valve opens, downstream air pressure increases and air flows through the servo assembly to the metering valve assembly, resetting the metering valve assembly to a desired value for the servo assembly supply pressure.

With air flowing through the unit, the servo assembly acts to regulate the downstream air pressure at a constant, predetermined value. Downstream air pressure is sensed in the actuator housing assembly downstream of the butterfly valve. As downstream air pressure increases, pressure acting on the servo piston overcomes the force of the piston spring and the piston lifts the ball valve from the seat, allowing air to bleed from the pneumatic actuator opening chamber. As control air pressure in the pneumatic actuator opening chamber increases, the pneumatic actuator piston spring acts to move the butterfly valve toward the closed position, reducing the flow of air through the regulator. When downstream pressure decreases, the servo assembly piston spring moves the piston and seats the ball valve, stopping the bleed-off of control air pressure from the pneumatic actuator opening chamber. This cycle modulates the butterfly valve to control downstream pressure.

Downstream pressure is also sensed through the internal downstream air pressure sensing port and transmitted to the normally closed overpressure limiter. In the event of malfunction of the servo assembly, the resulting high pressure acts on the overpressure limiter piston, overcoming the spring force and unseating the ball valve. This vents the pneumatic actuator opening chamber to the atmosphere. When opening chamber pressure drops, the actuator spring moves the actuator piston, rotating the butterfly valve toward the closed position and reducing the flow of air through the unit.

When the butterfly valve rotates more than 6° from the full closed position, the position indicator switch is actuated, indicating an open butterfly valve. As the solenoid becomes de-energized, control air passes through the switcher valve to the pneumatic actuator closing chamber. At the same time, the pneumatic actuator opening chamber is vented to the atmosphere through the switcher valve. The force of the control air pressure in the

closing chamber and the piston spring then moves the butterfly valve to the closed position.

3. Controls and Indications:

A. Circuit Breakers (CBs):

The engine starting system is protected by the following CBs:

Circuit Breaker Name:	CB Panel:	Location:	Power Source:
ENGINE START	P	K-6	ESS 28V DC Bus
WARN LTS PWR #3	P	F-2	ESS 28V DC Bus
WARN LTS PWR #5	P	F-5	ESS 28V DC Bus
WARN LTS PWR #6	P	G-1 or G-2 **(1)**	ESS 28V DC Bus
WARN LTS PWR #7	P	G-2 or G-3 **(1)**	ESS 28V DC Bus
OVHD ANN LTS PWR #1	P	A-1 or C-1 **(1)**	ESS 28V DC Bus
OVHD ANN LTS PWR #2	P	A-2 or C-2 **(1)**	ESS 28V DC Bus
OVHD ANN LTS PWR #3	P	A-3 or C-3 **(1)**	ESS 28V DC Bus

NOTE(S):

(1) Depending on effectivity.

4. Limitations:

A. Flight Manual Limitations:

(1) Starter Duty Cycle:

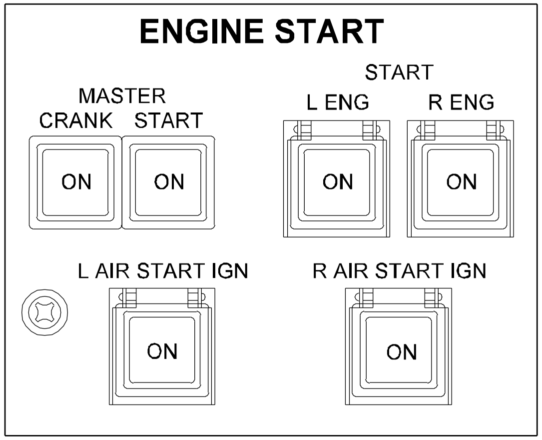
Continued use of starter is limited to three (3) crank cycles with a maximum of thirty (30) seconds per cycle. Delay three (3) minutes between start attempts. After three (3) cycles, delay use of starter for at least fifteen (15) minutes.

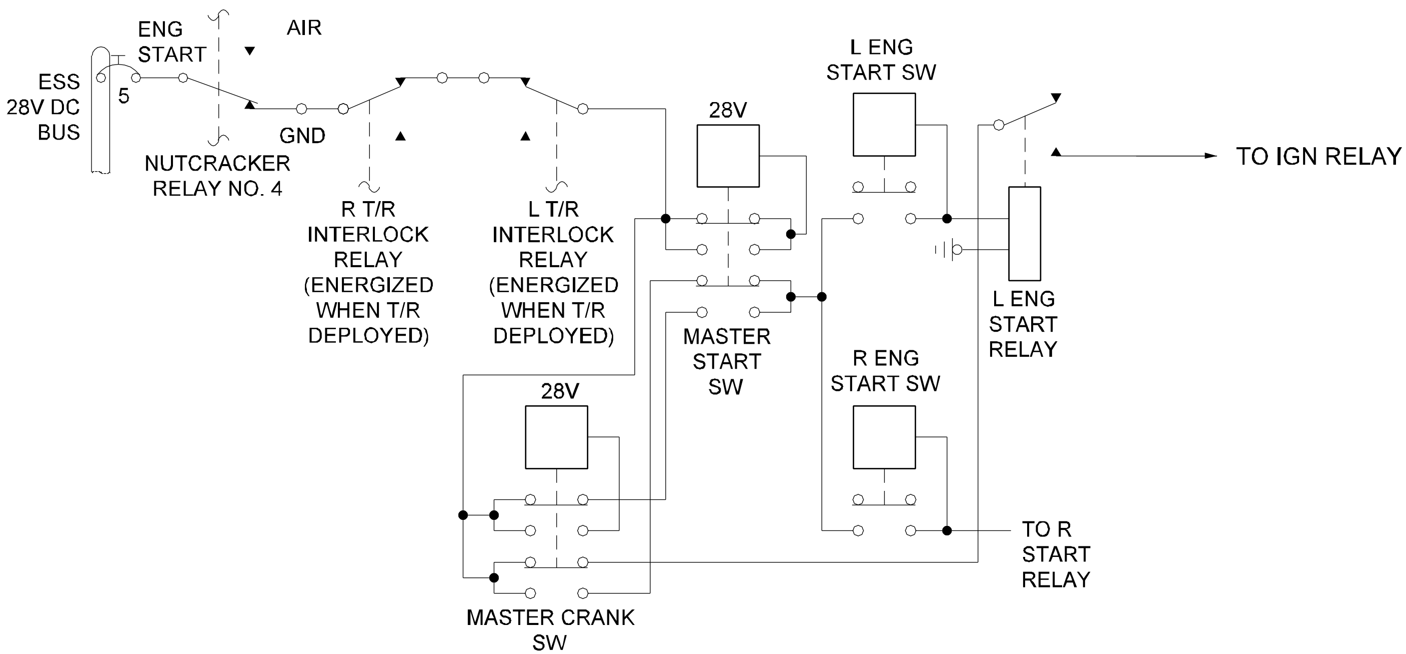
(2) Starter Re-engagement:

The starter may be re-engaged at HP RPM speeds up to starter cutout of 42% HP RPM.

(3) Airstart Envelope:

See Figure 3: Airstart Envelope.





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Engine Start System
Simplified Block Diagram
Figure 1

MASTER CRANK

OFF:
- Blue ON legend is extinguished
- De-energizes start relay
- Automatically closes isolation valve (on ground only)

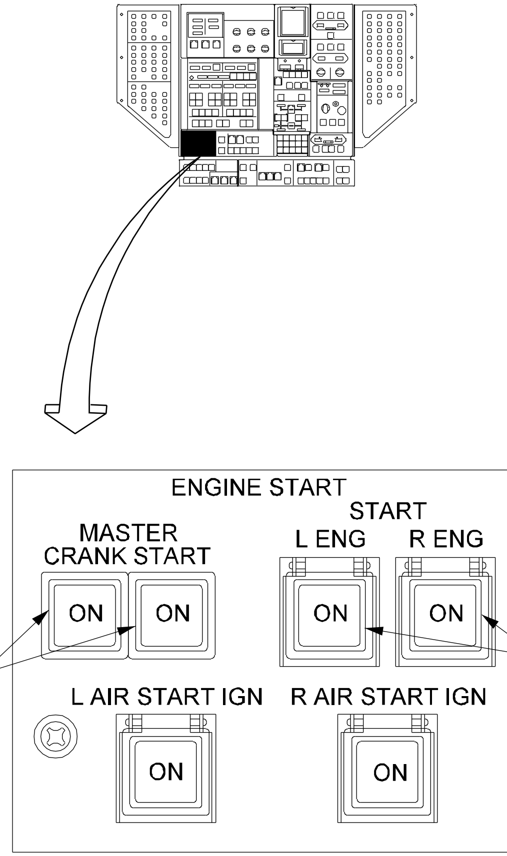
ON:
- Blue ON legend illuminates
- Allows the START switches to energize the start relay without energizing the auto-ignition relay
- Allows for cranking of engine without ignition or starting of engine
- Automatically opens isolation valve (on ground only)

MASTER START

OFF:
- Blue ON legend is extinguished
- De-energizes start and auto-ignition relay
- Automatically closes isolation valve (on ground only)

ON:
- Blue ON legend illuminates
- Allows the START switches to energize the start relay and the auto-ignition relay
- Allows for cranking of engine without ignition or starting of engine
- Supplies ignition when starter is selected
- Automatically opens isolation valve (on ground only)

NOTE: For aircraft 1000 through 1155 (excluding 1034) having ASC 135, aircraft 1034, and aircraft 1156 and subsequent, selection of the MASTER CRANK or MASTER START switch closes the LEFT ECS PACK valve (on ground only). The valve will automatically reopen when the MASTER CRANK or MASTER START switch is deselected.



L/R ENG START

OFF:
- Blue ON legend is extinguished
- De-energizes start relay
  - Can be used to manually terminate power that energizes start relay (in lieu of automatic power termination at 42% HP RPM)

ON:
- Blue ON legend illuminates
- Start relay energized

NOTE: For aircraft 1000 through 1155 (excluding 1034) having ASC 135, aircraft 1034, and aircraft 1156 and subsequent, selection of the L ENG START or R ENG START switch closes the RIGHT ECS PACK valve (on ground only). The valve will automatically reopen when the start valve closes.

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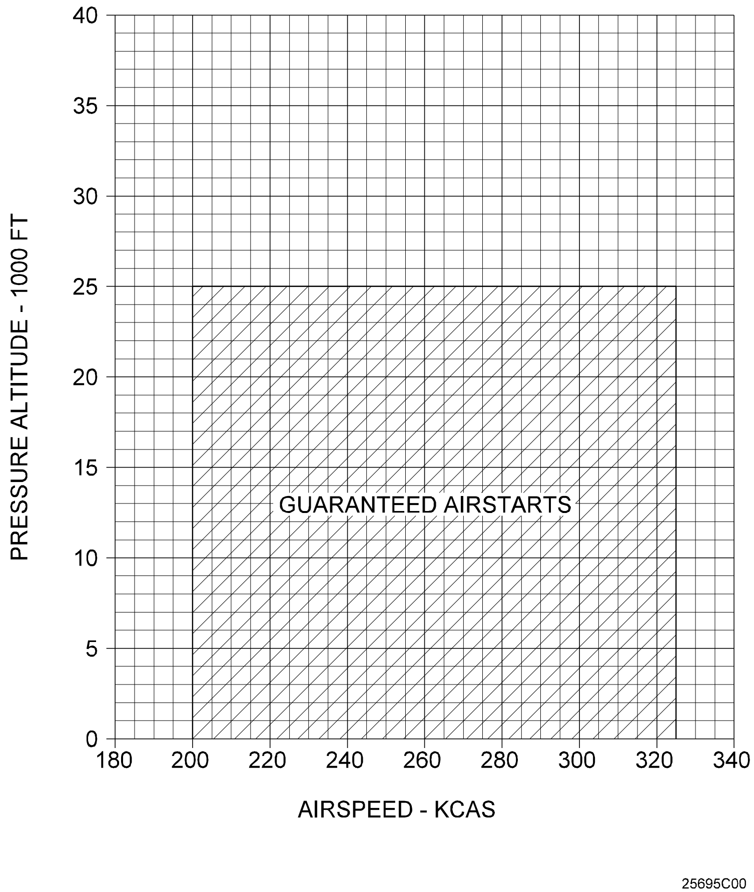
Engine Start System
Controls
Figure 2



Airstart Envelope
Figure 3

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CHAPTER 2A − PRODUCTION AIRCRAFT SYSTEMS
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